DESIGN STUDY

of

SUPERSONIC ALL-WEATHER INTERCEPTOR AIRCRAFT

A. V. ROE CANADA LIMITED

SECRET

DESIGN STUDY

of

SUPERSONIC ALL-WEATHER

INTERCEPTOR AIRCRAFT

IN ACCORDANCE WITH

RCAF SPEC. AIR-7-3

REPORT NO. P/C-105/1

MAY 1953

A. V. ROE CANADA LIMITED

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APPENDICES

1.0 INTRODUCTION

1.1 General

In April 1953 the R.C.A.F. issued Specification AIR 7–3[1], "Design Studies of Prototype Supersonic All-Weather Aircraft", to A.V.Roe Canada for the purpose of selecting the optimum aircraft capable of meeting R.C.A.F. Operational Requirement OR 1/1–63[2] "Supersonic All-Weather Interceptor Aircraft".

However, following the conveyance by the R.C.A.F. to A.V.Roe Canada of the recommendations[3] contained in the "Final Report of the All-Weather Interceptor Requirements Team of March 1952, A.V.Roe Canada submitted two brochures to the R.C.A.F. in June 1952. These described in very considerable detail two separate proposals; one for a single engined aircraft[4], the C 104/1; the other for a twin engined aircraft[5], the C 104/2. Both of these were intended to meet the conditions laid down by the Requirements Team. The advantages and disadvantages of these proposals were discussed in the brochures and at several meetings with the R.C.A.F.

The general consensus of opinion among the R.C.A.F. seemed to be in favor of the twin engined proposal. Accordingly A.V.Roe Canada continued its studies of this proposal and has investigated general refinements which make it possible to offer a performance that can easily exceed the original requirements in all respects, whereas the aircraft described in the C 104/2 brochure was deficient in some respects.

When, in AIR 7–3, the R.C.A.F. confirmed their preference for a twin engined proposal, it became evident that the experience gained by A.V.Roe in studying this type of configuration for the past year would be applicable, and could be drawn on to produce most of the data required by the design study called for in AIR 7–3 almost immediately.

Accordingly an R.C.A.F. Team visited A.V.Roe from April 27 to 30, 1953[6] to elucidate the requirements underlying AIR 7–3, and to discuss the results of the Avro studies which had a bearing on this Specification. Since the new requirements are really only an elaboration of the draft requirements[3] to which Avro had been working for more than a year, it was found possible to answer most of the questions raised by the R.C.A.F. on the spot and to produce a preliminary draft of this report[7] which is submitted in compliance with AIR 7–3.

1.2 Object of the Design Study

The R.C.A.F. team made it clear that they wanted to determine the absolute minimum size of airplane that would just meet their Specification. If there were any penalties or risks involved in doing this, they wanted to evaluate these against the gains to be achieved by more generous configurations. The R.C.A.F. studies had indicated that performance in excess of their requirements was of very little use, so that every effort should be directed to getting the lightest and hence cheapest aircraft that would do this job. Since the Avro proposals exceeded the requirements in everything except altitude performance, it was assumed that a considerable weight saving could be achieved by just meeting the requirements. This view was set forth in R.C.A.F. Report DDA 12[8].

1.3 Method of Presentotion

In order to meet these objectives, it is necessaryto establish that the basic configuration is potentially the lightest and best for the job, and then to compare a family of aircraft of this configuration, all designed to just meet the Specification. In this comparison, the effect of fitting engines made by three different manufacturers is included. Also some general data which apply to all aircraft in the family are required to complete the picture.

2.0 BASIC CONFIGURATION

2.1 Wing

To achieve supersonic speeds in level flight by means of turbo-jet engines with afterburning, it is essential that the supersonic drag be reduced to the absolute minimum possible. This requires the use of the lowest t/c ratio wing that is technically possible. Now, Convair have made several design studies[9] which show that the weight per square foot of a delta wing is practically independent of the t/c ratio down to a t/c of 3%. The Convair curve is reproduced as Fig. 1 of this report. Weights estimated at Avro from scantlings obtained by using methods involving an elaboration of NACA TN2232[10] and which requires the solution of 30 simultaneous equations on I.B.M. machines have resulted in similar conclusions. The comparison of conventional swept wings with delta wings on Fig. 1 shows that there is no doubt that the delta configuration is by far the lightest, for low t/c ratios.

Due to the large root chord of a very thin delta, the absolute thickness is still adequate to provide room for the stowage of the necessary fuel and undercarriage. It can be seen from the drag breakdown given as Fig. 21 of the C104/2 Brochure[5] that the drag of the fuselage is such that any unnecessary increase in its size to provide for the stowage of these items would increase the total drag very materially and hence add to the fuel load.

The reason for resorting to a tailless configuration is that for a highly swept low aspect ratio layout there is really no place where a tail can advantageously be located. If the tail is directly behind the wing it either restricts the high ground angle required with a low aspect ratio delta wing or results in an excessively long and heavy undercarriage. If the tail is moved up higher it is very difficult, if not impossible, to support it on a very thin fin. Also the large increase in downwash at the stall renders it strongly destabilizing so that the stalling characteristics are objectionable. The Gloster Javelin being subsonic, has a thick enough fin to support a tail, but does not avoid the considerable limitations imposed by a poor performance at the stall.

In order to increase the moment arm of the control surfaces and hence reduce the high drag of the elevators, some studies of canard configurations have been made, both by Avro and and the N.A.E. These have not proved very fruitful in showing any advantage sufficient to warrant further investigation. A prohibitive reduction in low speed $C_{L_{max}}$ with only moderate static margins is only one of the many difficulties with this configuration.

Having decided that a tailless design is the lightest and most efficient, it is necessary to choose an apex angle sufficiently high to give adequate damping[12] in the transonic region. This requires that the apex be about 60°. The difficulties that have been encountered by tailless airplanes employing less than this amount of sweep are too well known to require discussion

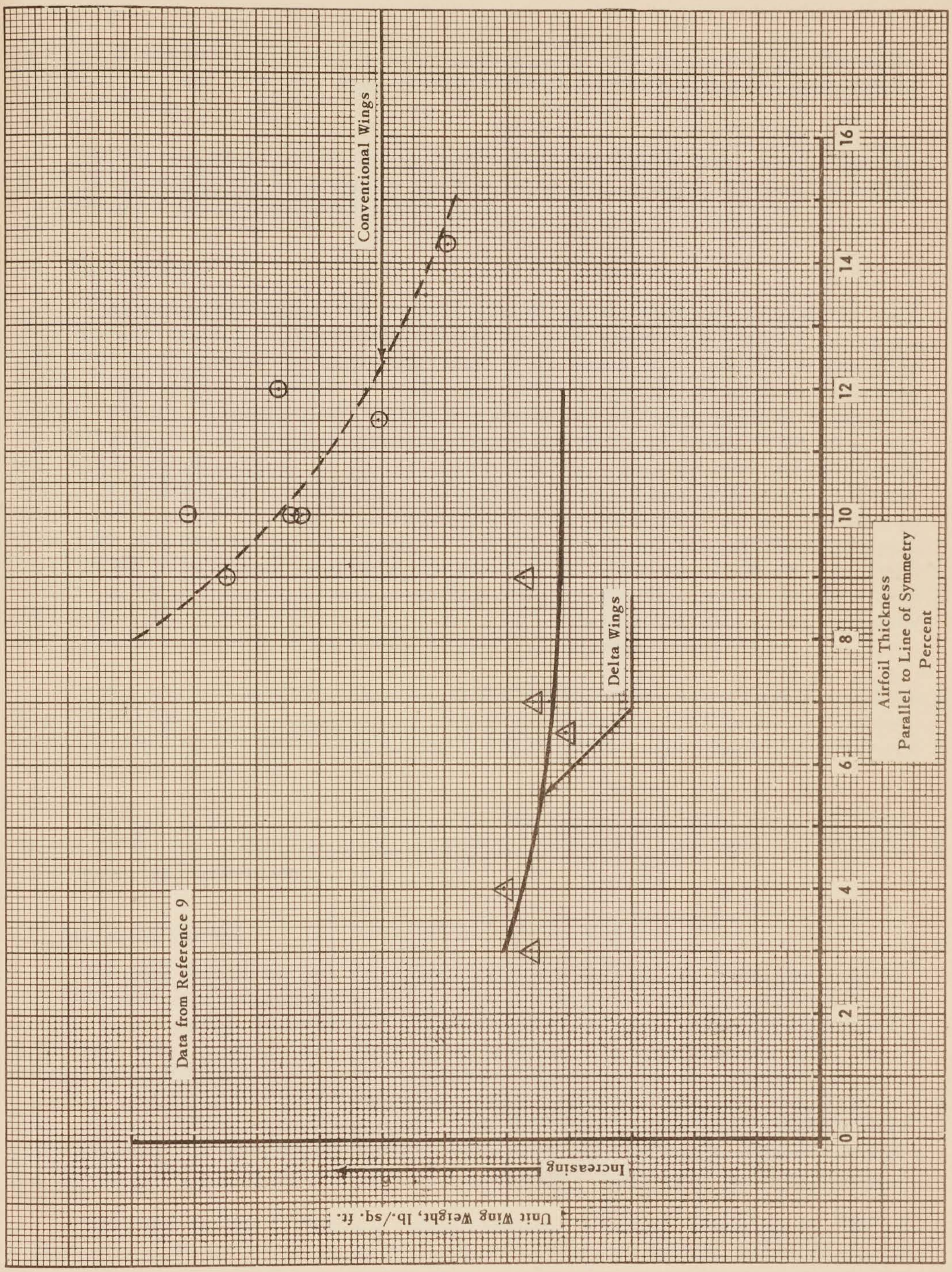

Fig. 1 Wing Weight as a Function of Airfoil Thickness
Comparison between Delta & Conventional Planform

here. It is sufficient to say that the damping of very large rocket propelled 60° delta models have been measured[13] in free flight and have exhibited satisfactory characteristics over the whole Mach range.

Having established that the optimum configuration is a tailless delta with a t/c as close to 3% as is possible with due regard for the room required for stowages, it remains to examine the effect of the various installations on the design.

2.2 Undercarriage

Both theory and tests on the Avro 707 indicate that the static ground angle for a 60° delta should be about 17°. This requires a relatively long undercarriage. In order to secure a reasonable width of track the upper pivot points of the legs must be outboard on the wings. Folding backwards is impossible in a thin wing, since it will cut through most of the wing bending structure. Therefore inward retraction is necessary. If the wheels are to be housed in the fuselage a low wing arrangement is necessary. If the wheels are housed in the wing a high wing arrangement is possible. This has the advantage that when the main undercarriage is clear of the fuselage, the accessibility and flexibility of installation of both engines and armament is greatly improved. Since the engine accessories are normally on the bottom, it is possible to carry the main wing box straight through the fuselage with the high wing arrangement and still have virtually perfect access for servicing the engines from underneath. On the other hand with a low wing, either very poor engine accessibility is achieved or the main box is reduced to a multiple spar construction underneath the engine. This lowers the efficiency of the wing structure so that its extra weight is greater than that saved by the simpler undercarriage.

Using data representative of the Convair F 102 multispar low wing construction and the Avro C 104/2 high wing construction the saving in wing weight is 3,500 lb. for the high wing version as against a loss on the undercarriage of 350 lb. giving a net saving of 3,150 lb. for an aircraft similar to the C 104/2.

Although somewhat more complicated, the undercarriage installation for the high wing airplane results in a lower gross weight, and gives considerably better access and flexibility to the engine and armament bays.

2.3 Engine Installation

The high wing layout with the engines slung from the wing and covered by large non-structural doors as shown in figure 26 of the C 104/2 brochure[5] is ideal for service and maintenance. It also permits the installation of different makes of engines with a minimum of rework to the basic airframe. In this case, any accessories that come in awkward places can be accommodated by small bulges in non-structural fairings. This feature is also especially important when it is considered that none of the engines under consideration have even been run at this date. There are bound to be modifications during the course of development, some of which would undoubtably be embarrassing to a tight installation, and would cause excessive delay in adapting the airframe, or might even result in a non standard engine detail becoming necessary.

With a low wing installation it is virtually essential to have the fuselage surrounding the engine stress carrying, in order to provide torsional stiffness for the wing and to support the fin. Engine removal must then be through stress carrying doors or out the rear end of the fuselage. Of these two methods, the latter is probably preferable. It has the disadvantage however that any part or accessory that falls outside the basic envelope must be made clear at all points on the withdrawal path. It should be regarded as purely fortuitous, if an installation of this kind involving engines still in the design stage, escaped without major structural fouls during the course of development.

Hence there is no reason to dispute the advantage of the high wing arrangement as far as the engine installation is concerned.

2.4 Armament

The most promising fire control and armament configuration for this fighter appears to be the Hughes MX 1179 system together with 6 Falcon guided missiles and 50 – 2" diameter folding fin rockets. It may be necessary to substitute other equipment if this does not work out as planned. The major design studies are however based on the assumption that this system will be fitted. The installation of externally stowed missiles is commented on later in this report.

The method of installation of the electronic equipment, that is easiest to design and maintain in service is believed to be where all this equipment is mounted in a crate as shown in Fig.32 of the C104/2 brochure[5]. In the larger versions of the aircraft studied in this report it is possible to adopt this configuration with the fuselage envelope required for balance. On some of the smaller versions it is necessary to compress the fuselage to such an extent that the electronic equipment must be spread out along the lower corners of the fuselage. This gives a much more complicated wiring and air conditioning problem, and adds about 150 lb. to the weight.

The internally stowed guided missiles are lowered on swinging arms. Light doors are arranged to open by means of a linkage while the missiles are being extended and closed when they are fully extended. This will give considerably less interference to the airflow during firing than if the doors remained open.

For the larger versions, the missiles are arranged in two rows with two abreast in front and four abreast behind. This gives greater freedom for sequencing the firing ripple, than the arrangement of two rows of three missiles as is required to compress the fuselage for the smaller versions.

The 2" diameter rockets will be housed in an extensible elevator similar to that being designed for the C 100 MK 4, where possible.

2.5 Radome and Cockpit

The MX 1179 or any other equivalent system requires the introduction of accurate air data in several computations. It has been concluded by Hughes that the only place to sense these data to the required degree of accuracy on an aircraft of this type is at the end of a nose mounted boom. Experience on the C 100 with the air data problem leads us to concur in this view. They also concluded that, for supersonic speeds, the radome should be moderately pointed. Accordingly Hughes have laid down a contour that is a compromise between the aero-dynamic and radiation requirements and is suitable for the mounting of a nose-boom. A relatively long term development program has been laid down for the particular contours decided upon for the Convair F 102 and other aircraft. These contours are being used for all these studies.

Having, of necessity, put the radome in the front to give it an adequate field of view, the pilot can most readily be located in a conventional cockpit behind the radome with a canopy which gives him a view over the radome. In order to simplify the problem of glazing this canopy, the optical surfaces have been constructed of two flats formed into a wedge. This makes it possible to use flat glass panels which are best suited to resisting the higher temperatures and pressures encountered on these designs.

2.6 Camber

Camber has been proposed as a means of reducing the elevator drag at high altitudes. A saving of 1,000 lb. of fuel to complete the specified missions and an increase in the ceiling of about 5,000 ft. are the order of the gains that are hoped for.

The following is an explanation of the way these improvements are achieved. With no camber or effective C_{M_0} the elevator angles to trim are always up, as shown in Fig. 3. If the wing is cambered, a couple is produced which causes the elevator angle to trim to be zero under any selected condition depending on the amount of camber. An example is shown as Fig. 4. Since the elevator drag is proportional to the square of the deflection, it can be seen that there will be a marked reduction in the elevator drag at high altitudes. Thus at a Mach number of 1.5 at 50,000 ft., the elevator angle without camber is 7.3°, while with the camber assumed in Fig. 4 it is 3.0°. This results in the elevator drag associated with the cambered wing being only 17% of the elevator drag of the uncambered wing for this case. The saving due to camber for other conditions can be found by comparing Fig. 5 with Fig. 2.

The difficulty with camber is in predicting and controlling the conditions occurring in the transonic region at low altitudes. Here relatively high down elevator angles are associated with a high dynamic pressure to give excessively high hinge moments required to trim. Now there is a very considerable difficulty in estimating the vagaries of the various derivatives that go to make up the trim angles in the transonic regime. The uncertainty in the Mach effect on the camber effectiveness at transonic speeds is shown in Fig. 6, where the C_{M_0} due to camber is estimated by two methods. The effect on the elevator hinge moment of these assumptions regarding C_{M_0}, and of the selection of the conditions under which the elevator angle for trim is zero is shown in Fig. 7. It can be seen that a limitation of the airplane to Mach number below about .95 at low altitudes is exceedingly likely. The higher the peak hinge moment, the higher the altitude at which the limitation is removed. The effect on the flight envelopes is shown on Figs. 8, 9, 10 and 11.

The incorporation of camber into the design requires a nice compromise between the gains at high altitude and the limitations at low altitude, based on an accurate knowledge of the aerodynamic properties of camber in the transonic regime. Since data on this point are virtually non-existent, wind tunnel tests have been scheduled, as required by para 12.01.01 of R.C.A.F. Spec. AIR 7-3 in the 4'0 x 5'0 transonic throat of the wind tunnel at the Cornell Aeronautical Laboratories, Inc. It is felt that the size and freedom from shock reflection problems make this throat the best facility available for this work. In fact, in view of the dubious reliability of virtually all other techniques suitable for measuring zero intercepts at transonic speeds, it is probably the only facility where this work could be done in a satisfactory manner.

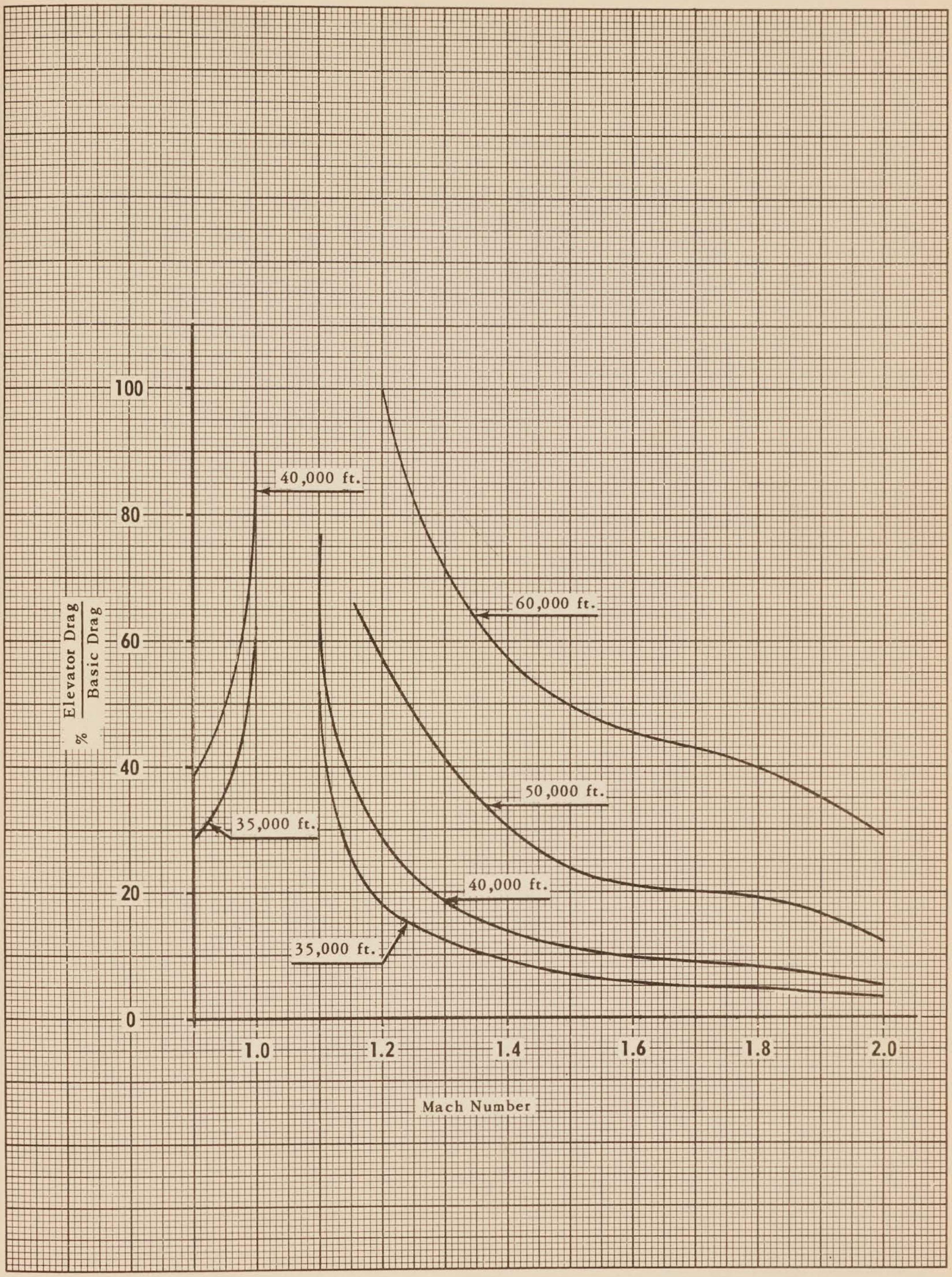

Fig. 2 $\frac{\text{Elevator Drag}}{\text{Basic Drag}}$ in Trimmed Flight

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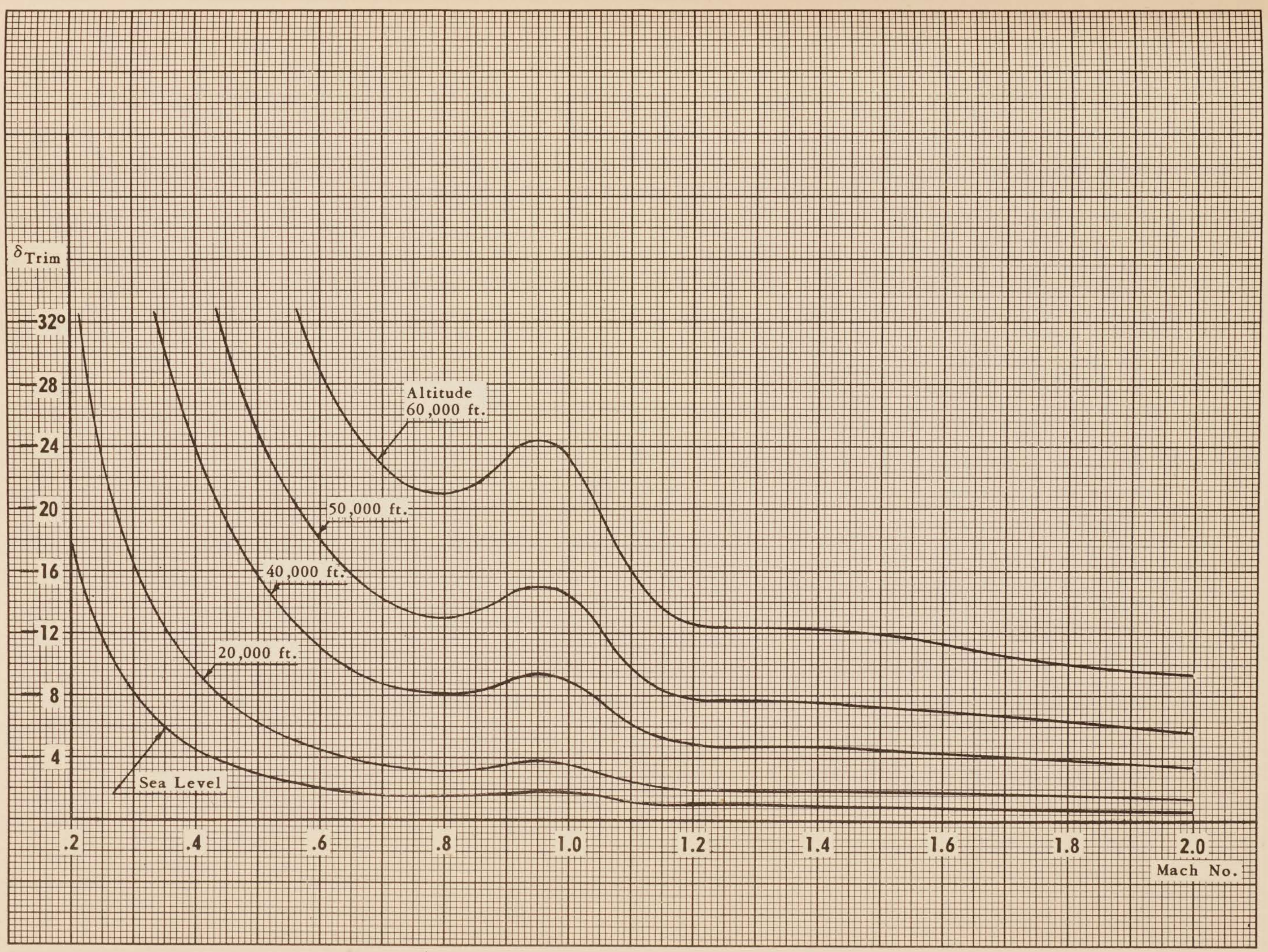

