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CANADIAN GOVERNMENT PROGRAM
FOR THE AVROCAR
PHASE 1

PERFORMANCE

500/PERF/427

AVRO AIRCRAFT LIMITED

500/PERF/427

CANADIAN GOVERNMENT PROGRAM FOR THE AVROCAR

PHASE I

PERFORMANCE

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The number of pages in this report including Title Page and Table of Contents, is 59.

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1.0 INTRODUCTION

This program began with the investigation of a re-engined version of the Avrocar with three T 58 engines (shaft drive and gas generator versions) in place of the J 69's. However, this did not produce attractive designs consistent with the program objective of up to four hours duration with useful economics. This objective was combined with VTO as a desirable but non-essential feature. Therefore after the initial study, effort was directed to developing two distinct concepts involving a GETOL and a VTOL vehicle. A detailed account of this development will be found in Ref. 1.

Examples of the first concept are found in Figs. 1 and 2. This was that ground effect take-off and landing (GETOL) would allow a more economical vehicle. Although there is an undeniable operational limitation, the low power of the two Astazou engines and the consequent savings in engine and fuel weight have led to a relatively cheap and simple aeroplane. Aside from the power reduction this aircraft differs from the Avrocar in using a higher aspect ratio wing and propellers for horizontal thrust.

The second concept was simply the provision of enough power to take-off vertically with the large weight of fuel required for a long endurance. This led to the designs in Figs. 19 and 27. Their two J 85's are nearly twice as powerful as the three J 69's of the Avrocar and their all up weight is considerably greater. There is sufficient power for speeds up to the region of the critical Mach number.

2.0 GETOL VERSIONS - 2 ASTAZOU II

2.1 Original Version

The simplest form of the GETOL version, illustrated in Fig. 1, relies upon the peripheral jets to provide a ground cushion for hovering and those same jets, inclined rearward, to provide first a jet flap and then horizontal thrust. Figs. 3, 4 and 5,(obtained from Ref. 2) show thrust, drag, and fuel flows against Mach number for operation at sea level and 10,000 feet. The top speed is seen to be just over M = .35 at both altitudes.

This version is not pursued further because the extra weight of propellers was deemed acceptable. The data given in the graphs referred to above remains useful, however, since it can be used to describe the hovering case when no power is supplied to the propellers. Since the same engines (522 SHP and .66 SFC at take-off, 473 SHP and .69 SFC at maximum continuous power) are used for the propeller variants the fuel flow data, Fig. 5, remains valid.

2.2 Propeller Versions

Ref. 1 lists several Astazou powered, propeller driven GETOL aircraft, but Fig. 2 shows the final configuration and is fairly representative, performance-wise of the others.

The thrust with the propellers was taken to be $\frac{.85 \times SHP \times 550}{V}$ so that the thrust horsepower is 85% of the shaft horsepower. The maximum thrust occurs at zero speed and was arbitrarily limited to 3.5 lb. per SHP. The propeller thrust curves with the drag curves superimposed (from Ref. 2) are shown in Figs. 6, 7 and 8.

The top speed is very little higher than that shown for the original version. The increased drag and weight nearly makes up for the extra thrust of the propellers at top speed but the propeller version has a significantly greater specific air range. (Nautical air miles per pound of fuel).

Since the cockpit is not to be pressurized operation will not normally exceed 10,000 ft. altitude. However, data for flight at 20,000 ft. has been included to show what can be done under extra-ordinary conditions.

2.3 Performance in the Ground Effect

It is expected that the ground cushion will be used, not only for take-offs and landings, but also to allow extended duration for scouting over moderately smooth country. Then we need to know how high and how fast we can move over the country and what the fuel flow will be. The answers to these questions are supplied by Figs. 9 and 10 which assume no aerodynamic lift and no benefits from intake ram recovery, valid assumptions for very low speeds.