Fig. 3 Elevator Angle to Trim – No Camber

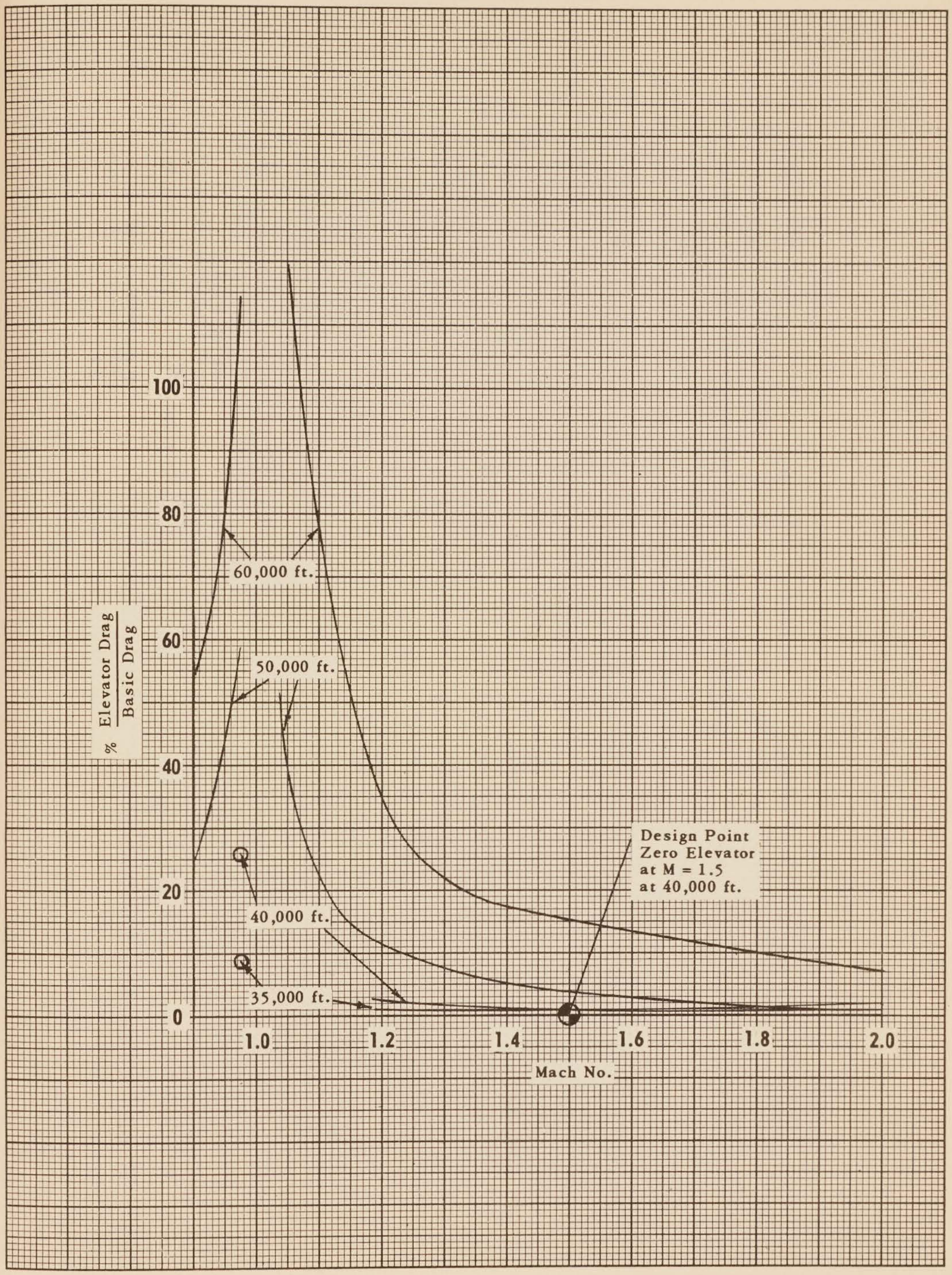

Fig. 5 0.5% Camber – % $\frac{\text{Elevator Drag}}{\text{Basic Drag}}$

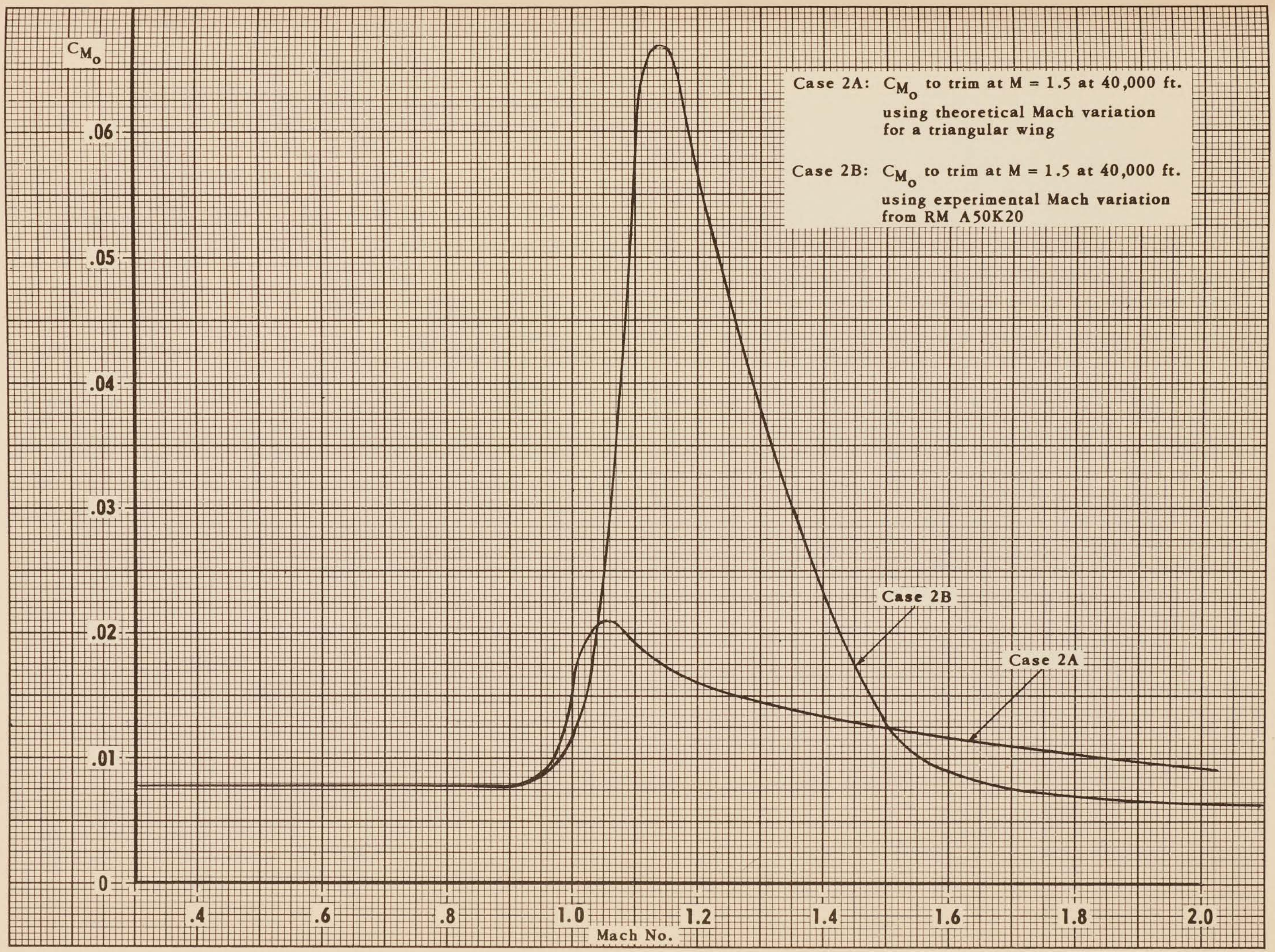

Fig. 6 C_{M_o} vs Mach No.

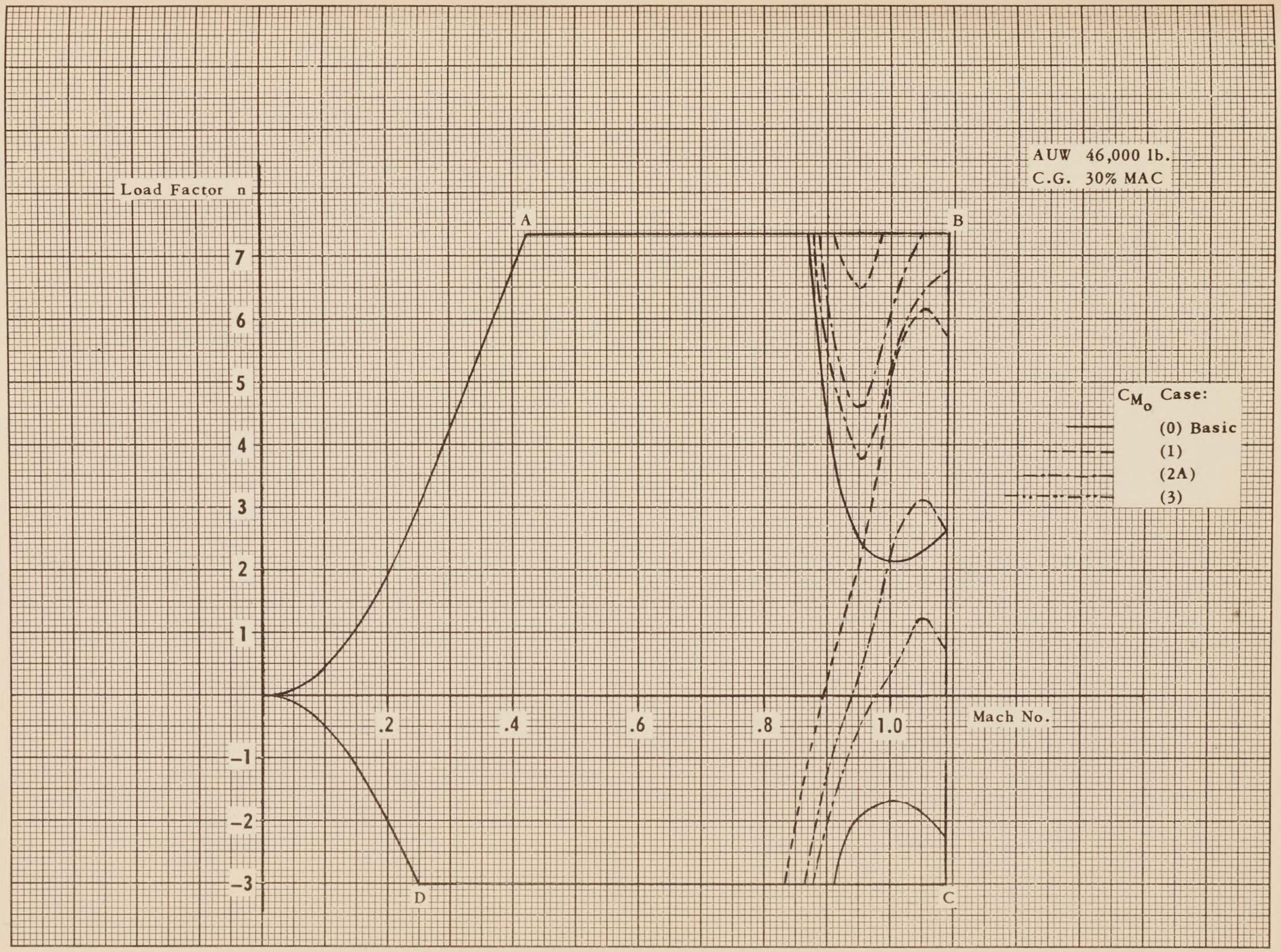

Fig. 8 Flight Envelope Limitations Variation with C_{M_o}
Sea Level

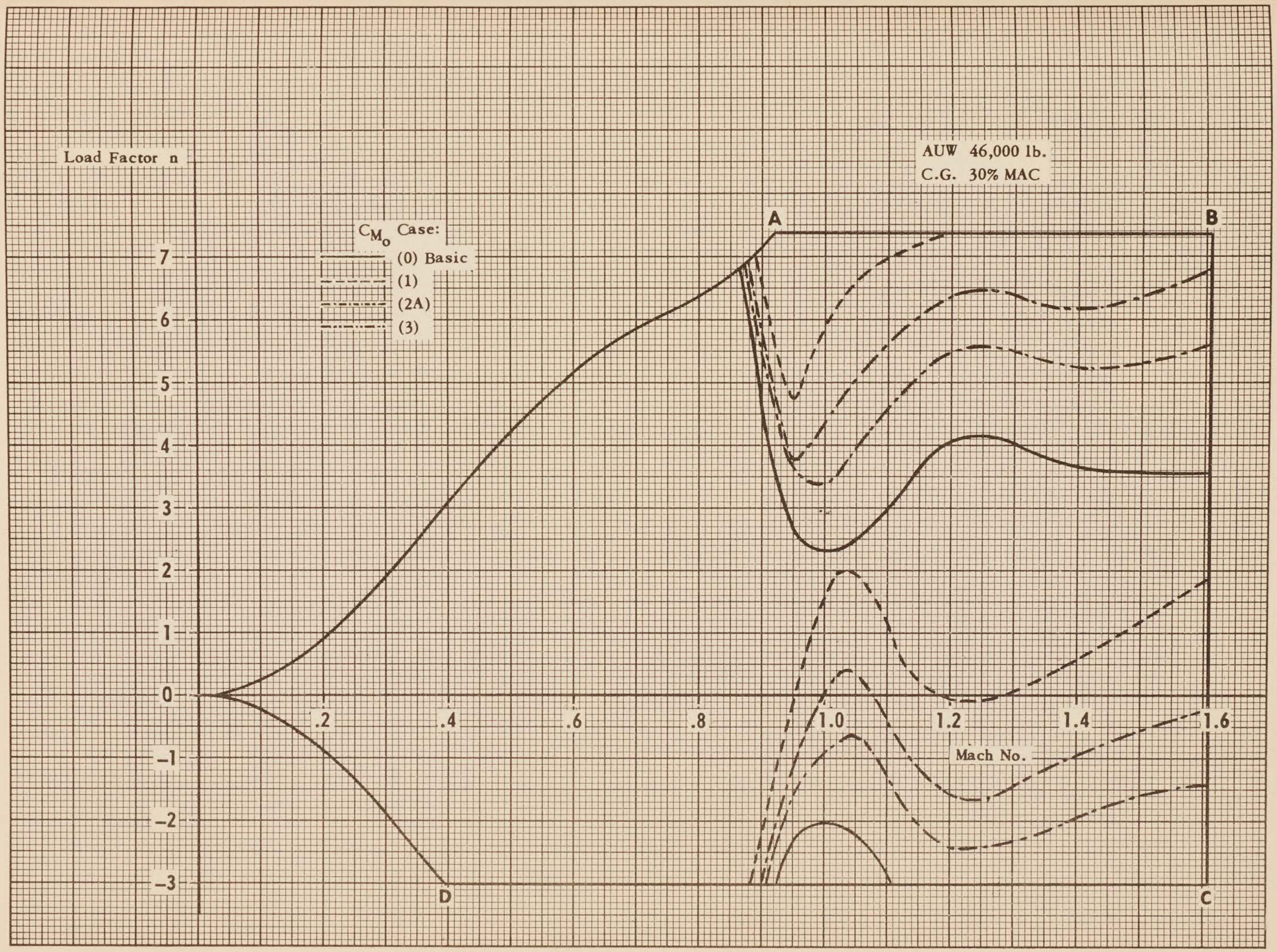

Fig. 9 Flight Envelope Limitations
Variations with C_{M_o} 20,000 ft.

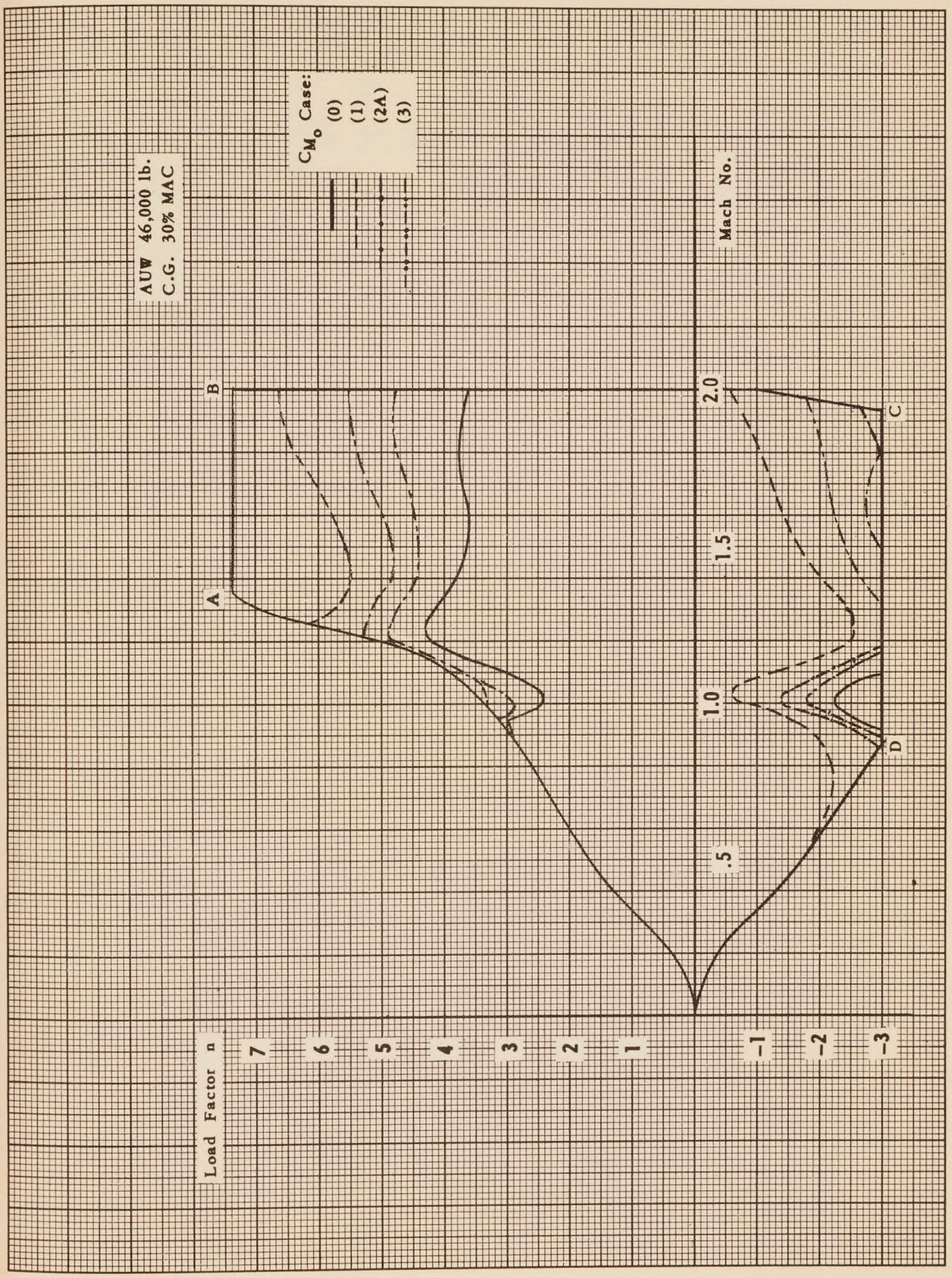

Fig. 10 Flight Envelope Limitations – Variation with C_{M_o}

40,000 ft.

3 COMPARISON OF ENGINES

3.1 Comparative Data

In Specification AIR 7–3, it is requested that proposals be made for three engines which are being made to very similar specifications by different manufacturers. A comparison of the basic data for these engines is shown in Table 1.

TABLE 1

	RB 106 SHORT INSTALLATION	RB106 LONG INSTALLATION	B.01.4	J.67
Thrust, S.L. Static, Military Rating (without A/B), lb.	15,000	15,000	14,100	13,200
Thrust, S.L. Static, Maximum (with A/B), lb.	24,000 at M = .6 at S.L.	24,000 at M = .6 at S.L.	21,000	21,500
Mass Flow, lb./sec.	226	226	212	225
Net Dry Weight, Engine (no A/B), lb.	3,786	3,606	3,600	–
Net Dry Weight, Engine and A/B, lb.	4,576	4,676	4,670	–
Installed Weight,* lb.	4,751	4,851	4,750	5,100
Mounting	3 Point	3 Point	4 Point	either 3 Point or 4 Point
Maximum Dimension for Installation. Engine Accessories Mounted, Width, in.	42	42	42	45
Depth, in.	40	40	46	48
Length, in.	217.0	255.5	250.0	250.3
Air Tapping Available, percent	5	5	5	5
Power Available from Gearbox, horsepower	250	250	150	–

* Installed weight does not include starter. Airesearch Air Turbine Starter is assumed carried as equipment. If liquid fuel starter is required add approximately 100 lb., the actual weight depending on the type of the starter and the number of starts required without ground servicing.

3.2 Comparative Performance

A comparison of the thrust and fuel consumption vs. speed for the three engines at 50,000 ft. is given on Fig. 12.

3.3 Discussion

3.3.1 Information Available

3.3.1.1 Rolls Royce RB 106

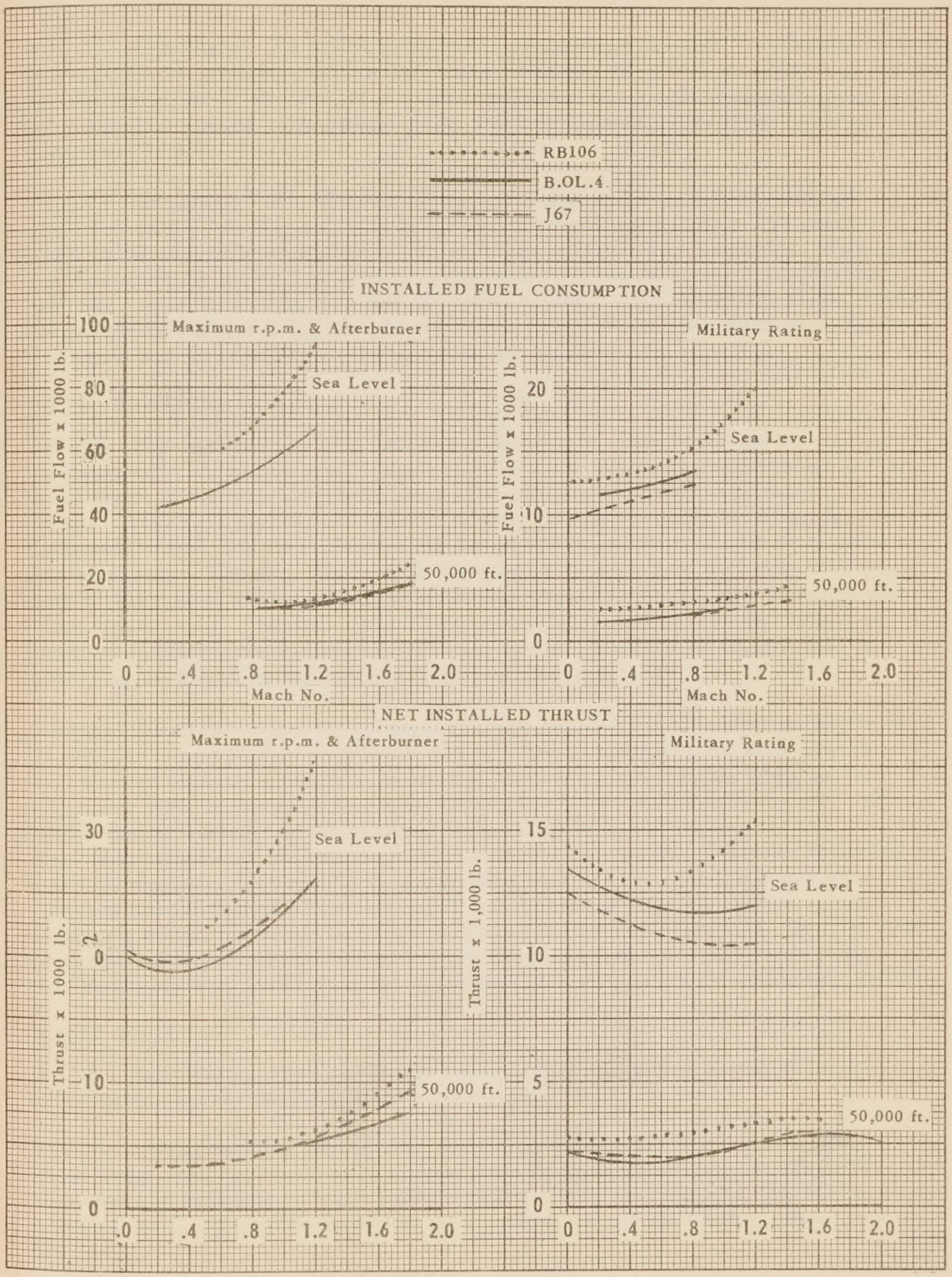

Fig. 12 Comparative Performance of Engines as Installed in C105/2, RB 106, J 67, & B.OL.4

3.3.1.1.1 Performance: All data is contained in a preliminary brochure [17]. Thrusts and fuel consumptions are given for combat and maximum continuous r.p.m. with no afterburning and for full afterburning only. Mr. Lombard of Rolls Royce stated that partial afterburning thrust could be arranged if required. Accordingly thrust and fnel consumption for these conditions were interpolated by Avro.

3.3.1.1.2 Installation Data: The data given for the bare engine on BT Sch. 8080 and for the afterburner on BT Sch. 8079 only give indications of some of the featnres. The nozzle is only shown in the most rudimentary fashion with no details of the method of actuation. While envelope dimensions are given, it is extremely donbtful if these can be accepted with confidence at this time, since it is understood that the engine is still in the preliminary stages of detail design. Accordingly, it is felt that the installation data given are more of the nature of targets that may or may not be achieved when the details are completed.

3.3.1.2 Bristol Olympus B.OL.4

3.3.1.2.1 Performance: Complete data on the performance of the bare engine is available in the brochure [18]. Additional data [19] has been given for the engine with afterburning. A conventional nozzle is assumed. At present Lucas are developing an afterburner for this engine at low priority. They have produced evidence [20] to show that they have the necessary fundamental background to deal with the combustion and control system problems, as well as, or better than any other group. However, they have not even seriously considered the design of a variable nozzle. They are at present thinking in terms of getting a license from Solar to build a conventional unfaired type of nozzle. Since this is no longer competitive with the Rolls Royce and Wright proposals for faired convergent-divergent nozzles, Bristol have admitted thay they may have to rely on Wrights for the development of a suitable nozzle.

3.3.1.2.2 Installation Data: A drawing of the bare engine is given in the brochure [18] which is sufficient for preliminary installation studies. Enough information for much more than this has not been furnished although it is possible that some additional data might be secured if necessary. Data on the afterburner is so rudimentary [20] as to be virtually useless. Although Bristol may be compelled to use a similar design of nozzle to that used on the J 67, other features of the afterburner are bound to vary considerably. Dr. Hooker of Bristols states that he would not cantilever the afterburner from the engine as done on the J 67, but would make a flexible joint between the afterburner and engine and arrange for a separate support system. This is probably because the engine has not been stressed for the afterburner loads. Since the details of an afterburner that could be used for this project are not even in the preliminary design stage, it is impossible to do anything beyond very general design studies of the installation problems.

3.3.1.3 Wright J 67

3.3.1.3.1 Performance: Complete figures on all aspects of the performance of the J 67 have been obtained from the Wright Aeronautical Corporation.

3.3.1.3.2 Installation Data: Reasonably complete installation data have been supplied. These wonld permit even the details of an airframe installation of both engine and afterburner to be proceeded with immediately. It also indicates that the design of mechanical details is in a very advanced stage. All the data appears to be very reasonable and the result of careful and extensive work.

3.3.2 Comments

3.3.2.1 Performance

3.3.2.1.1 Bare Engine: The performance of the B.OL.4 and the J 67 are very similar. This is not surprising, since the J 67 was based on the Olympus. The RB 106 is able to gain some advantage by overspeeding the low pressure parts at high forward speeds and altitudes. This feature is mainly one of developing a suitable control system, to achieve this. It is quite possible that similar systems might be adapted to either of the other engines.

3.3.2.1.2 Engine with Afterburner: The reason for the poor showing of the B.OL.4 is that the afterburner is assumed to have a simple nozzle as opposed to a convergent-divergent nozzle for both the J 67 and RB 106. The figures used for the B.OL.4 were given by Dr. Hooker of Bristols as recently as April 22, 1953. However, since that date, Bristols have stated that they might be able to obtain the design of nozzle worked out by Wrights under the terms of the agreement between the two companies for the exchange of technical information. Although no new figures have been received from Bristols, it is reasonably safe to assume that if they use the Wright afterburner, the performance of the two engines will be virtually identical.

The improvement in the afterburner performance of the RB 106 over the J 67 is probably due to the maximum temperature of the RB 106 being assumed as 2000°K while the J 67 only uses 1670°K. Since Rolls Royce are only using 1500°K at present they are quoting on an engine in a later stage of development than Wrights, who have, even now, a program to increase the thrust at least 12%. Since the temperatures used on the J 67 are relatively easy to achieve, there should be little difficulty in obtaining the figures claimed at an early date. However, since temperatures over 2000°K are not feasible, Rolls Royce may be regarded as having pushed their development to the limit with the figures they quote.

3.3.2.2 Installation: As stated above, the installation data furnished by the three manufacturers is very unequal. On only one of the engines, the J 67, is there enough information to do anything more than a preliminary design study of the engine-afterburner installation features. In view of this, it would be very rash to commit the lines of an airframe to a very tight installation for either the B.OL.4 or RB 106 at this time. It may not be entirely a coincidence that the J 67, on which most is known, takes up the most room. In any case, from the available data, it appears that the envelope designed for the J 67 will accommodate either of the other engines, with very little to spare in the case of the B.OL.4 and slightly more for the RB 106.

Since all authorities seem to be in reasonable agreement on the velocities that can be used in an afterburner, and all three engines have virtually the same mass flow, there would not seem to be much room for variations between the sizes of the afterburners required. As was found on the installation of the Solar afterburner on the C 103, provision for the nozzle actuating mechanism and for cooling airflow, enlarged the size required very considerably over what was originally thought to be adequate. Since full account is known to have been taken of all these features only on the J 67, it was agreed by the R.C.A.F.[6] that the envelope allowances for this afterburner should be assumed for all engines. With this proviso, the size required for all three engines becomes very similar, and there is little logic in not making allowance for all three. This policy is really the only possible one in any case, when one considers that not one of the engines have actually run yet.

4 COMPARISON OF AIRCRAFT

4.1 Preamble

Having previously established that the high wing delta layont is the preferable configuration, we shall now compare in detail a family of aircraft of varying sizes which are all designed to this configuration. Size will be varied by using wings of different area, keeping the aspect ratio and sweepback constant. Smaller wings require shorter fuselages for reasons of weight – balance. The fin and rudder area is largely determined by the one-engine-inoperative condition and, for similar thrust engines, may be kept the same for all aircraft considered in the family. The possible effect of fitting engines made by three different manufacturers on the size of the fuselage will be investigated. The size of fuselage and wing will of course influence the space available for internal fuel and also the installation of armament, avionics and fixed equipment. The length of the undercarriage is determined by the ground-angles required for landing and take-off, which are the same for all aircraft considered in the family, and the effect on the stowage problem of the retracted undercarriage in the wing must therefore be investigated. The size of the aircraft which fall in this family will therefore affect:

Weights
Performance
Installation Features

Each of these criteria will be analyzed and tabulated in snbsequent paragraphs of this chapter. The aircraft considered within this family are:

	Code No.	Fig. No.
High wing delta with 1000 sq. ft. wing area	C 105/1000	24
High wing delta with 1100 sq. ft. wing area	C 105/1100	25
High wing delta with 1200 sq. ft. wing area	C 105/1200	26
High wing delta with 1300 sq. ft. wing area	C 105/1300	27
High wing delta with 1400 sq. ft. wing area	C 105/1400	28

The Powerplants considered are:

Two Rolls Royce turbo-jet engines – RB 106 plus afterburners
Two Bristol turbo-jet engines – B.OL.4 plus afterburners
Two Cnrtiss-Wright turbo-jet engines – J 67 plus afterburners

In accordance with AIR 7-3 para 4.01.02 one aircraft in this family is shown converted to accommodate a crew of two, the 1200 sq. ft. version is chosen for this (code number C 105/1200/T shown in Fig. 29). The effects of such a conversion are discussed in subsequent paragraphs. Briefly it can be stated that any aircraft in the family can be converted by means of fitting a longer front-fuselage and the fitting of ballast as required; however, the relative effect on gross weight and performance is obviously more pronounced on the smaller aircraft.

In Appendix A of this design study an aircraft with a 900 sq. ft. wing area has been analyzed. This wing is too small for housing the nndercarriage in a high wing configuration and it is therefore necessary to adopt the low wing layout, with the undercarriage retracting sideways into the fuselage.

In Appendix B of this design study a delta aircraft of entirely different configuration is discussed and reasons given why such a layout is unsatisfactory.

4.2 Weights

4.2.1 Detailed weight calculations form the basis on which a weights comparison of all aircraft in the family are tabulated (Refer to Table 2). The aircraft weight is broken down into items which are grouped together under the following main headings:

Engines and afterburners
Powerplant items
Equipment including military load
Vertical tail structure
Fuselage structure
Wing structure
Undercarriage structure
Fuel (internally stowed)

The weight of engines and afterburners is of course determined by the engine manufacturer and for purpose of comparison of aircraft in this family the weight of the R.R. RB 106 engine has been used throughout Table 2; this weight has been taken from a Rolls Royce brochure[17]. Of the powerplant items, the weight of the fuel tanks is the only item which varies slightly for different size aircraft. The weight of fixed equipment including military load will remain the same for varying size of aircraft except for small variations in the weights of flying controls, hydraulic equipment which provides the source of power for the flying controls and undercarriage, and the avionic equipment. The weight of the latter item has been derived from information supplied by the Hughes Company and varies only depending on whether it is possible to group the majority of its items in one crate or whether it is necessary, due to fuselage space limitations, to spread them out along the sides of the fuselage. The latter arrangement requires extra length of wiring, and air conditioning ducting and is therefore somewhat heavier. The weight of the vertical tail structure will remain constant as explained before. The total weight of these first four main items will therefore remain very nearly constant for the various aircraft in the family and this is shown in the weight table. The weight of the remaining main items: fuselage, wing, undercarriage and internal fuel will vary with the size of the aircraft and its consequent gross weight. The weight estimation of these structural items is based on preliminary stress analysis and comparisons with other current and future aircraft. All aircraft in this family have been subject to static balance calculations in order to achieve the desired centre of gravity and, as an example, one such a calculation, i.e. for the 1200 sq. ft. version, has been included in this design study, refer to Table 3.

4.2.2 Weights Summary

TABLE 2

	C105/1000	C105/1100	C105/1200	C105/1300	C105/1400
ENGINE & AFTERBURNER	9,502	9,502	9,502	9,502	9,502
POWER PLANT FIXED ITEMS:					
Fuel Tanks	275	280	300	320	340
Fuel System	420	420	420	420	420
Fire Extinguishers	65	65	65	65	65
Accessory Gears & Drives	15	15	15	15	15
Engine Controls	20	20	20	20	20
GROUP TOTAL	795	800	820	840	860
(Cont'd)					

TABLE 2 (Cont'd)

	C105/1000	C105/1100	C105/1200	C105/1300	C105/1400
EQUIPMENT:					
Instruments	50	50	50	50	50
Probe	50	50	50	50	50
Surface Controls	675	685	700	715	725
Hydraulic System	660	670	680	690	700
Electrical System	700	700	700	700	700
Radar & Electronics	1,950	1,800	1,800	1,800	1,800
Ejector Seat	132	132	132	132	132
Emergency Provisions	15	15	15	15	15
Oxygen	20	20	20	20	20
Air Conditioning &	–	–	–	–	–
Low Pressure Pneumatics	625	625	625	625	625
Anti-Icing System	300	300	300	300	300
Brake Parachute	75	75	75	75	75
Exterior Finish	75	75	75	75	75
Crew	230	230	230	230	230
Oil	40	40	40	40	40
Residual Fuel	225	225	225	225	225
Armament Provisions	410	410	410	410	410
Armament – Rockets	520	520	520	520	520
Missiles	792	792	792	792	792
GROUP TOTAL	7,544	7,414	7,439	7,464	7,484
VERTICAL TAIL	900	900	900	900	900
SUB TOTAL	18,741	18,616	18,661	18,706	18,746
FUSELAGE	5,600	6,050	6,148	6,340	6,690
WING	7,870	8,095	8,557	8,879	9,049
UNDERCARRIAGE	2,129	2,139	2,109	2,150	2,200
OPERATIONAL WEIGHT EMPTY	34,340	34,900	35,475	36,075	36,685
FUEL FOR COMBAT MISSION	12,900	12,800	12,900	13,000	13,100
GROSS WEIGHT	47,240	47,700	48,375	49,075	49,785

4.2.3 Balance Calculations: The following horizontal balance calculations and centre of gravity positions are for the C 105 with 1200 sq. ft. wing area. The calculation is typical for the other aircraft.

Centre of gravity positions of the various items are located in feet aft of a vertical datum as shown on Fig. 13. The formula which converts these centre of gravity positions into percent of the mean aerodynamic chord of the 1200 sq. ft. wing is as follows:

$$\% \text{ M.A.C.} = \frac{A - 36.32}{30.22} \times 100$$

Where A is the centre of gravity position in feet aft of the nose datum.

TABLE 3

NO.	ITEM	WEIGHT IN POUNDS	ARM IN FEET	MOMENT IN FOOT POUNDS
	STRUCTURES:			
1	Wing	8,557	51.57	441,284.49
2	Tail	900	63.50	57,150.00
3	Body	6,148	34.70	213,335.60
	LANDING GEAR:			
	Retracted:			
4	Main gear inclnding jacks	1,710	44.40	75,924.00
5	Nose gear including jacks	375	13.22	4,957.50
6	Tail skid	24	64.40	1,545.60
	Extended:			
7	Main	1,710	48.20	82,422.00
8	Nose	375	17.15	6,431.25
	POWER PLANT & SERVICES			
9	Engine-Afterburner Units	9,502	56.27	534,677.54
10	Fuel Tanks	300	45.40	13,620.00
11	Fnel System	420	44.20	18,564.00
12	Fire Extinguishers	65	59.40	3,861.00
13	Accessory Gears	15	51.80	777.00
14	Engine Controls	20	21.30	426.00
	EQUIPMENT:			
15	Instruments	50	11.00	550.00
16	Probe	50	−2.45	−122.50
17	Surface Controls	700	54.10	37,870.00
18	Hydraulic System	680	48.15	32,742.00
19	Electrical System	700	21.80	15,260.00
20	Radar & Electronics	1,800	17.25	31,050.00
21	Armament Provisions	410	32.50	13,325.00
22	Ejector Seat	132	13.50	1,782.00
23	Emergency Provisions	15	12.90	193.50
24	Oxygen	20	15.50	310.00
25	Air Conditioning & Low Pressure Pneumatic System	625	24.50	15,312.50
26	Anti-Icing System	300	44.60	13,380.00
27	Brake Parachnte	75	65.60	4,920.00
28	Exterior Finish	75	39.85	2,988.75
	WEIGHT EMPTY – LANDING GEAR UP	33,668	45.61	1,535,683.98
	– LANDING GEAR DOWN		45.85	1,543,655.73
	NON-EXPENDABLE USEFUL LOAD:			
29	Crew (one pilot)	230	13.00	2,990.00
30	Oil	40	53.10	2,124.00
31	Residual Fnel	225	45.40	10,215.00
	GROSS WEIGHT LESS FUEL & ARMAMENT			
	– LANDING GEAR UP	34,163	45.40	1,551,012.98
	– LANDING GEAR DOWN	34,163	45.63	1,558,984.73
	EXPENDABLE USEFUL LOAD:			
32	Armament – Rockets	520	29.00	15,080.00
33	Missiles	792	34.75	27,522.00
	OPERATIONAL WEIGHT EMPTY:	35,475		
	– LANDING GEAR UP		44.92	1,593,614.98
	– LANDING GEAR DOWN		45.15	1,601,586.73
34	COMBAT MISSION FUEL	12,900	45.50	586,950.00
	GROSS WEIGHT – LANDING GEAR UP	48,375	45.08	2,180,564.98
	– LANDING GEAR DOWN		45.24	2,188,536.73

2

N NDS

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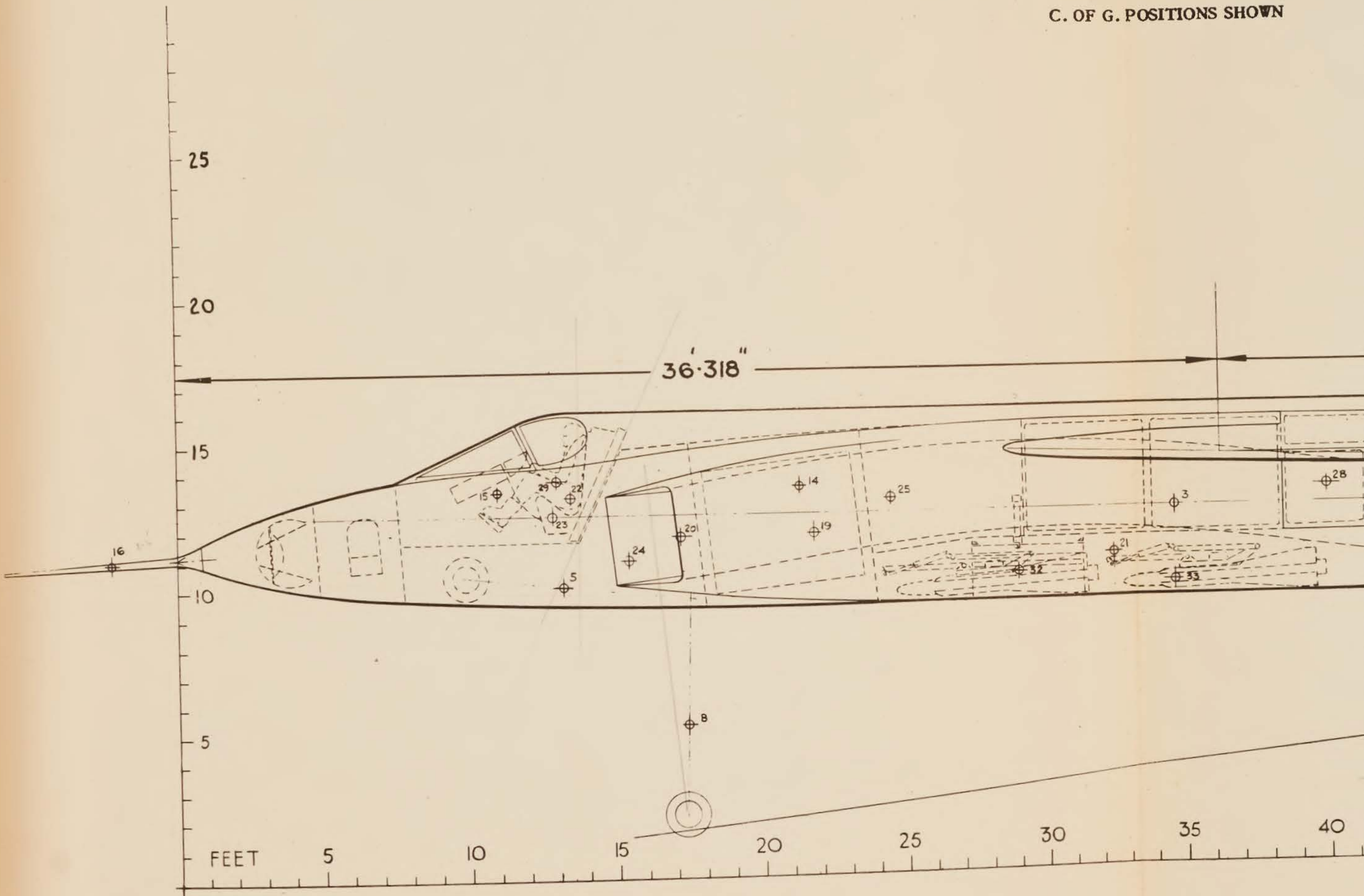
NOTE: EXTREME FORWARD AND AFT FLIG
C. OF G. POSITIONS SHOWN



CENTRE OF GRAVITY POSITIONS

(1) Design gross weight condition, landing gear down

$$\frac{45.24 - 36.32}{30.22} \times 100 = 29.52\%\ \text{M.A.C.}$$

(2) Design gross weight condition, landing gear up

$$\frac{45.08 - 36.32}{30.22} \times 100 = 28.99\%\ \text{M.A.C.}$$

(3) Furthest aft c.g.:
Design gross weight less fuel and expendable armament, undercarriage down

$$\frac{45.63 - 36.32}{30.22} \times 100 = 30.81\%\ \text{M.A.C.}$$

(4) Fnrthest forward c.g.:
Design gross weight less fuel, nndercarriage up

$$\frac{44.92 - 36.32}{30.22} \times 100 = 28.46\%\ \text{M.A.C.}$$

The estimated limits of centre of gravity travel as determined by aerodynamic requirements of stability and control are:

From 27% to 31% for the fighter versions
and
From 25% to 31% for the two seater version

4.3 Performance

4.3.1 Effect of Size: A comparison of the performance of a family of aircraft of varying size, as indicated by the wing area, is given in Table 4. For reference purposes, the fnll details of the three specified mission profiles are given in Tables 5, 6 and 7 for the 1,200 sq. ft. wing area aircraft which may be regarded as representative. The aircraft compared are all of the twin engine, high wing, delta configuration and are designed to carry the specified military load and engines. Since the purpose of this table is to compare airframes, all the data have been based on the same engine, namely the Rolls Royce RB 106.