2.3 Performance in the Ground Effect (cont'd)

A low forward speed can be maintained without directing any of the engine power to the propellers, (i.e. zero pitch) simply by the residual thrust of the Astazou engines. However, to make any real headway a percentage of the power must be directed away from the fan to the propellers.

For comparison, note that the maximum endurance, flying clear of the ground at sea level, involves a consumption of about 370 lb. of fuel per hour, or 260 lb. per hour on a single engine. Single engined flight at low speed, near the ground or away from it, would only be possible if the mechanical interconnection is such that one engine can drive both propellers without motoring the other engine. Such a feature has not been planned initially but may be needed to provide a suitably low minimum controllable airspeed in the event of an engine failure on take-off.

2.4 Take-Off Performance

Ref. 3, dealing with a design basically similar to the propellerless version of Fig. 1, shows how a take-off over a 50 ft. obstacle takes 566 ft. This is for 4,400 lb. all up weight and a ground clearance during the take-off run of 3 ft. The corresponding landing distance is 357 ft.

For the propeller driven aircraft of Fig. 2, with its greater weight at take-off (6,800 lb.), a take-off run at 3 ft. may be possible but would take excessively long since 95% of maximum continuous engine power is required just to hover at this height. (This would leave about 15% power for propulsion, since take-off power is about 110% of maximum continuous). A take-off run at 2 ft. is not unreasonable for this small aeroplane and would only require about 66% maximum continuous power for lift and 44% power for thrust, while a run at 1.5 ft. would mean that the power could be apportioned 50% for lift and 60% for thrust.

The values of augmentation used: 3.0 at 3 ft., 4 at 2 ft., and 5 at 1.5 ft. come from Ref. 4. Dividing weight by augmentation gives the required vertical thrust and the corresponding fan power required can be read from Fig. 3 for M = 0. The vertical thrust does not fall off with forward speed and so if no aerodynamic lift is assumed during ground cushion operation a constant power is required for constant height.

Let us assume that the take-off is divided into 3 phases:

1. Ground run to unstick speed, calculated for a height of 1.5 and 2 ft. During this phase the engine power is practically constant and the division of power between fan and propellers is fixed. No horizontal thrust is assumed from the lifting jets and their momentum drag must be overcome by the propeller thrust.

2.4 Take-Off Performance (cont'd)

2. Acceleration to climb speed. For this phase the fan power will be reduced to 50%, leaving 60% maximum continuous power for the propellers. Since the fan flow feeds a jet flap deflected to about 60° then 50% of the nozzle thrust acts rearward, aiding the propeller thrust and the aerodynamic lift required is reduced by 86.6% of the nozzle thrust.

3. Climb to 50 ft. at constant speed. Several different climb speeds are studied to find the speed at which the sum of distances (2) and (3) are a minimum. During the climb-out the power to the fan, i.e. to the jet flap, is reduced further to 30% but the flap angle is the same.

The profile drag is based on a C_{D_o} of .0156 and the momentum drag has been assumed to be the difference between the nozzle thrust at speed and its static value recorded on Fig. 3. The residual thrust of two Astazous at take-off power is 160 lb. The induced drag will be about .182 $C_L{}^2$ where C_L is based on weight minus vertical thrust component. Then the performance is calculated by a step-by-step integration.

For the shortest take-off the aircraft should unstick at V_1 = 90 ft/sec. The corresponding ground run, S_1 = 780 ft. at a 2 ft. height, or 525 ft. at 1.5 ft. An earlier lift off would be possible because the jet flap allows high lift co-efficients but because of the induced drag it is preferable to stay in the ground effect until the climbing speed is nearly reached.

S_2, the distance for transition to climbing flight takes 140 ft. while the aircraft accelerates from 90 ft/sec. to 95 ft/sec. (V_2), the best climb speed. The climb angle is 7 degrees and the distance to clear a 50 ft. obstacle, S_3, is 400 ft. Then the total take-off distance is 1320 ft. with a 2 ft. run and 1065 ft. with a 1.5 ft. run.