The performance is considerably better than that given in the C 104/2 brochnre[5]. This is almost entirely due to the higher thrust that can be obtained from the engine and the fact that better fuel consumption with partial afterburning have been assumed on the strength of later data on this subject. Although the basic RB 106 engine has greater thrust than the engines previously assumed, the main increases can be attributed to the afterburner, which is assumed to operate at temperatures np to 2000°K and to have a convergent-divergent nozzle. At the higher Mach numbers, the nse of an adjnstable angle wedge ramp at the intake causes a large gain in the intake efficiency, by making nse of an oblique shock system in place of the normal shocks previously assumed.

The estimated drags are based on the same data as was used before, with the exception of the elevator drag, which is based on new evidence[16]. These drags are almost twice those formerly used.

TABLE 4

Performance & Weights Comparison of the C 105/2
With Gross Wing Areas of 1,400, 1,300, 1,200, 1,100, 1,000 Sq. Ft.
(2 RB 106 Engines, 3% t/c 0.5% Camber)
Armament Stowed Internally

GROSS WING AREA	SQ. FT.	1,400	1,300	1,200	1,100	1,000
GROSS WEIGHT	lb.	49,800	49,100	48,400	47,700	47,200
FUEL LB.	Supersonic Mission[1]	13,100	13,000	12,900	12,800	12,600
	Subsonic Mission[2]	13,100	12,800	12,700	12,800	12,900
	Long Range Mission[3]	18,800	19,300	20,300	21,500	23,000
INTERNAL FUEL CAPACITY	lb.	20,400	17,600	16,500	14,200	12,900
SIZE OF EXTERNAL TANKS REQUIRED	gals.	–	200	500	1,000	1,350
RANGE WITH 500 GAL. EXTERNAL TANK[4]	N.M.	1,500	1,500	1,500	1,200	920
COMBAT CEILING – FT. ½ FUEL WEIGHT	0.95 M.N.	56,500	56,100	55,100	54,100	52,900
	1.50 M.N.	66,000	65,300	65,100	64,900	64,300
	1.75 M.N.	68,600	68,300	67,800	67,300	66,900
TIME TO 50,000' FROM STANDING START	mins.	3.4	3.3	3.2	3.1	3.1
'g' AT 50,000' AT 1.5 M.N. AT ½ FUEL WEIGHT		2.15	2.14	2.14	2.09	2.00
LANDING DISTANCE FROM 50 FT. – FT. STANDARD DAY	½ Fuel Weight	5,130	5,410	5,630	6,100	6,610
	5 mins. Fuel Reserve Wt.	4,470	4,740	4,900	5,300	5,720
TAKE-OFF DISTANCE TO CLEAR 50 FT. – FT. HOT DAY	Overload Weight	2,830	3,060	3,360	3,800	4,600
	Gross Weight	2,280	2,380	2,440	2,650	2,850

NOTES:
1. Supersonic Mission as detailed in Table 5.
2. Subsonic Mission as detailed in Table 6.
3. Long Range Mission as detailed in Table 7.
4. 500 gallon external tanks appear to be the maximum permissible

Although the wing area of the largest aircraft is 40% greater than that of the smallest aircraft covered by this study, the gross weight increase is only 2,600 lb. or 5.5%. Thus as the size of the aircraft is increased, both the wing and span loading are decreased. This results in a very large reduction in landing distance as the wing area is increased. The altitude performance is only slightly improved on the larger aircraft, due to the drag of the extra wetted area compensating for the lower induced drag to a great extent.

Thus the only marked differences due to size are in the fuel capacities, which will be commented on in para 4.4.2 and the landing distances.

The estimates of landing distance are based on data obtained from tests on the Avro 707B using the "early touchdown technique" in which a tail parachute is streamed from the tail for braking. These data correlate very well with similar data for conventional aircraft when the square of the approach speed is used as a parameter. However the approach speed used during these tests was very high when compared with the stalling speed, causing the landing distances to be relatively very long when compared with those for conventional airplanes. This does not necessarily have any particular significance, since there was no effort made to make short landings in the test program on which data are available, and the runways used were always very much longer than necessary. Also the approach speed itself was quite low, due to the low wing loading of the test vehicle. For these reasons, the pilots had no incentive to find the minimum safe approach speed. They do however feel that it could be reduced considerably below what was used in these tests. Accordingly, on the strength of this, the estimated approach C_L s have been increased from the values of about 0.4 which were actually recorded in instrumented landings to 0.5. This has the effect of reducing the estimated landing distance by 20%. The value of 0.5 was chosen so that the C_L at touch down would still be low enough so that the aircraft would not be faced with any of the undesirable flying qualities which might be caused by the non-linear behaviour of some of the derivatives at very high incidence.

Although it is felt that the landing distances quoted can be obtained or even bettered by the use of a suitable landing technique, there is an element of risk in accepting them, insofar as the data on which they are based have not been fully substantiated. From the figures given in Table 4, it is evident that there is only any real risk in not meeting the specified landing distances, if the wing area is 1,100 sq. ft. or less.

TABLE 5

C 105/2 – 1,200 Sq. Ft. – 2 RB 106 Engines

Supersonic Mission

Combat Radius of Action = 200 N.M.
Gross Weight = 48,400 lb.
Fuel Weight = 12,900 lb.
Total Time to Combat = 18.4 mins.

	DISTANCE N.M.	TIME MINS.	FUEL CONS. LB.	A/C WEIGHT LB.
START	–	–	–	48,400
TAXI AND WARM UP	–	4.0	660	47,740
T.O., CLIMB AND ACCELERATION TO 50,000' AT 1.5 M.N., MAX. THRUST, AFTERBURNER LIT	39	3.2	3,740	44,000
CRUISE OUT AT 50,000' AT M = 1.5	161	11.2	2,230	41,770
COMBAT AT 50,000' M = 1.5	–	5.0	3,050	37,408*
DESCENT TO 35,000'	29	3.8	165	37,243
CRUISE BACK AT 35,000' AT ECONOMICAL CRUISE	107	11.8	935	36,308
STACK AT 35,000' MAX. ENDURANCE SPEED	–	15.0	875	35,433
DESCENT TO SEA LEVEL	64	7.8	710	34,723
APPROACH, MAX. ENDURANCE SPEED	–	5.0	535	34,188
TOTAL	400	66.8	12,900	

* 1,312 lb. ammunition fired

TABLE 6

C 105/2 – 1,200 Sq. Ft. – 2 RB 106 Engines

0.5% Camber

Subsonic Mission

Combat Radius of Action = 300 N.M.
Gross Weight = 48,200 lb.
Fuel Weight = 12,700 lb.
Total Time to Combat = 35.9 mins.

	DISTANCE N.M.	TIME MINS.	FUEL CONS. LB.	A/C WEIGHT LB.
START	–	–	–	48,200
TAXI AND WARM UP	–	4.0	660	47,540
T.O. AND CLIMB TO 35,000 FT. ECONOMICAL CLIMB	37	4.2	1,100	46,440
CRUISE OUT AT M = 0.95 ECONOMICAL CRUISE AT 35,000 FT.	240	26.0	2,400	44,040
ACCELERATE TO M = 1.5 AND CLIMB TO 50,000' MAX. THRUST, AFTERBURNERS LIT	23	1.7	1,375	42,665
COMBAT AT 50,000' M = 1.5, MAX. THRUST AFTERBURNER LIT	–	5.0	3,050	38,303*
DESCENT TO 35,000'	29	3.8	170	38,133
CRUISE BACK AT M = .95 ECONOMICAL CRUISE AT 35,000'	207	23.0	1,825	36,308
STACK AT 35,000' MAX. ENDURANCE	–	15.0	875	35,433
DESCENT TO SEA LEVEL	64	7.8	710	34,723
APPROACH MAX. ENDURANCE	–	5.0	535	34,188
TOTAL	600	95.5	12,700	

* 1,312 lb. ammunition fired

TABLE 7

C 105/2 – 1,200 Sq. Ft. – 2 RB 106 Engines

Long Range Mission

Overload Weight = 56,300 lb.
Fuel Weight = 20,300 lb.
Range = 1,500 N.M.

	DISTANCE N.M.	TIME MINS.	FUEL CONS. LB.	A/C WEIGHT LB.
START	–	–	–	56,300
WARM UP AND TAXI	–	4	660	55,640
TAKE-OFF AND CLIMB TO 35,000' ECONOMICAL CLIMB	42	4.7	1,405	54,235
CRUISE AT .95 M.N. AT 35,000' ECONOMICAL CRUISE	1,394	153	15,830	38,405
STACK AT 35,000' MAX. ENDURANCE	–	15	995	37,410
DESCENT TO SEA LEVEL	64	7.8	805	36,605
APPROACH MAX. ENDURANCE	–	5.0	605	36,000
TOTAL	1,500	189.5	20,300	

4.3.2 Effect of Engines: In order to compare the different engines which are the subject of this study, one airframe was selected from the family and the performance calculated for it using the three different engines, as shown on Table 8.

TABLE 8

Performance and Weights Comparison of C 105/2
With Gross Wing Area of 1,200 Sq. Ft. and
(2 RB 106, 2 J 67, and 2 B.OL.4 Engines, 3% t/c 0.5% Camber)

ENGINES			J 67	RB 106	B.OL.4
GROSS WEIGHT		lb.	48,100	48,400	48,000
FUEL LB.	Supersonic Mission[1]		11,900	12,900	12,500
	Subsonic Mission[2]		11,700	12,700	12,100
	Long Range Mission[3]		19,400	20,300	19,800
INTERNAL FUEL CAPACITY		lb.	16,500	16,500	16,500
SIZE OF EXTERNAL TANKS REQUIRED		gals.	400	500	450
RANGE WITH 500 GALS. EXTERNAL TANK[4]		N.M.	1,570	1,500	1,540
COMBAT CEILING FT. ½ FUEL WEIGHT	0.95	M.N.	54,100	55,100	52,500
	1.50	M.N.	64,000	65,100	62,500
	1.75	M.N.	66,500	67,800	63,500
TIME TO 50,000' FROM STANDING START		mins.	4.0	3.2	4.8
'g' AT 50,000' AT 1.5 M.N. AT ½ FUEL WEIGHT			1.91	2.14	1.76
LANDING DISTANCE FROM 50 Ft. – Ft. STANDARD DAY	½ Fuel Weight		5,650	5,630	5,600
	5 min. Fuel Reserve Wt.		4,980	4,900	4,900
TAKE-OFF DISTANCE TO CLEAR 50 Ft. – Ft. HOT DAY	Overload Weight		4,160	3,360	4,190
	Gross Weight		2,940	2,440	3,010

NOTES:
1. Supersonic Mission as detailed in Table 5
2. Subsonic Mission as detailed in Table 6
3. Long Range Mission as detailed in Table 7
4. 500 gallon external tanks appear to be maximum permissible.

As mentioned in section 3.3, the difference in the performance between the versions with the Bristol B.OL.4 and the Wright J 67 are entirely due to the type of nozzle used for the afterburner. If these two companies co-operate on their afterburner development, as they are already doing on the basic engine, there is no reason to believe that the performances would not be virtually identical.

Similarly, some of the extra performance of the RB 106 is due to assuming it to be developed to a higher degree than the other two engines.

4.3.3 Fuel Capacity

4.3.3.1 Internal Fuel Capacity: One of the most important differences in the five versions is in the margin of fuel capacity available for those forseeable contingencies which are likely to increase the fuel consumptions to such an extent that the specified radii of action would not be achieved in the final airplane unless the margin of fuel capacity were adequate. An assessment of this problem is given in Table 9. The following discussion is an appreciation of the factors that should be taken into consideration.

(1) Engine Fuel Consumption:

It is normal practice to add 5% to engine manufacturer's brochure figures for fuel consumption even for existing well tried engines. It would be very optimistic and unrealistic not to add at least this margin plus at least another 5% to the figures given for engines that have not even been run yet.

(2) Increase In Aircraft Weight:

Virtually no aircraft have been built that have not increased in weight over the preliminary design estimate. Since a growth of 10% in weight has in the past proved to be usually on the low side, it is not unreasonable to budget for an increase in fuel consumption due to this figure.

(3) Drag:

The estimates of drag, especially at supersonic speeds, could easily be in error by 10%. In fact, independent estimates made on the C 104, by various people varied by about that amount. Furthermore, the improvement due to camber which itself is of that order, has not at this time been substantiated by any experimental evidence which would justify making any commitments based on achieving the full amount of improvement that can be estimated theoretically. Accordingly it would seem reasonable to accept the possibility of a 10% increase in the estimated drag.

TABLE 9

Effect of Contingencies on Fuel Capacity Margin
and on Supersonic Mission Radii
of C 105/2 with Gross Wing Areas of
1,400, 1,300, 1,200, 1,100 & 1,000 Sq. Ft. (2 RB 106 Engines)

GROSS WING AREA – SQ. FT.		1,400	1,300	1,200	1,100		1,000	
t/c, WING THICKNESS TO CHORD RATIO	%	3	3	3	3	3½	3	4
ESTIMATED FUEL CAPACITY MARGIN	%	56	35	28	13	23	0	29
MARGIN IF FUEL CONSUMPTION IS 10% GREATER	%	41	23	16	3	12	–9	17
MARGIN IF SUPERSONIC DRAG IS 10% GREATER	%	48	29	22	7	17	–3	25
MARGIN IF WEIGHT IS 10% GREATER	%	44	25	18	5	14	–7	19
MARGIN IF ALL CONTINGENCIES OCCUR IN THE SUPERSONIC MISSION	%	25	9	3	–9	–1	–18	5
SUPERSONIC MISSION RADIUS WHEN ALL CONTINGENCIES OCCUR	N.M.	>200	>200	>200	158	195	96	>200

From Table 9, it is clear that with the 1,200 sq. ft. version there is an adequate margin of fuel capacity available to cater for these effects but in the 1,000 sq. ft. aircraft there is just enough fuel for the subsonic combat mission let alone provide a margin for contingencies. This means that some reduction in the radii of action of the smaller aircraft is virtually certain.

This situation can best be remedied by increasing the wing t/c for the 1,100 and 1,000 sq. ft. versions to 3½ and 4% respectively as shown by the table. It can be seen from Fig. 1 that this increase in wing thickness would result in virtually no saving of weight. Some reduction in performance below that given in Table 4 would however result. An allowance for this has been made in preparing Table 9.

Although thickening of the wing may seem like an adequate answer to the fuel capacity problem, it results not only in a reduced performance; e.g. the "g" available in a turn at 50,000 ft. is reduced 3% for the 1,100 sq. ft. version and 7% for the 1,000 sq. ft. version, but also it is felt that the chances of being able to take full advantage of camber deteriorate very rapidly as the thickness is increased. No very tangible systematic evidence can be offered in support of this, but it is generally appreciated that the wiggles in the derivatives that occur in the transonic regime become more severe and unpredictable the higher the t/c. Thus, with the relatively high t/c's used during the last war, a great deal of the compressibility trouble arose from the unmanageable characteristics which are associated with thick cambered wings. Removing the camber and thinning the wings avoided these troubles. It is evident from the discussion given in section 2.6 that to get the best from camber these troubles must not be re-introduced.

Thus for thickened wing versions of the 1,100 and 1,000 sq. ft. proposals, there is a decidedly increased risk of not being able to achieve the saving in fuel load due to camber, and hence to cancel out the small reduction in weight which was otherwise claimed. Added to this, there is the certainty of a performance penalty even greater than that due to size along, which may be compounded by the increased fuel load caused by not being able to derive full benefit from camber.

4.3.3.2 External Tanks: A single external drop tank mounted under the fuselage from the centre keel structure is very easy to install on all aircraft in the family under consideration. If an adequate margin of internal fuel capacity is provided on the smaller versions by thickening the wings, the external tank required need not be as large as shown on Table 4. The situation for the thickened wing versions is shown on Table 10 as well as the margins required for the contingencies described above.

A study of the clearances involved shows that a tank of much over 500 gal. capacity is not very practical. Other locations and configurations have been considered and are believed to involve very severe difficulties. Some of these things are discussed in more detail in para 4.4.2.2. Accordingly it is evident that as the aircraft becomes smaller the risk of not being able to achieve a range of 1,500 miles increases. Thus, it is reasonable to assume that there is virtually no hope of the 1,000 sq. ft. version meeting the long range requirements, and only fair chance for the 1,200 sq. ft. wing, while the 1,400 sq. ft. wing it is virtually certain that this range can be achieved.

TABLE 10

Effect of Contingencies on Fuel Capacity Margins and Ranges of the C 105/2 with 500 Gallon External Tanks and Gross Wing Areas of 1,400, 1,300, 1,100 & 1,000 Sq. Ft. (2 RB 106 Engines)

GROSS WING AREA – SQ. FT.		1,400	1,300	1,200	1,100	1,000
t/c, WING THICKNESS TO CHORD RATIO	%	3	3	3	3½	4
ESTIMATED FUEL CAPACITY MARGIN	%	21	7	0	– 6	–6
MARGIN IF FUEL CONSUMPTION IS 10% GREATER	%	10	–2	–9	–15	–14
MARGIN IF WEIGHT IS 10% GREATER	%	13	0	–7	–13	–13
MARGIN IF ALL CONTINGENCIES OCCUR IN THE LONG RANGE MISSION*	%	3	–9	–16	–21	–21
LONG RANGE MISSION* RANGE WHEN ALL CONTINGENCIES OCCUR	N.M.	>1,500	1,350	1,280	1,170	1,180

*Long Range Mission as used here is for 500 gallon External Tanks; not the External Tankage required for 1,500 Nautical Mile Range noted in Table 4.

4.3.4 Effect of Altitude:

The effect of the cruising altitude on the long range mission is shown on Fig. 14 for the 1,200 sq. ft. aircraft which may be regarded as typical.

The effect of altitude on weight of fuel required for combat is shown on Fig. 15.

4.3.5 Performance with External Armament: (Ref. AIR 7–3, paras 3.04.01 and 10.03.04)

The penalties on the performance of a supersonic airplane due to externally mounted armament are bound to be severe, when one considers that the drag of the basic airplane would be increased by something of the order of 20% at $M = 1.5$ by 4 externally mounted missiles of Velvet Glove dimensions. However even more serious difficulties may arise due to the effect of the missiles on the C_{M_o}, which in turn has a profound effect on the elevator drag and the flight envelope limitations. Some recent tests[21] of missiles mounted on a symmetrical delta wing showed a change in C_{M_o} of about .003 at $M = 1.5$, which is about 25% of that due to the proposed camber. Although these tests were both subsonic and supersonic, they did not cover the transonic region. However it can be seen from Fig. 16, that wind tunnel tests on the C 100 with and without Velvet Glove missiles showed a very large and erratic variation of C_{M_o} at Mach numbers in the neighborhood of 0.9. Accordingly there is reason to doubt that the effects are as mild as may be inferred from ref. 21, especially in the transonic region. This will make the problem of developing a camber suitable for the clean airplane, which is also reasonable when external missiles are fitted, a very dubious business. It seems virtually certain that if external missiles are to be allowed for in the basic design, that a compromise camber would have to be accepted together with much larger flight envelope limitations than would otherwise be necessary.

Lack of certain data on camber, which can only be established by test, makes the problem of the combination of external stowage of missiles and camber extremely difficult. At present a test program has been instituted to start an investigation of camber. In order to get similar information on the effect of external missiles, it would be necessary to set up a similar program for the combination of missiles and camber. Since there are a large number of permu-

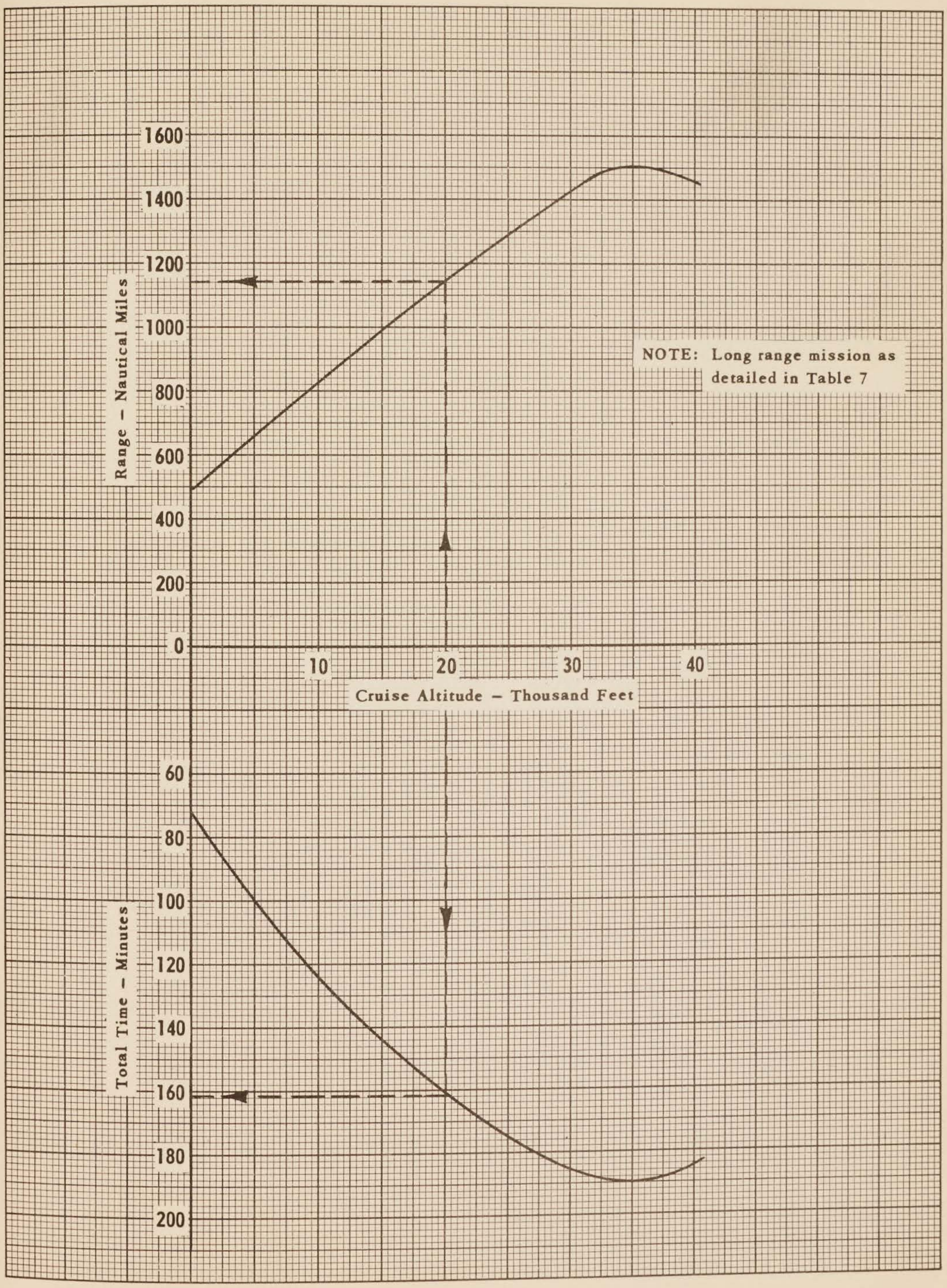


Fig. 14 1200 sq. ft. (2–RB 106 Engines)
Range & Total Time vs Altitude

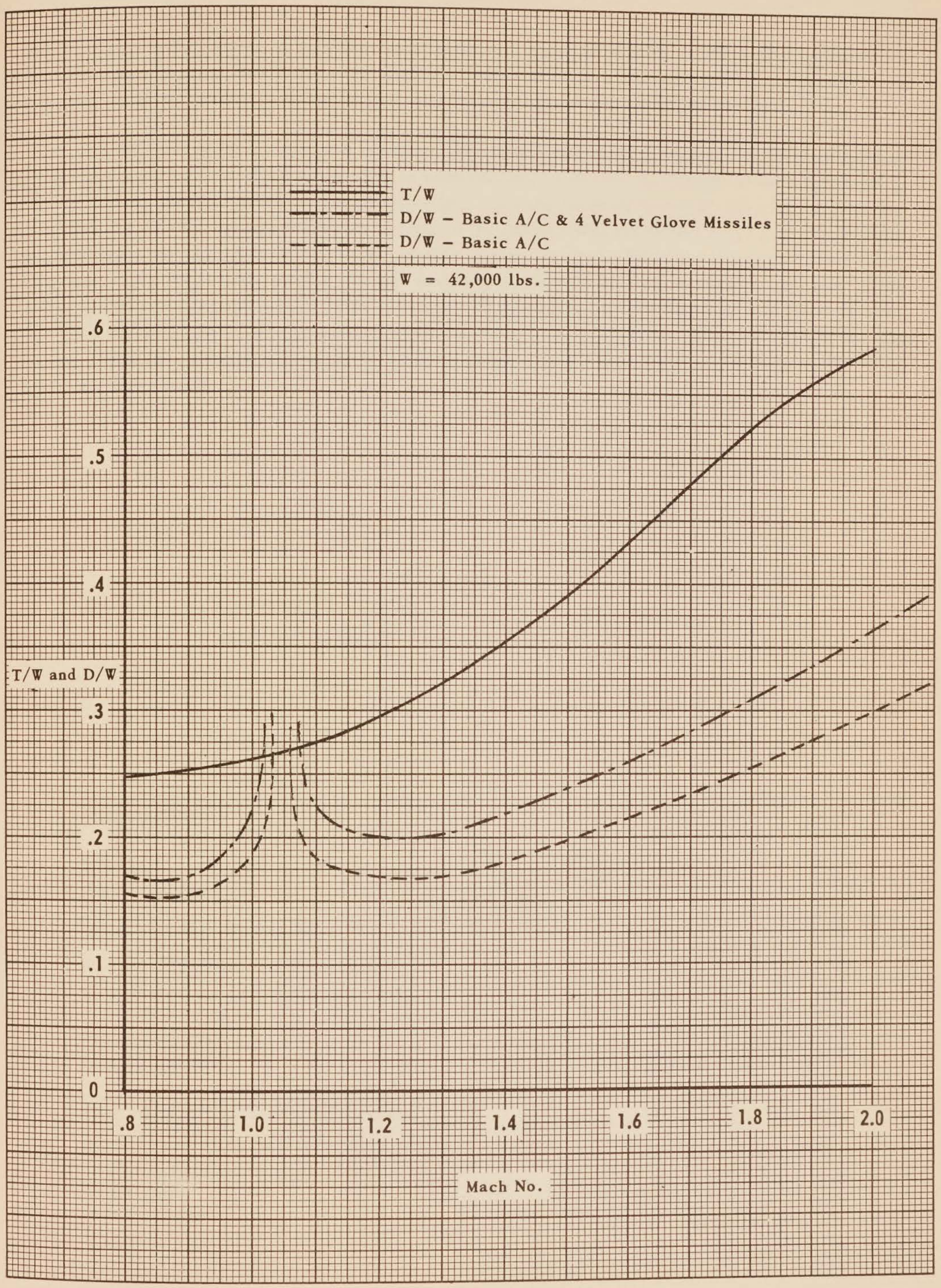


Fig. 17 1,200 Sq. Ft. – 2 RB 106 Engines
Thrust and Drag to Weight Ratio at 50,000 Ft.

tations and combinations of missile types and arrangements together with varying amounts of camber, this program would be very lengthy and expensive. However until some results are obtained, it is not really possible to give any reasonably reliahle performance data.

In order to give some appreciation of the effect of external missiles on the performance some data have been worked out on the assumption that there is no change in the elevator drag. In Fig. 17 are shown curves of thrust and drag, with and without external missiles at 50,000 ft. The high drag in the transonic region is caused by the elevator as shown on Fig. 5. This makes it impossihle to maintain level flight in this region at high altitudes either with or without missiles. The high drag of the aircraft carrying external missiles is readily apparent. Estimates of the performance without any allowance for changes in elevator drag are given in Table 11. From these figures and the preceding discussion, it is evident that there is a great incentive to use internally stowed armament if at all possible, and only to go to the externally stowed type as a last resort.

TABLE 11

C 105/2 – 1,200 Sq. Ft. – 2 RB 106 Engines

Performance with 4 – External Velvet Glove Missiles

GROSS WEIGHT		lb.	50,200
FUEL LB.	Supersonic Mission[1]		14,700
	Snbsonic Mission[2]		13,800
	Long Range Mission[3]		23,200
INTERNAL FUEL CAPACITY		lb.	16,500
SIZE OF EXTERNAL TANKS REQUIRED		gals.	900
RANGE WITH 500 GAL. EXTERNAL TANK[4]		N.M.	1,320
COMBAT CEILING FT. ½ FUEL WEIGHT	0.95	M.N.	54,000
	1.50	M.N.	62,600
	1.75	M.N.	65,300
TIME TO 50,000' FROM STANDING START		mins.	4.4
'g' AT 50,000' AT 1.5 M.N. AT ½ FUEL WEIGHT			1.95
LANDING DISTANCE FROM 50 FT. – FT. STANDARD DAY	½ Fuel Weight		5,730
	5 Min. Fuel Reserve Wt.		4,900
TAKE-OFF DISTANCE TO CLEAR 50 FT. – FT. HOT DAY	Overload Weight		3,680
	Gross Weight		2,620

4.4 Installation Features

4.4.1 Powerplant Installation

4.4.1.1 Installation drawings are shown in this design study for the following engines complete with afterburners and accessories:

	Refer to
Rolls Royce RB 106	Fig. 18
Bristol B.OL.4	Fig. 19
Curtiss Wright J 67	Fig. 20

For each of these engines the smallest practical rear-fuselage envelope has been shown. The width of the structural centre-beam between the engines is determined by considerations of stowage space requirements for flying controls, hydraulic-, electric-pneumatic- and fuel -pipes and -connectors, the tail-skid and the brake-parachute. Of these, the brake-chute is probably the most critical and in a letter received from the Irvin Air Chute Company[22] it is stated that a width of 9 inches is the practical minimum for the stowage of the brake-chute and coil spring for the auxiliary pilot-chute. The installation of the brake-chute is shown in Fig. 21. The only other place where a brake-chute container could be fitted would be on the fin, but this position is not favourable because it would reduce the rudder span and also would cause extra interference drag; poor aerodynamic lines in this region may also cause buffeting as was found to be the case on the Gloster Javelin. Hence it is concluded that the width of the centre beam between the engines must be at least 9 inches. The overall width of the rear fuselage is determined by the size of engine c/w accessories, afterburner with nozzle operating mechanism, air space for cooling air, and depth of structural formers. The overall height of the rear fuselage is determined similarly and also by considerations of adequate air space between the bottom wing skin which is an integral fuel tank wall and the engine. Since it was agreed by the R.C.A.F. that it was unlikely that the afterburners, complete with nozzle operating mechanism, on any of these engines would have substantially differing dimensions, when detail design on these items is finished, the fuselage size can be fixed by that required to house the J 67 afterburner on which most detailed information is at present available. This may be enlarged upon as follows:

4.4.1.2 From information received from Curtiss-Wright, the maximum width over the afterburner-nozzle operating jacks is 45 inches. Air space requirements around the engine, fix the structural boundaries in the region of these jacks as being 2.5 inches on either side of the jacks. The width of the structural formers around the outside of the engine has to be 3 inches and the width of the centre beam between the engines must be 9 inches, as stated in the preceding paragraph. Therefore the total width of the fuselage will be: $2 \times 45 + 4 \times 2.5 + 2 \times 3 + 9 = 115$ inches. The maximum depth of the fuselage is determined by the depth of the engine with accessories, and this just fits with fuselage depth requirements further forward.

4.4.1.3 Information on the B.OL.4 afterburner is extremely scanty and gives no details at all on the nozzle operating mechanism. It would appear however, that if a similar nozzle operating mechanism were used as on the J 67 it might be possible to house it within the same width of fuselage as calculated for the J 67 installation. Since the engine drawing received from Bristol (Drg. B68092) shows the engine to be about 2 feet shorter than the J 67, it is possible to mount the OL.4 somewhat further back than the J 67 and this means that it appears possible to reduce the fuselage depth by 2 inches; however, considerations of fuselage depth requirements forward of the engine bay show that this reduction in fuselage depth cannot be maintained. This is further explained in the next paragraph.

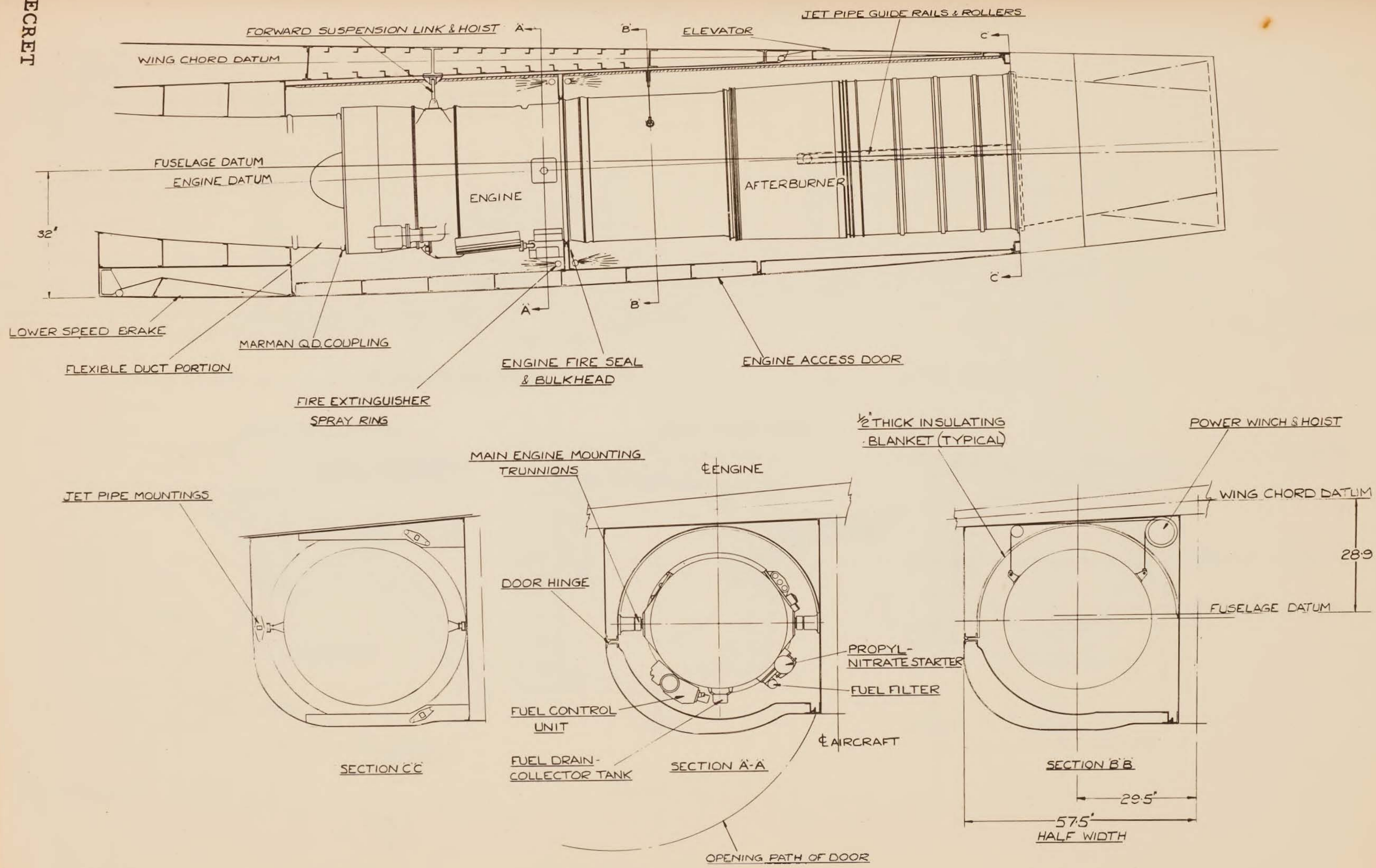


Fig. 18 Engine Installation – Rolls-Royce RB–106

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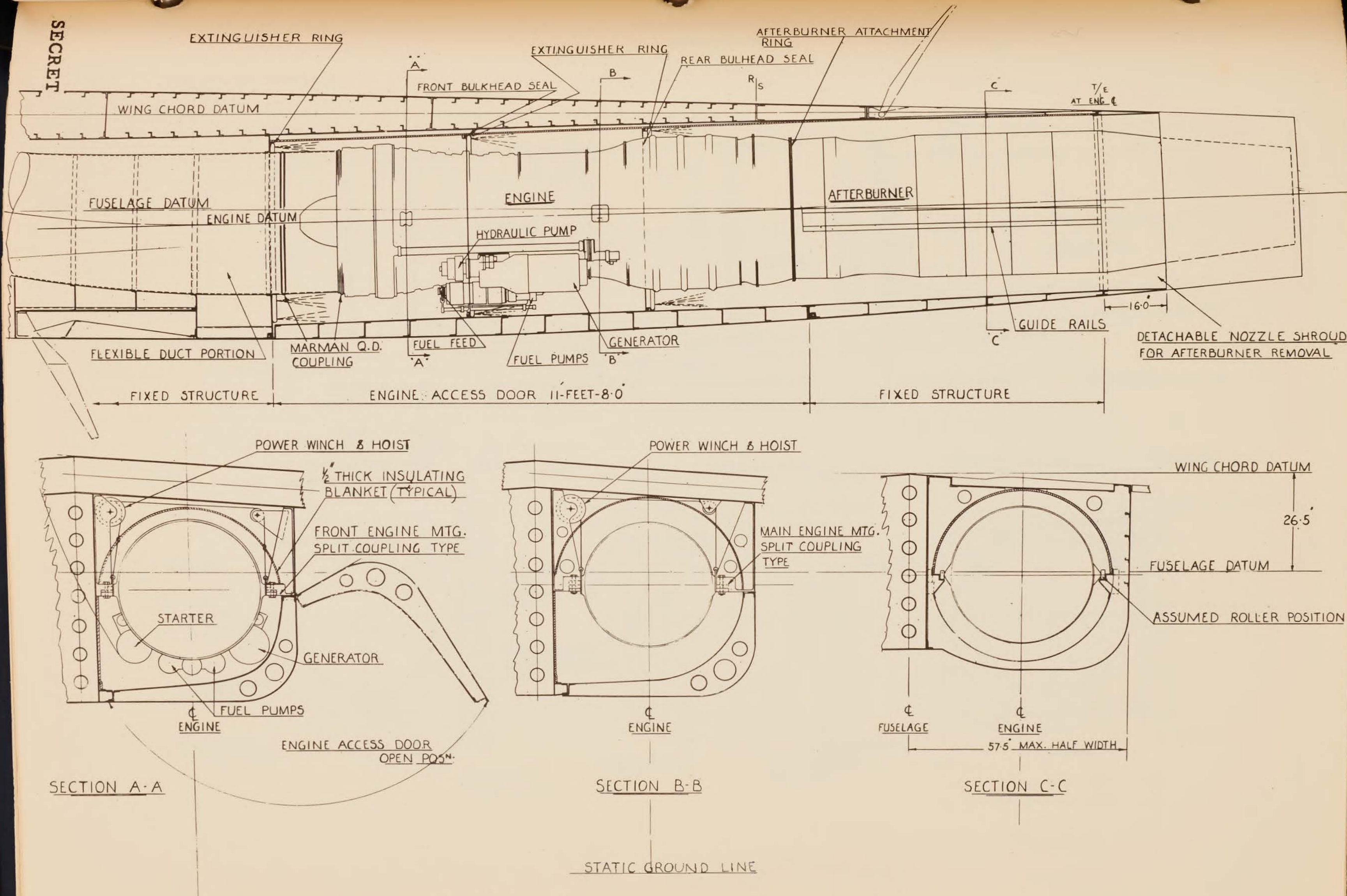

Fig. 19 Engine Installation – Bristol "Olympus" B–OL4

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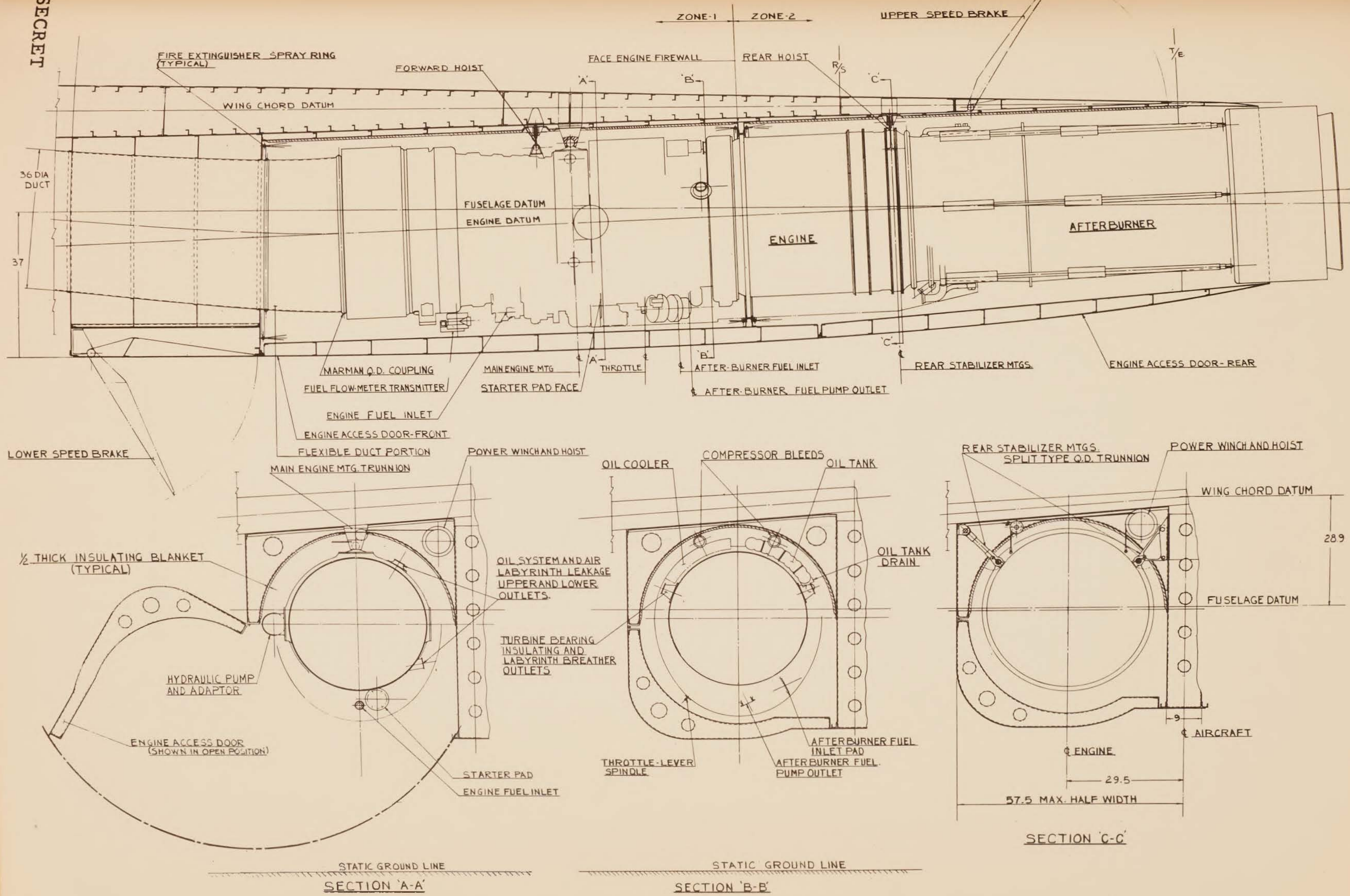
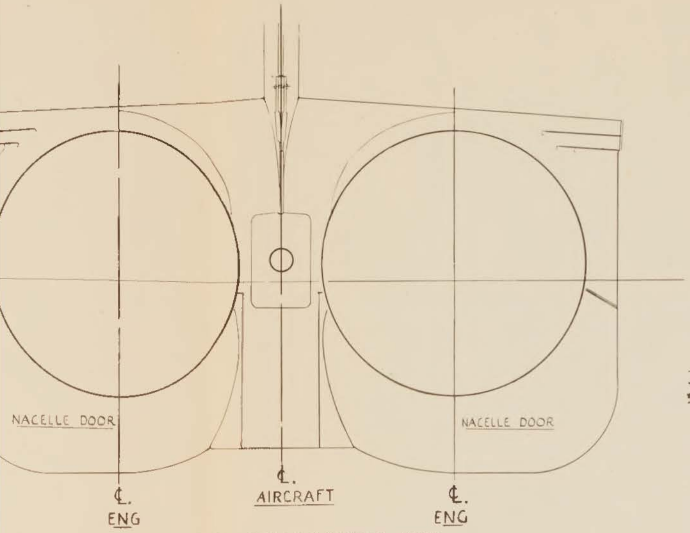
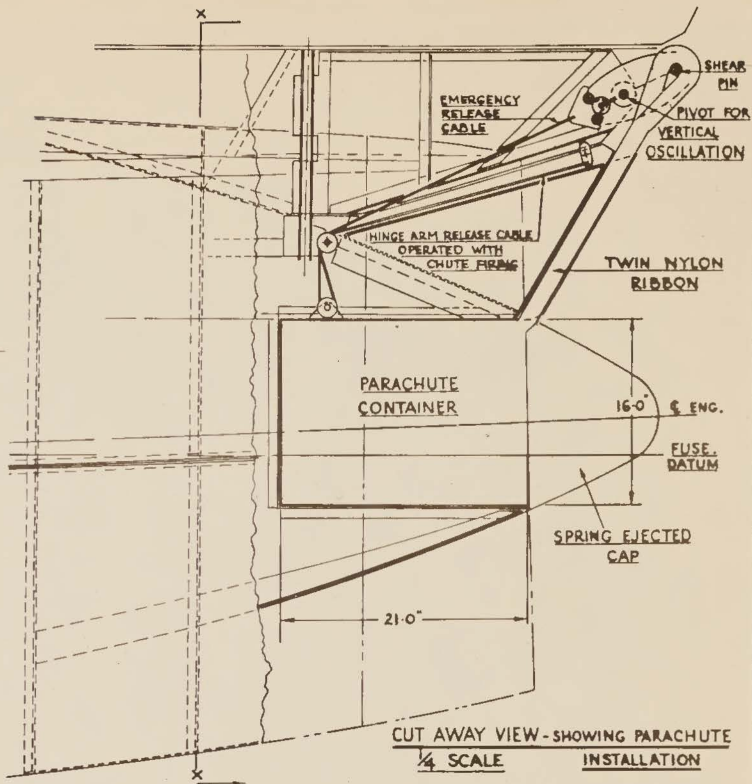


Fig. 20 Engine Installation — Curtis-Wright J-67

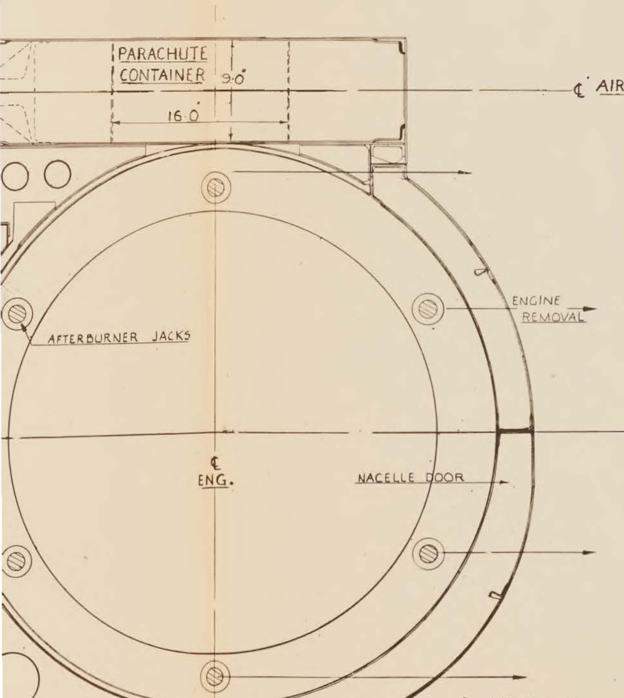


END VIEW LOOKING FWD.
1/8 SCALE

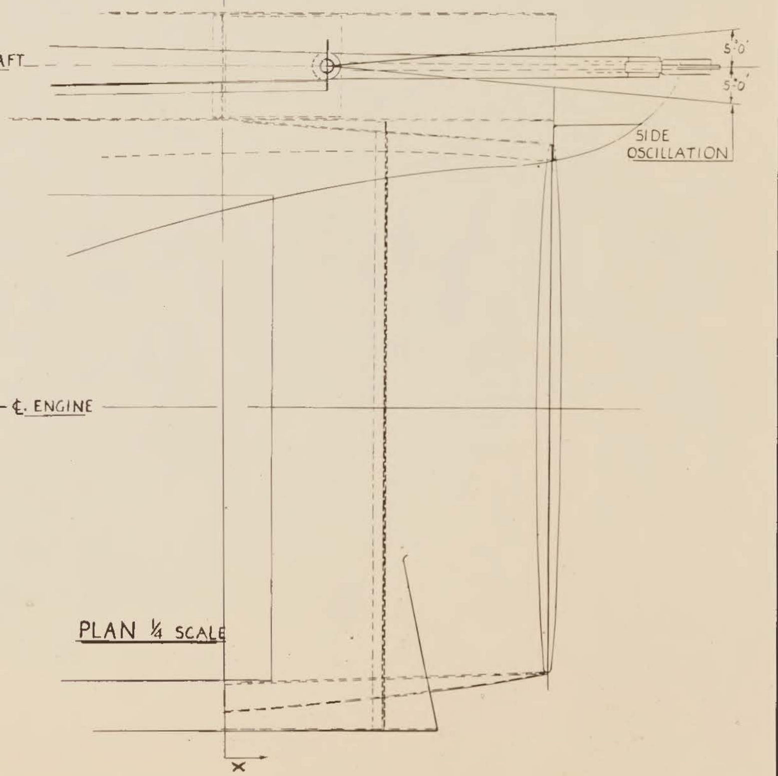


CUT AWAY VIEW - SHOWING PARACHUTE INSTALLATION
1/4 SCALE

DETAIL "A"



SECTION X-X
1/4 SCALE



PLAN 1/4 SCALE

4.4.1.4 Information received from Rolls Royce includes a drawing of a proposed afterburner (Drg. BT Sch. P 8079), but again this shows no details whatsoever on the nozzle operating mechanism and it does not appear as if any thought has so far been given to space requirements for same. The nozzle indicated on the drawing consists of only four lines and it would be very unwise, at this stage, to commit our fuselage lines on such scanty information. The maximum diameter quoted on the Rolls Royce drawing for the afterburner is 40.6 inches. Assuming that a nozzle operating mechanism has to be fitted around this afterburner as used on the J 67, we get the following: 1 inch clearance between nozzle operating jacks and afterburner, 2.75 inch diameter jacks, 1 inch clearance between jacks and surrounding structural formers, 3 inch wide formers and 9 inches wide centre beam, giving a total width of the fuselage of: 2(40.5 + 1 + 1 + 2.75 + 2.75 + 1 + 1) + 2 x 3 + 9 = 115 inches, which is the same width as was calculated for the J 67 installation. Since the overall height of the RB 106 engine is 5.75 inches less than the J 67 with accessories, it would appear possible to reduce the fuselage depth by about this amount, see Fig. 18. However, considerations of fuselage depth requirements forward of the engine bay show that this rednction in fuselage depth cannot be maintained. The critical fuselage section, as far as depth is concerned, is the section at the transverse wing spar just in front of the stowed undercarriage. As will be seen from the general arrangement drawings of the aircraft, this section encompasses the said spar, the intake-ducts and the armament bay. Now since this spar supports the forward part of the wing, it is essential that it be very stiff in order to prevent undesirable warping of the wing airfoil and, in order to keep structural weight to a minimum, it is therefore necessary that the lower flange of this transverse spar is not arched over the intake duct, but remains straight across the fuselage where it passes over the intake-duct. The diameter of the intake-duct cannot be decreased and neither can the depth of the armament bay, as will be apparent from a study of the general arrangement drawings. Hence the fuselage depth is determined by above considerations rather than by the engine installation.

4.4.2 Fuel Stowage

4.4.2.1 The internal fuel capacity of the various aircraft considered is determined by considerations of practical installation and balance about the desired centre of gravity of the airplane. Integral wing tanks must be resorted to on all the aircraft in the family, in order to make full use of the limited amount of space available in the very thin wings. Due to the fact that the wing fuel is situated aft of the c.g. it is necessary to balance this by fuel contained in the centre fuselage forward of the engines. Fuselage tanks will be situated between and above the intake ducts of the engines and may be of the bladder cell type. No fuel can be carried in the wing leading edge because this space is reserved for hot air anti-icing. When looking at the general arrangement drawings of the aircraft, it might be asked why no fuel is carried in the outer wings or say the fin; the answer to this is, that even if this were a practical installation, it would be necessary to balance this extra amount of fuel with more fuel in the fnselage, so that the length of the fuselage would have to be increased and this would then mean that the aircraft centre of gravity would be too far forward in the fuel empty condition. If it were then attempted to correct this by moving the wing forward relative to the fuselage, it would be seen that the gronnd-angle in the tail-down attitude would decrease, unless it were also possible to increase the length of the main undercarriage. The latter cannot be done, as will be explained later, and the ground-angle required for landing is already as small as all available evidence permits. In computing the fuel capacity in pounds, the specific gravity of the fuel has been taken as 0.75 and each tank has an expansion space equal to 3% of its normal capacity, in accordance with AIR 7-3 para 6.04.03. Here follows a table showing the internal fuel capacities of aircraft in the family, all with 3% thick wing:

TABLE 12

C 105/1000	12,900 lb.
C 105/1100	14,200 lb.
C 105/1200	16,500 lb.
C 105/1300	17,600 lb.
C 105/1400	20,400 lb.

It would appear to be possible to increase tbe thickness-chord ratio of the smaller wings with the same size fuselage without exceeding the permissible c.g. range and the internal fuel capacities are then:

for C 105/1000 with t/c of 4% . . . . 16,600 lb.
for C 105/1100 with t/c of 3.5% . . . . 15,900 lb.

4.4.2.2 External fuel capacity is required for carrying fuel to permit a minimum overload range of 1,500 n.m. with combat armament installed,in accordance with AIR 7–3 para 3.07.01. The tanks shall be jettisonable in flight and shall be capable of rapid installation and removal while the aircraft is on the ground, in accordance with AIR 7–3 para 6.04.06. For reasons of c.g. balance there are only two positions where external tanks can be fitted, i.e. either suspended from tbe wing outboard of tbe undercarriage or alternatively, suspended from the fuselage belly. With regard to suspension from the wings, experience with a similar problem on the C 103 project has shown that the difficulty of providing a suitable wing structure to cope witb aeroelastic effects would be almost insuperable on a wing of the order of thickness required for supersonic flight such as is now contemplated. Even if the aeroelastic problem could be solved, tbe weight penalty involved would be prohibitive. Furthermore, the fact that the bigh wing is some distance from the ground is not conducive to the fulfilment of the "rapid installation" requirement. It is therefore, concluded tbat the only satisfactory solution is to have one drop-tank suspended as a pod from the centre beam of the rear fuselage. This type of streamlined pod tank is cheap to manufacture, can be rapidly installed or removed, can be safely jettisoned in flight, allows the engine-access doors to be opened for servicing, allows the lower speed brakes to be opened, does not interfere with the aircraft's control surfaces and has a relatively low drag. A so-called "slipper-tank" has been considered but has a higher drag, is not easily jettisoned, interferes with engine servicing and the speed brakes cannot be opened witb it installed. The largest size pod tank that can be fitted has a capacity of about 500 Imp. gallons or 3,750 lb.; for larger tanks the clearance that can be maintained between the tank, the fuselage and the ground become marginal, as can be seen from the general arrangement drawings of the various aircraft where a 500 gallon tank is shown on the 1,200 sq. ft. wing version.

4.4.3 Landing Gear Installation

4.4.3.1 As will be seen from the general arrangement drawings of the various aircraft, the landing gear consists of an orthodox tricycle arrangement with a retractable tail skid fitted between the afterburners. The nose gear retracts forward into a space below the cockpit and is of simple design for all aircraft in the family. The solution of the main gear retraction and stowage problem requires a great deal of ingenuity but can be done quite satisfactorily for the larger aircraft of the family, for the smaller winged aircraft tbis becomes progressively more difficult. Such an undercarriage can just be installed inside the 1,000 sq. ft. wing and then only by means of an excessively complicated mechanism and relatively large local bulges on the airfoil of the wing next to the fuselage. For wing areas below 1,000 sq. ft. tbe problem is insoluble and one has to resort to the low wing configuration (this has been done in Appendix A of this study). The reason why it is more difficult to stow the main undercarriage into the smaller wings is bound up with:

(a) The required ground-angle of the aircraft in the tail-down attitude.

(b) The required position of the main wheels relative to the aircraft's c.g. in the fore and aft position.

(c) The required location of the undercarriage leg attachment to the wing structure and the forward slope of the extended leg.

Now requirements (a) and (b) are determined by aerodynamic considerations of stability and lift, necessary to execute safe take-offs and landings. This has been previously explained para 3.3.2.1.3 of reference 5. Requirement (c) means that once the location of the wheels in the extended position relative to the aircraft c.g. and therefore relative to the wing mean chord is fixed, it is highly desirable that the position of the undercarriage pivot axle in the wing is located such that the centreline of the undercarriage leg is approximately at right angles to the wing chord in a fore and aft plane. Were the pivot axle further aft relative to the wheel, it can be shown that extremely large moments due to ground reactions would be thrown on the leg, the pivot attachment fittings and on the wing sparbox, which would increase the weight of these items disproportionately. The above considerations mean that for the smaller wings the main gear must be shortened in addition to the twisting and tilting motion already required of the bogie chassis. Detailed design studies have shown that this shortening of the leg is necessary for wings smaller than 1,200 sq. ft. In order to clearly demonstrate the difference between the relatively simple mechanism for the gear in the 1,200 sq. ft. wing and tbe complicated arrangement necessary in the smaller aircraft, these mechanisms have been described in detail in the following paragraphs.

4.4.3.2 Main Undercarriage for Aircraft with 1,200 sq. ft. Wing (Refer to Fig. 22)

This undercarriage is designed so as to obviate the undesirable slamming down of the front wheel of the bogie when the rear wheel contacts the ground in the normal tail-down landing attitude of the aircraft and the general design is a development of the original proposal described in para 3.8.2 of reference 5. The bogie chassis is linked at the front wheel axle to the main leg by means of a member which is free to shorten but cannot extend. This is done by means of an air loaded telescopic strut which is fully extended for landing. On touch-down of the rear wheels the bogie chassis rotates about the front axle attachment and closes the main shock absorber at half velocity and prevents the front wheel acquiring an additional downward velocity. As soon as both wheels are in contact with the ground, this strut telescopes along with the main shock absorber which is a liquid spring housed inside the leg. Due to tbe inclined pivot axle of the gear where it attaches to the wing, it is necessary to twist the bogie chassis about the main leg during retraction and also it must be tilted about its attachment axle to the main leg. These motions are obtained mechanically as the undercarriage retracts by an actuating rod attached at one end to a point on the wing structure offset from tbe main pivot axle and at its other end to a torque sleeve situated around the lower portion of the main leg. This torque sleeve is provided with a profiled cam slot and this slot engages witb a roller which is fixed to the main leg. The torque sleeve is also provided with splines which engage with splines on the main leg when the torque sleeve is in its "up" position, i.e. gear extended, and which are disengaged when the torque sleeve is slid down, i.e. gear retracted. To the torque sleeve are attached the conventional torque scissor links which attach also to the bogie chassis. When the undercarriage starts being retracted, the sleeve starts moving down the leg and disengages the splines, further retraction forces the sleeve to rotate around the leg by virtue of the profiled cam slot and roller and this rotation is communicated to the bogie chassis via the scissor links. Tilting of the bogie chassis is automatically done during the downward movement of the torque sleeve by virtue of the telescopic air loaded strut which also attaches to this sleeve and which pushes the front of the bogie chassis down relative to its attachment to the main leg. The side stay of the undercarriage is telescopic and incorporates internal locks. The retraction jack operates directly onto the main pivot. It will be seen that the main gear can just be stowed within the airfoil contour of the wing and requires no bulges.

4.4.3.3 Main Undercarriage for Aircraft with 1,000sq. ft. Wing (Refer to Fig. 23)

This undercarriage must be shortened 12 inches during retraction in addition to the motions described in the previous paragraph, in order to stow it into the space available inside the wing. The springing medium, side stay, retracting jack and method of twisting the bogie chassis are all identical to that used on the 1,200 sq. ft. wing. The tilting of the bogie chassis only is different in so much as the shortening of the undercarriage tilts the bogie about its front attachment to the airloaded strut. The method of shortening is completely hydraulic and should there be a pressure loss, an emergency system is required. On selecting undercarriage "UP", hydraulic pressure is applied at 'A' and valve 'D' opens allowing fluid to pass from the shock absorber cylinder into the recuperator. At the same time pressure is applied at 'C' which forces the jack cylinder (attached to the shock absorber) along the piston rod and thereby effects the shortening; on selecting undercarriage "DOWN" pressure is applied at 'B' which forces the floating piston in the recuperator to move towards the shock absorber and thereby re-charge the shock absorber. When the shock absorber is charged, pressure is released from 'A' and closes the valve 'D'. No pressure is applied to the cylinder, as the shock absorber pressure will automatically extend the strut. The maintenance difficulties will be severe with this type of undercarriage, because of the increase in number of seals which can only be serviced by a complete dismantling of the leg. The valve 'D' is required to hold a pressure of 41,900 p.s.i. with no leakage and presents the problem of a maintenance-free high pressure seal. As can be seen from the drawing, this undercarriage will not fit inside the airfoil contour and bulges in the upper and lower surfaces of approximately 2 inches depth around the bogie are required. Since the stowage bay is now shorter in the spanwise direction, the side stay must now lie along the side of the rear wheel in the retracted position. This therefore increases the width of the bay in the chordwise direction and aggravates the problem of designing satisfactory doors and fairings. Summarising, it can be stated that although it might be possible to make such an undercarriage work, the problems involved are such as would necessitate a lengthy and therefore expensive development programme and involve considerable risk regarding the amount of maintenance that will probably be required in service. This also applies to an undercarriage for the 1,100 sq. ft. wing, although a mechanical method of shortening the leg appears possible here, because the amount of shortening required is about half that required on the 1,000 sq. ft. wing.

4.4.4 Armament Installation

4.4.4.1 As discussed in para. 4.3.5 of this study, it is concluded that no adequate data are available to permit a true comparative picture to be presented of a family of aircraft fitted with external armament. Therefore, this comparison deals with internally stowed armament which, in accordance with AIR 7-3 para 10.03.04.01, is based on 6 "Falcon" missiles plus 50 - 2 inch F.F.A.A. rockets. The weight of the missiles is 6 x 132 = 792 lb. The weight of the rockets is 50 x 10.5 = 520 lb. The ejection mechanism has been calculated to weigh 410 lb. Reference may also be made to the armament installation described in the C 104/2 brochure[5].

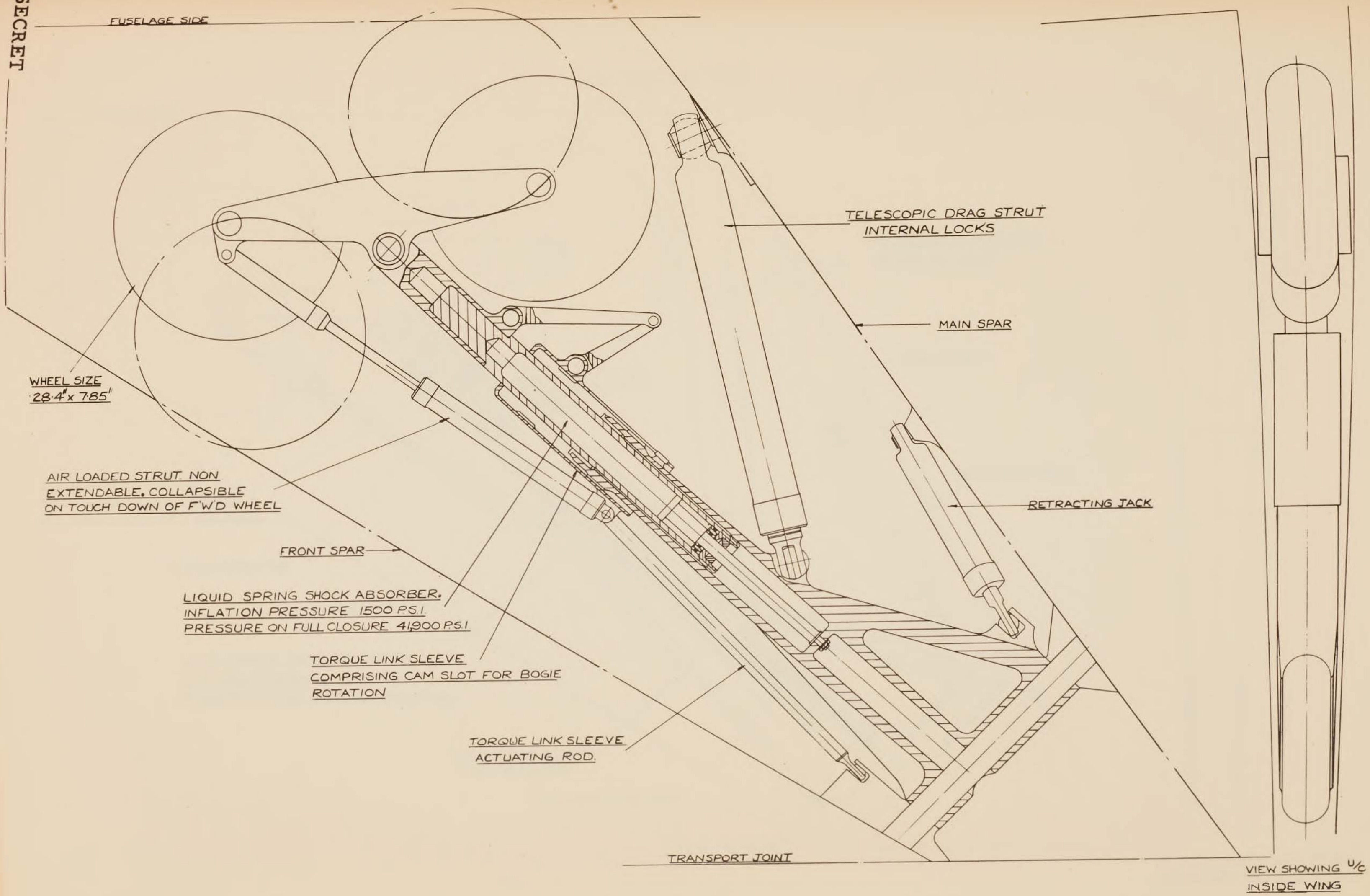


Fig. 22 Undercarriage Installation – for 1200 sq. ft. Wing

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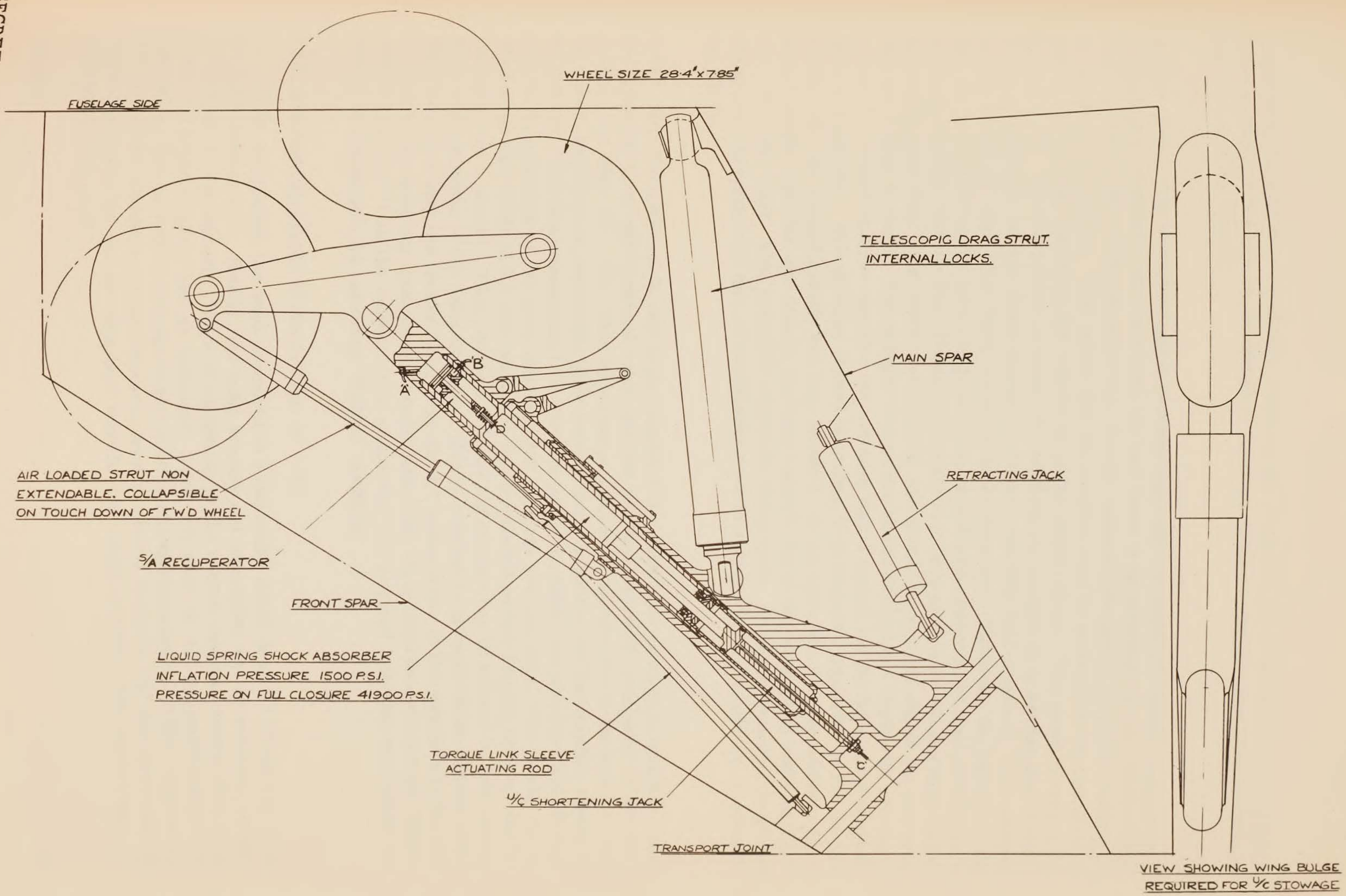


Fig. 23 Undercarriage Installation – for 1000 sq. ft. Wing

Considering the aircraft in the family which we are comparing, there are two methods of installation possible, depending on the length of fuselage which is available from considerations of c.g. balance of the aircraft.

4.4.4.2 With a very short fuselage it becomes impossible to install the avionic equipment in a packaged crate forward of the armament bay and hence it is necessary to stow these avionic boxes on either side of the armament bay, thereby narrowing the amount of space available for the missiles and rockets. This means that the missiles must be installed in two rows of three abreast and the only place for the rockets is in the missile doors. This layout is used in the Convair F 102. However, there is reason for considerable concern as to whether this is satisfactory, because the missile doors become too heavy, with their load of rockets, to be operated quickly enough to close after firing a missile of the front row. Therefore, the missiles of the back row must fire over an open cavity for some considerable distance and there is therefore a distinct possibility that the flow disturbances caused by these open doors will seriously disturb the missile trajectory. The whole door opening and missile firing cycle should not take more than two to three seconds. The missiles themselves should be fired in a ripple with a timing of at least 0.1 second between missiles but not more than 1 second for the firing of the complete salvo of 6 missiles. With missiles stowed in tandem, it is obviously impossible to fire the rear missile, after the forward missile has been lowered. At the same time it is impossible to fire a forward missile after the rear missile has been lowered, due to the damage that would be inflicted by the motor blast. This obviously affects the firing cycle. As may be seen from the general arrangement drawings, this type of installation must be resorted to on the 1,000 sq. ft. aircraft, see Fig. 24.

4.4.4.3 With the 1,100 sq. ft. and larger aircraft it has been found possible to install the avionic equipment in a self-contained crate and the preferred armament installation is therefore possible, see Figs. 25 – 28, This installation consists of two missiles in the front row, with the rockets housed in an extendable pack between these missiles, and four missiles abreast in the back row. With this arrangement it is possible to fire the middle two missiles while allowing time for the extension of the outer missiles that have not yet been fired. This overlap can be added to the time delay of 0.4 seconds which can be inserted without bringing the total time for the ripple over 1 second. Thus, between 0.5 and 0.6 seconds can be allowed for door opening and missile extension of the outer missiles. This is believed to be ample for the purpose, since these doors have very low inertia, being only about 9 inches wide. In this way the missiles will not have to fire over any open cavity beyond their own. In a letter received from the Hughes Aircraft Company[23] on the subject, it is stated that this proposed launching arrangement appears to be satisfactory for launching "Falcon" missiles.

4.4.5 Avionics Installation (Refer to Figs. 24 – 28)

This installation has been described in considerable detail in para 3.17 and Fig. 32 of the C 104 brochures[5]. Briefly, the radar scanner and transceiver are, by necessity, located in the nose of the aircraft. Items of equipment in the cockpit comprise control units, radar screen for target display, and an instrument which will show all the integrated navigational information. The cockpit installation is described in subsequent paragraphs of this design study. The bulk of the avionic equipment however, must be stowed elsewhere and this space must be temperature controlled and the equipment must be very easily serviced. The required space is of the order of 55 cu. ft.

The preferred type of installation consists of a crate provided with a self-contained winch motor which houses all the necessary avionic boxes in a very compact and flexible manner. This arrangement greatly simplifies the servicing problem and at the same time makes it possible to design a light and compact air-conditioning system for same. Whether this arrangement can be fitted or not, depends on considerations of static weight balance in conjunction with the size of wing being fitted. It has been found possible to fit this type of installation on all aircraft of the family which have a wing area of 1,100 sq. ft. and greater.

For the airplane with 1,000 sq. ft. wing area, the fuselage is too short to accommodate such a crate and one has to resort to a distributed installation similar to that proposed for the single engined version of the C 104(4). This installation is somewhat heavier because of the additional wiring and ducting that is required, also there will be a weight penalty because every box has to be individually shock-mounted and a number of access doors will be required. Furthermore, as described in the paragraphs on the armament installation, this arrangement necessitates stowage of the missiles in two rows of three abreast with its attendant disadvantages.

4.4.6 Equipment Installation (Refer to Figs. 24 – 28)

The type of equipment necessary has been described in considerable detail in the C104/2 brochure(5). Briefly, the equipment consists of the following:

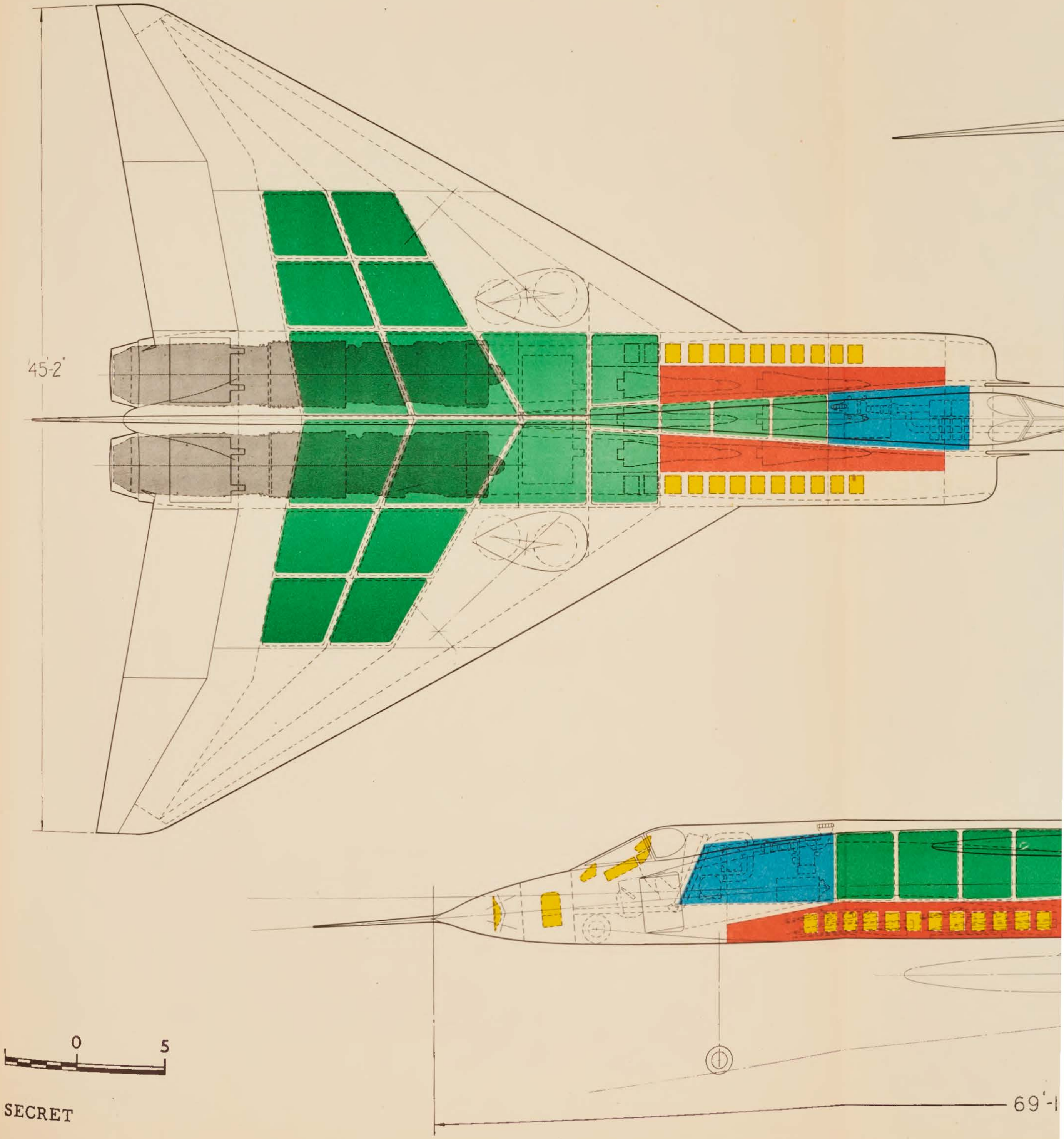
Low pressure air supply bled from the engine compressors
Air-conditioning and pressurization equipment
Electrical equipment
Hydraulic equipment

The amount of space required to house this equipment has been studied in detail, in order to arrive at the absolute minimum required. As is indicated on the general arrangement drawings of the aircraft of 1,100 sq. ft. wing area and greater, the air-conditioning and hydraulic equipment is located between the intake ducts and behind the avionic crate. Provision has here been made for the stowage of an auxiliary gas turbine compressor for ferrying purposes. This is similar to the ferrying unit required for the later marks of the CF100 aircraft, except that this unit for the C 105 will be much cheaper because it will deliver only compressed air and no electrical power and it will be stowed internally instead of it having to be an externally mounted pod. The electric generating equipment will be mounted between the air-intakes and behind the pilot's bulkhead where there is just sufficient space to house same.

The amount of space required for this equipment is virtually the same for all aircraft considered in the family.

5 TWO SEATER VERSION

5.1 In accordance with AIR 7–3 para 4.01.02, all aircraft discussed in this design study are capable of being converted to accommodate a crew of two, a pilot and a navigator/radar operator, to ensure the capability of conversion to dual pilot trainers or the acceptance of an alternative fire control system. The actual conversion is accomplished by fitting another front fuselage to the airplane at the transport joint between front- and centre-fuselage. This transport joint is located at the bulkhead which supports the nose undercarriage and by this means conversion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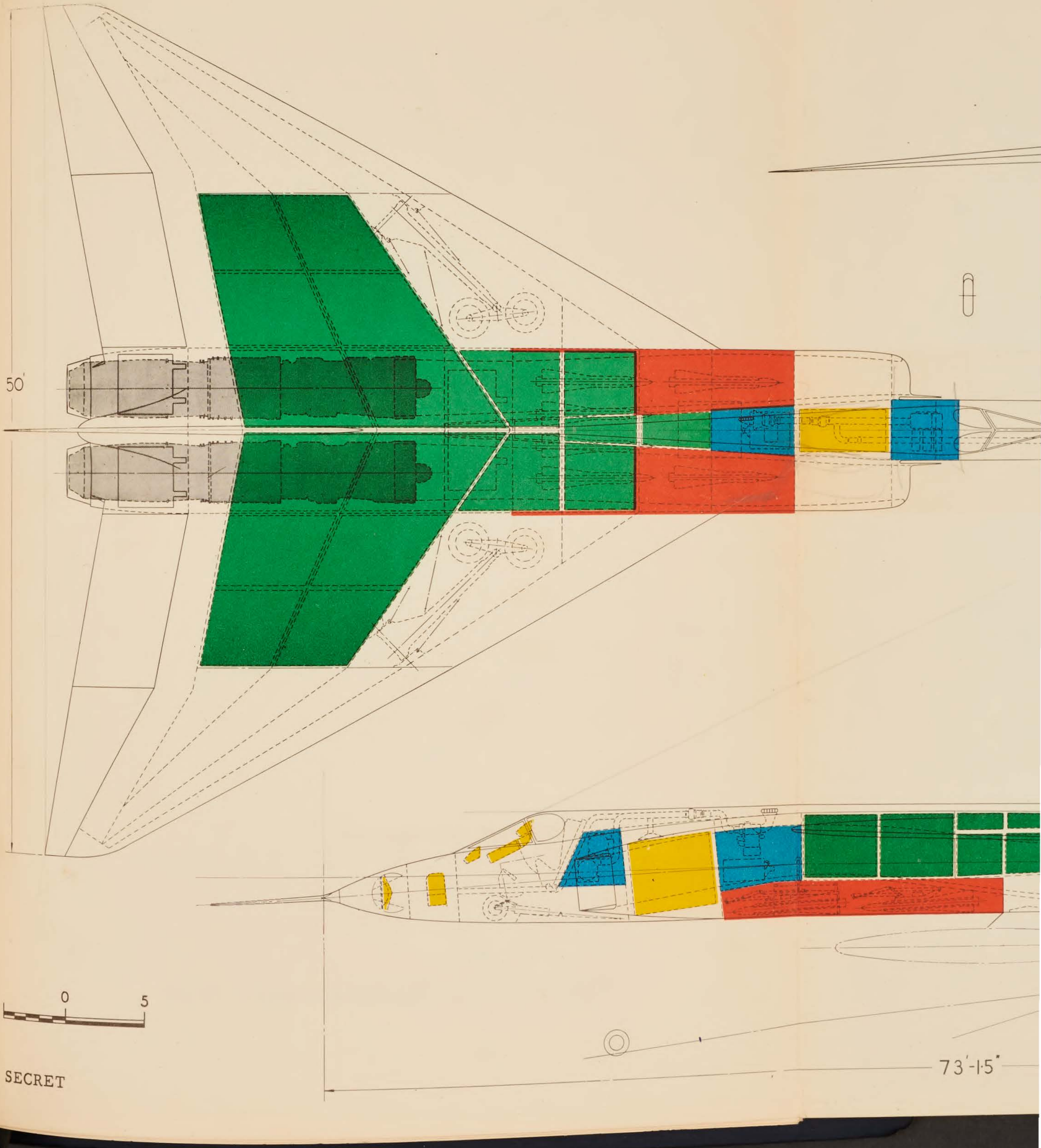

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can be accomplished with the minimum of re-work, keeping the size of the basic airplane to the absolute minimum possible. A general arrangement drawing of a two seater version of the airplane with 1,200 sq. ft. wing area is shown in Fig. 29. The increased length of the fuselage and other provisions for another crew member will of course increase the weight of the airplane and as an example, a table showing the weight breakdown of the 1,200 sq. ft. aircraft is presented. In this table, the weight of the avionic equipment has been kept the same as for the MX 1179 system. Balance calculations on this airplane show that it is necessary to install 500 pounds of ballast in the aft portion of the fuselage in order to prevent the c.g. of the aircraft moving too far forward. For the weight of the engine, the R.R. RB 106 data are used.

5.2 Weights Summary for C105/1200/T

TABLE 13

ITEM	WEIGHT IN LB.
ENGINES AND AFTERBURNERS	9,502
POWERPLANT FIXED ITEMS:	
Fuel Tanks	300
Fuel System	420
Fire Extinguishers	65
Accessory Gears and Drives	15
Engine Controls	20
GROUP TOTAL	820
EQUIPMENT:	
Instruments	50
Probe	50
Surface Controls	700
Hydraulic System	680
Electrical System	700
Radar and Electronics	1,800
Ejector Seats	264
Emergency Provisions	30
Oxygen	40
Air-conditioning and L.P. Pneumatics	625
Anti-icing System	300
Brake Parachute	75
Exterior Finish	75
Crew	460
Oil	40
Residual Fuel	225
Armament provisions	410
Armament – rockets	520
– missiles	792
Ballast	500
GROUP TOTAL	8,336
STRUCTURE:	
Vertical Tail	900
Fuselage	6,456
Wing	8,557
Undercarriage	2,109
GROUP TOTAL	18,022
OPERATIONAL WEIGHT EMPTY	36,680
FUEL FOR COMBAT MISSION	13,250
GROSS WEIGHT	49,930

5.3 It will be seen that this two seater version is 1,555 lb. heavier than the corresponding single seat version. Of this increase, 500 lb. is due to ballast. It will be clear that for the smaller aircraft in the family this increase in weight will be somewhat more than 1,555 lb. and for aircraft with wings greater than 1,200 sq. ft. it will be less than 1,555 lb. This increase in gross weight will affect the aircraft performance to a small extent as follows:

	C 105/1000	C 105/1400
increase in fuel required for mission	4%	2%
increase in landing distance	3%	2%
increase in time to combat altitude	5%	3%
decrease in combat ceiling	1.5%	1%

6 COCKPIT LAYOUT

6.1 In accordance with AIR 7–3 para 7.00 a layout of the cockpit's instrument and console panels together with a list of all flight and engine instruments is included in this design study. Fig. 31 shows a sideview of the cockpit and Fig. 32 shows the interior arrangement. The latter sketch should be read in conjunction with Table 14 which lists the proposed instruments and controls in the cockpit. The outside width of the cockpit requires to be 43 inches minimum and it should be noted that this dimension is used for all aircraft considered in this design study.

6.2 The cockpit as shown, has been designed for a Martin Baker Automatic Ejection Seat, with a telescopic gun giving an escape velocity of 80 ft./sec. According to information received from the Institute of Aviation Medicine[24], it appears that the limitations imposed on the speed of the proposed aircraft at low altitude are such that the use of this ejection seat is feasible. Reference should be made to Fig. 30, which has been reproduced from the I.A.M. report, and which shows the human tolerance to ejection at various speeds plotted against altitude, compared with the aircraft's "speed versus altitude" flight envelopes.

6.3 The joy-stick's hand grip has been designed especially for use in aircraft fitted with the MX 1179 system by the Hughes Aircraft Company. The control-column has been positioned so as to leave the cockpit floor area clear, in order to assist servicing inside the cockpit area and also in order to bring the three main flying control circuits into one unit under the cockpit floor, as shown in Fig. 31. This latter arrangement makes for convenient servicing of the control-box through the nose-undercarriage well.

6.4 All main instruments have been positioned so as to have minimum parallax and minimum "mirror effect". The main flying panel conforms as close as possible to AD 3001 within the limits imposed by available space and by the changed precedence of the Cross Point Indicator and deletion of the Direction Indicator and Artificial Horizon, due to fitting the MX 1179 automatic navigation system. This main flying panel is mounted at an angle of 25 degrees from the vertical and since certain instruments, i.e. Turn and Bank, Cross Point and Accelerometer will not work properly in this attitude, these may be mounted normal to the vertical without affecting the layout of the panel. Dual indicators are proposed for engine instruments in order to occupy less space and to permit easy comparison between the two engines. The Fuel Indicator shows "flow per minute" to each engine and "total remaining fuel" in order to permit easy calculation of the remaining flight time. All controls and switches have been so located, that they can be reached by the pilot with the harness in the locked position. Although not shown in Fig. 32, it is proposed to mount a stand-by compass on the windscreen arch. The optical sight, which retracts forwards and downwards over the top of the radar indicator, will be power-operated.

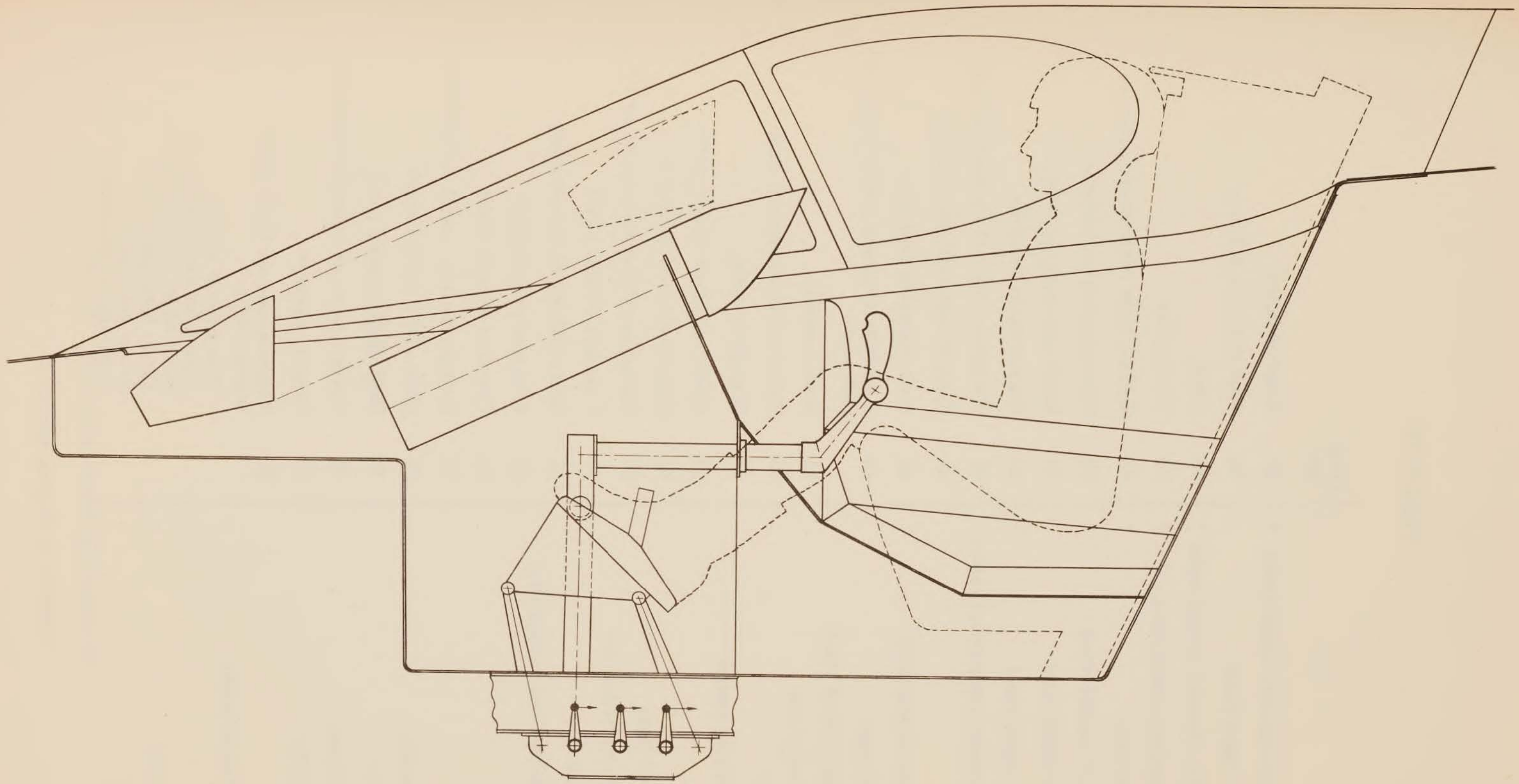

Fig. 31 Cockpit Side View

TABLE 14

ITEM NO.		
1	Data Receiver (sub-channel) control panel	*
2	Cockpit lighting control panel	
3	Data receiver (R.F. Channel) control panel	*
4	A.R.C. communications control panel	*
5	Headphone control panel	*
6	Ground to air I.F.F. control panel	*
7	Exterior lighting control panel	
8	Air to air I.F.F. control panel	*
9	Hydraulic and pneumatic pressure indicators	
10	Brake lever	
11	Armament selection control panel	*
12	Anti-icing control panel	
13	Starting and re-light control panel	
14	Braking chute control lever	
15	H.P. fuel controls	
16	Throttle levers friction control	
17	Throttle levers	
18	Speed brake control lever	
19	Undercarriage position indicators	
20	Undercarriage controls	
21	Fire warning indicators and extinguisher operating button	
22	Trim indicator	
23	Altimeter	
24	Canopy control handle	
25	Air speed indicator	
26	Rate of climb indicator	
27	Canopy lock indicator	
28	Machmeter	
29	Radar indicator control panels	*
30	Accelerometer	
31	Cross-point indicator	*
32	Optical sight	*
33	Radar indicator	*
34	Turn and bank indicator	
35	Clock	
36	Tachometer	
37	Optical sight controls and indicator	
38	Oil temperatures indicator	
39	Radio and magnetic compass	*
40	Oil pressures indicator	
41	Flowmeter and fuel contents indicator	
42	Exhaust temperatures indicator	
43	Fuel pressures indicator	
44	Fuel booster pumps control switch and indicators	
45	Rudder pedals	
46	Radar and power control panel	*
47	Emergency brake	
48	Flight sequence control panel	*
49	Electrical power indicators	
50	Emergency flying instruments switch	
51	Computer counter panel	*
52	Electrical power control panel	
53	Computer control panel	*
54	Computer Control panel	*
55	Cockpit heating control and indicator	
56	Glide slope control panel	*
57	DME–OMNI control panel	*
58	Cockpit pressure control and indicator	
59	Oxygen regulator	
60	Flight & antenna hand control, incorporating: Trim control switch Auto pilot over-ride switch Nose wheel steering switch I.F.F. interogate switch Range gate switch Lock and action switch	*

**Supplied by Hughes Aircraft Co.*

6.5 Deviations from Requirements Contained in C.A.P. 479

Reference para 91: The pilot's seat (Martin-Baker) does not at present incorporate 6 inches fore and aft adjustment and back-rest angle adjustment. Arm-rests are not provided since it is felt that these would impair the pilot's freedom of arm movement. The pilot may need to take over manual control at very short notice under high 'g' conditions and it is therefore felt that his feet should remain on the rudder pedals at all times. Hence, foot-rests will not be needed, but a ramp will be fitted so that the feet can be moved on to the ejector-seat foot-supports with the minimum of effort.

Reference para 93: The downward view over the nose, for all aircraft considered, is 13 degrees.

Reference para 94: It is not proposed to incorporate direct vision apertures in the canopy windows. Automatic de-misting and anti-icing will be provided for all the windows and the MX 1179 system incorporates an automatic landing procedure. The canopy can of course be jettisoned at all times and at all speeds.

Reference para 156: The position of the radio and radar control panels cannot conform entirely with this requirement because of the number of panels involved, but this may be changed later, when this equipment has been finalized by the Hughes Company.

Reference para 158: The starting- and relight-buttons have been located near the throttle levers.

Reference para 159: There will be about 70 circuit breakers and about 30 fuses for the whole of the electrical system. It is obvious that not all of these can be located inside the cockpit. It is therefore proposed to locate only the main ones in the cockpit and the location of these must be decided on a mock-up.

Reference para 164: The anti 'g' control is automatic.

6.6 Deviations from Requirements in AIR 7–3

Reference para 7–02–03: A direction indicator and an artificial horizon are not fitted because the functions of these are catered for by MX 1179. In the event of failure of the radio link in the MX 1179 equipment, the gyro transmitter in this equipment may be connected directly to the cross-point indicator by means of the emergency indicator switch. The cross-point indicator will then take over the functions of the direction indicator and artificial horizon and, together with the compass, will provide sufficient data for the pilot to return to base.

TABLE 15

Wing Area Sq. Ft.	1,000	1,100	1,200	1,300	1,400
Wing T/C %	4	3½	3	3	3
Normal Gross Weight[1] Lb.	47,200	47,700	48,400	49,100	49,800
Mission Distance when allowances are made for Contingencies on Fuel Consumption[2]					
(a) Supersonic Radius N.M.	200	195	200	200	200
(b) Overload Range N.M.	1,180	1,170	1,280	1,350	1,500
Ceiling (M = 1.5) Ft.[3]	64,300	64,900	65,100	65,300	66,000
Landing Dist. Ft.[3]	5,720	5,300	4,900	4,740	4,470
Installation[4]					
(a) Undercarriage	Very Complicated	Complicated	Good	Good	Good
(b) Electronics	Dispersed Tailored	Crated	Crated	Crated	Crated
(c) Armament and Equipment	Poor	Crowded	Good	Good	Good

NOTES: 1. For Details see Table 2.

2. For Details see Tables 9 and 10.

3. For Details see Table 4.

4. For Details see Section 4.4.

The main result of this study is that the gross weight of an aircraft with the specified military load and engines can only be varied within very narrow limits, even with fairly large changes in the aircraft size. Increased aircraft size results in improved performance within increased margins for contingencies. The installations are not so tight and hence can be engineered in less time and will result in a more serviceable aircraft. On this score, there is reason to doubt that there is nearly as great a saving on cost by going to the smaller versions as figures based on weight alone would indicate.

Hence it is evident that it is appropriate to strike a compromise. With a wing area of 1,100 sq. ft. or less, the undercarriage becomes more difficult, and the wing must be thickened to accommodate extra fuel. The tighter installations and the extra aerodynamic risks involved in the thicker wings make these versions undesirable, when one considers the very small weight saving involved balanced against the penalties. On the other hand the larger versions i.e., 1,300 and 1,400 sq. ft., appear to have more than the minimum necessary amount of room required to make simple installation of such things as the landing gear and the various items of equipment. It is accordingly felt that the 1,200 sq. ft. version represents the most satisfactory compromise between the minimum weight and the maximum performance and flexibility.

In conclusion, it is thought appropriate to draw attention to the fact that in the Operational Requirement[1] for this aircraft, it is stated that the threat demands the replacement of existing interceptors as early as 1957. This demands a tight design and production schedule. Hence it is evident that to make the best possible showing it is essential that the selected version incorporate the smallest aerodynamic risks and not be too cramped to complicate the detail design.

8 LIST OF REFERENCES

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3. R.C.A.F. Memorandum 360–3–94 (AMTS/DDA) "Advanced All-Weather Interceptor Requirements" – June 1952.

4. A.V.Roe Canada Limited "Single Engine All-Weather Fighter C 104/1".

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7. A.V.Roe Canada Limited "C 104/2 Minimum Weight Study" – April 30, 1953.

8. R.C.A.F. Development Study Report DDA – 12 "Estimation of All-up Weight of Interceptor Aircraft to R.C.A.F. Specification AIR 7–3. F/L I.S. McLeish and W/C D.A. MacLulich. – April 1953.

9. Lowry, R.E. – "Problems and Solutions of Delta Wings", S.A.W.E. May 1952.

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12. N.A.C.A. Report 971 "Theoretical Stability Derivatives of thin Swept-back or Swept-forward Trailing Edges for a limited Range of Supersonic Speeds. F.S. Malvestalo and K. Margolis. 1950.

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15. A.V.Roe Canada Limited – "Comparative Performance of Avro C 104/2 and Convair F 102 Supersonic Fighters – Wright J 67 Engines". April 1953.

16. N.A.C.A. RM A52D01C "Aerodynamic Characteristics of Two 25 Percent-Area Trailing Edge Flaps on an Aspect Ratio 2 Triangular Wing at Subsonic and Supersonic Speeds". J.W. Boyd. July 1952.

17. Rolls Royce – "Preliminary Design Study and Installation Data for the RB 106 Engines".

18. Bristol – "Olympus B.OL.4 Jet Propulsion Engine".

19. Bristol – Curve Sheets TED 2590, 2591, 2592, and 2593. "Performance of the Bristol B.OL.4 with Reheat".

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21. N.A.C.A. RM A52C10a "Lift, Drag, and Pitching Moment of Low-Aspect Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 3 with Air-to-Air Missile Models Mounted Externally". Donald Conrad, June 1952.

22. Irvin Air Chute Limited – Letter from H.K. Epple, Project Supervisor, May 13, 1953.

23. Hughes Aircraft Company – Letter from N.I. Hall, Associate Head of Guided Missile Lab. – ref. 4 M 25/301, Sept. 19, 1952.

24. I.A.M. 423 – Minutes of a Conference held at the Institute of Aviation Medicine, R.C.A.F. April 2nd, 1953, on C 104.

APPENDIX A

Aircraft with a 900 Sq. Ft. Wing

1. In para 4.2 of this report it is shown that the smaller the wing area, the lighter the aircraft. Although it appears that a point of diminishing returns has been reached with the 1,000 sq. ft. aircraft, it cannot be said that this gives the absolute minimum weight theoretically possible and regardless of all penalties involved. Accordingly, a study has been made of a still smaller aircraft with only 900 sq. ft. of wing area. A general arrangement drawing of this airplane is shown in Fig. 33.

2. As discussed in para 4.4.3 it was found to be impossible to stow the main undercarriage in a high wing aircraft with a wing area less than 1,000 sq. ft. It is therefore necessary to adopt the low wing configuration with the undercarriage retracting sideways into the fuselage belly.

3. The main problem centres around the question of fitting external fuel tanks such as are required for the ferry mission. This has been fully discussed in para 4.4.2.2 and the difficulty is due to the virtual impossibility of dealing with the aeroelastic problem on such a thin, highly swept wing. Even with external wing tanks fitted, of 150 gallons capacity each (as shown in Fig. 33), it is necessary to increase the t/c ratio of the wing to 4% and to fill the complete wing from centre line to tips with fuel in order to just meet the ferry range requirement without any margin for contingencies.

4. It will be seen from the drawing that the fuselage length of this aircraft requires to be longer than the length of the 1,200 sq. ft. version in order to fit fuselage fuel tanks so as to balance the fuel in the wing. The extra weight incurred in this manner can only be taken off again by the deletion of all transport joints, i.e. making the fuselage and wing as one component each.

5. It was previously shown in para 2.2 that unless the wing main spar box is carried through the fuselage, the weight of a low wing would be greater than for a high wing. In view of this and the fact that this main spar box also contains fuel where it passes underneath the engine, the engine accessibility in the lower region is virtually non-existent. Since large access doors in the stressed monocoque fuselage are not permissible for a minimum weight aircraft, the engines will have to be removed through the rear-end for servicing, with all its attendant disadvantages.

6. It will also be necessary to crowd the armament and avionics as had to be done on the 1,000 sq. ft. high wing version (refer to paras 4.4.4.2 and 4.4.5) with its attendant disadvantages, although it was found possible to install the rockets in front of the missiles. The latter is feasible beaause of the long fuselage required to balance the airplane.

7. A weight and performance summary for this aircraft is given in Tables 16 and 17 respectively. It will be seen that performance figures are becoming somewhat marginal in some respects.

8. It may be concluded that the penalties involved in carrying weight reduction to this extreme are out of all proportion to any gains achieved. Therefore it is felt that an aircraft of this type cannot really be considered a practical proposition in anything but a study of this nature, where it is desired to find a theoretical minimum weight.

TABLE 16
C 105/900
WEIGHTS SUMMARY

ITEM	WEIGHT LB.
ENGINE AND AFTERBURNER (LONG INSTALLATION)	9,702
POWER PLANT - FIXED ITEMS:	
Fuel Tanks	300
Fuel System	420
Fire Extinguishers	65
Accessory Gears and Drives	15
Engine Controls	20
GROUP TOTAL	820
EQUIPMENT	
Instruments	50
Probe	50
Surface Controls	650
Hydraulic System	640
Electrical System	700
Radar and Electronics	1,950
Ejector Seat	132
Emergency Provisions	15
Oxygen	20
Air-conditioning and L.P. Pneumatics	625
Anti-icing System	300
Brake Parachute	75
Exterior Finish	75
Crew	230
Oil	40
Residual Fuel	225
Armament Provisions	410
Armament - Rockets	520
Missiles	792
GROUP TOTAL	7,499
STRUCTURE	
Fin	900
Fuselage	5,700
Wing	5,749
Undercarriage	1,960
GROUP TOTAL	14,311
OPERATIONAL WEIGHT EMPTY	32,330
FUEL FOR COMBAT MISSION	13,300
GROSS WEIGHT	45,630

TABLE 17

C 105/2 - 900 SQ. FT. - 2RB106 ENGINES - 4% T/C.

PERFORMANCE

Gross Weight		Lb.	45,600
FUEL LB.	Supersonic Mission[1]		12,700
	Subsonic Mission[2]		13,300
	Long Range Mission[3]		22,700
INTERNAL FUEL CAPACITY		Lb.	20,400
SIZE OF EXTERNAL TANKS REQUIRED[4]		Gals.	300
RANGE WITH 500 GAL. EXTERNAL TANKS[4]		N. M.	1,600
COMBAT CEILING FT. 1/2 FUEL WEIGHT	0.95 M.		51,800
	1.50 M.		63,400
	1.75 M.		66,200
TIME TO 50,000' FROM STANDING START		Mins.	3.2
'G' AT 50,000' AT 1.5 M.N. AT 1/2 FUEL WEIGHT			1.95
LANDING DISTANCE FROM 50 FT. - FT. STANDARD DAY	1/2 Fuel Weight 5 Min. Fuel Reserve Wt.		7,000 5,950
TAKE-OFF DISTANCE TO CLEAR 50 FT. - FT. HOT DAY	Overload Weight Gross Weight		4,550 3,050

NOTES:
1. Supersonic Mission as Detailed in Table 5.
2. Subsonic Mission as Detailed in Table 6.
3. Long Range Mission as Detailed in Table 7.
4. It is unlikely that External Wing Tanks can be made satisfactory.

APPENDIX B

Engines Outboard Configurotion

During the last meeting with the R.C.A.F.[6], a configuration was discussed which attempted to get around some of the snags inherent in the 900 sq. ft. low wing configuration as as presented in Appendix 'A' of this report. The argument was that by positioning the engines outboard on the wings, weight could be saved because of bending moment relief and also solve the undercarriage stowage problem by retracting the single main gear into the fuselage with outriggers in the nacelles; at the same time engine accessibility would be good.

A drawing of this configuration is shown is Fig. 34, which is self-explanatory.

The main disadvantages of this design as compared with the orthodox configuration are as follows:

1. A tremendously large fin area is required to cater to the one-engine-inoperative condition. This adds weight and drag.
2. There is some possibility of choking of the airflow between the three fins at high speeds.
3. The interference drag is bound to be higher with this configuration.
4. Installing the engines in separate bodies require 58% more wetted area and 23% more frontal area.
5. The adequacy of lateral control is very much open to question.
6. The small fuselage cross-section will jeopardize the installation of armament, avionics and equipment.
7. It has been found impossible to balance this configuration without excessive lengthening of the front fuselage.
8. Even if none of the above disadvantages were present, aeroelastic considerations rule out the feasibility of attaching a heavy pod to an extremely thin wing in the speed bracket we are considering.

APPENDIX C

Single Engine Version with a Bristol BE.23 Engine

Since the issue of the C 104/1 brochure[4], Bristols have started the design of an engine with a breathing capacity 50% in excess of the B.OL.4. It was accordingly thought that, with this new engine, the BE.23, a single engine aircraft could be designed that would not be as marginal in some respects as the C 104/1.

A proposed layout for this airplane is shown in Fig. 35. The configuration is in general very similar to the C 104/1. Due to the extra breathing capacity of the engines, the ducts had to be considerably enlarged. Because the engine is somewhat heavier, and required a longer and heavier fuselage to balance it, the wing area was increased from 600 to 750 sq. ft. It is evident that the good features of the engine and electronic installations of the twin engine versions cannot be retained for the single engine low wing layout.

The weights are given in Table 18 and the performance in Table 19.

Although there is no doubt that going to a single engine layout is the only way to reduce the gross weight of the aircraft below 45,000 lb., it has several very serious drawbacks, which may be enumerated as follows.

1. Performance

As can be seen from Table 19 the performance is very much inferior to that of the twin engine versions. It should be noted that the figures given by Bristols are for a simple nozzle, and accordingly should be compared with those for the twin engine version with the B.OL.4.

There is no margin of fuel capacity available for contingencies for the short range missions, even with a 4½% t/c wing. Hence, the chances of getting as good results with camber as for the 3% wings on the twin engine version is very much reduced.

2. External Tanks:

As discussed in para 4.4.2 the fitment of external tanks on a low wing aircraft with such a thin highly swept wing may well be impossible for aeroelastic reasons. Accordingly, this airplane cannot be counted on for long range missions.

3. Installations

The installations of the engine and electronic equipment might be classed as reasonably satisfactory but servicing will be very much more difficult than with the twin engine version.

4. External Missiles

The penalties due to fitting external armament will be more severe than for the large aircraft inasmuch as the armament is a larger proportion of the total drag.

TABLE 18
C 105/750

WEIGHTS SUMMARY

ITEM	WEIGHT LB.
ENGINE AND AFTERBURNER	6,000
POWER PLANT - FIXED ITEMS:	
Fuel Tanks	212
Fuel System	220
Fire Extinguishers	35
Accessory gears and drives	8
Engine controls	10
GROUP TOTAL	485
EQUIPMENT	
Instruments	50
Probe	50
Surface Controls	650
Hydraulic System	640
Electrical System	700
Radar and electronics	1950
Ejector Seat	132
Emergency provisions	15
Oxygen	20
Air-conditioning and L.P. pneumatics	588
Anti-icing system	300
Brake parachute	75
Exterior finish	75
Crew	230
Oil	40
Residual fuel	225
Armament provisions	410
Armament - Rockets	520
Missilies	792
GROUP TOTAL	7462
STRUCTURE	
Fin	530
Fuselage	5303
Wing	4700
Undercarriage	1700
GROUP TOTAL	12233
OPERATIONAL WEIGHT EMPTY	26180
FUEL FOR COMBAT MISSION	11100
GROSS WEIGHT	37,280

TABLE 19

C 105/2 - 750 SQ. FT. - BE.23 ENGINE - 4.5% T/C

PERFORMANCE

Gross Weight	Lb.	37,400
FUEL LB.	Supersonic Mission[1]	11,200
	Subsonic Mission[2]	10,750
	Long Range Mission[3]	19,400
INTERNAL FUEL CAPACITY	Lb.	11,200
SIZE OF EXTERNAL TANKS REQUIRED[4]	Gals.	1,100
RANGE WITH 500 GAL. EXTERNAL TANKS[4]	N.M.	1,160
COMBAT CEILING FT. 1/2 FUEL WEIGHT	0.90 M.	48,500
	1.50 M.	53,000
	1.75 M.	53,000
TIME TO 50,000' FROM STANDING START	Mins.	5.5
'G' AT 50,000' AT 1.5 M.N. AT 1/2 FUEL WEIGHT		1.22
LANDING DISTANCE FROM 50 FT. - FT. STANDARD DAY	1/2 Fuel Weight	6,850
	5 Min. Fuel Reserve Wt.	5,820
TAKE-OFF DISTANCE TO CLEAN 50 FT. - FT. HOT DAY	Overload Weight	6,050
	Gross Weight	3,600

NOTES:
1. Supersonic Mission as Detailed in Table 5.
2. Subsonic Mission as Detailed in Table 6.
3. Long Hange Mission as Detailed in Table 7.
4. It is unlikely that External Wing Tanks can be made satisfactory.