Since the time taken for take-off is 19.9 and 15.4 seconds respectively, and since the fuel flow to two Astazous at take-off power is 689 lb/hr, the fuel consumed is only 3.8 lb. for a 2 ft. take-off and 3.0 lb. for a 1.5 ft. take-off.

2.5 Climb Performance

The rate of climb is V sin θ and the horizontal speed is V cos θ where:

$$\theta = \sin^{-1} \frac{\text{thrust - drag}}{\text{weight}}$$

2.5 Climb Performance (cont'd)

Consider two climbing weights; 6,800 lb., representing the maximum all up weight, and 6,200 lb., representing the weight when returning from an unrefueled mission. For simplicity and to obtain a conservative result, these weights are held constant for each climb. Now from the data of Figs. 6, 7 and 8 we can plot the rate of climb and the horizontal speed for each weight at sea level, 10,000 ft., and 20,000 ft. (Fig. 11). Then taking the optimum points from Fig. 11 we construct Fig. 12, the estimated variation of rate of climb, horizontal speed, and fuel flow during an optimum climb to 20,000 feet at maximum continuous power. A step by step integration under the curves of Fig. 12 yields the following information:

	Climb to 10,000 ft.		Climb to 20,000 ft.	
	6,800 lb.	6,200 lb.	6,800 lb.	6,200 lb.
Time taken (min.)	5.0	4.5	13.8	12.5
Fuel consumed (lb.)	48	43	112	101
Distance covered (n.m.)	11	10	32	29

2.6 Cruise Performance

In order to find the specific air range at a certain Mach number note the power required from Figs. 6, 7 or 8, as applicable, and the corresponding fuel flow from Fig. 5. The nautical air miles per pound (NAMPP) is $\frac{\text{speed (knots)}}{\text{fuel flow (lb/hr)}}$.

Curves of NAMPP vs Mach No. are found in Figs. 13, 14 and 15. Estimated specific air range for the Avrocar, as given in Ref. 6, is also plotted at sea level and 10,000 ft.

Note that at sea level and at 10,000 ft. (Figs. 13 and 14) there is a second, higher curve marked 'single engine'. Better specific air range is possible on one engine because the engine operates at a high efficiency (near maximum power) at the same time as the airframe is near its best lift/drag speed. At 20,000 ft. there is not sufficient power to maintain height on one engine.

It would not likely be practical to shut down an engine for the cruise without the mechanical interconnection that will allow one engine to drive both propellers without turning over the other engine. If only one engine is powered the drag of the other propeller and the rudder trim drag would tend to cancel the extra economy of single engine operation.

2.7 Payload - Range

Still air range is calculated by adding the distance covered in the climb to the range possible on the fuel remaining after the climb. No allowance is made for reserves but 12 lb. of the 960 is allowed for ground handling and take-off (i.e. 1 min. at full power).

There are a number of possible modes of descent ranging from the long, flat glide to a complex instrument let down procedure. The simple assumption used, that no fuel is consumed in the descent but no ground is covered either, is fairly conservative.

Only a single weight has been used for the specific air range curves but this value (6,600 lb.) is never very far from the actual weight which is a maximum of 6,800 lb.

With 400 lb. of payload (the second pilot and 200 lb. of cargo) the aircraft can take-off with full tanks and can perform as follows:

Cruise altitude	(ft.)	0	10,000	20,000
Climb distance	(n.m.)	0	11	32
Fuel for climb	(lb.)	0	48	112
Fuel for cruise	(lb.)	948	900	836
NAMPP at Max. range		.406	.516	.585
Cruise range	(n.m.)	385	464	489
Total range	(n.m.)	385	475	521

There are a number of possible variations on this mission. One engine might be shut down. Extra fuel tanks might allow 400 lb. more fuel in place of the payload. More than 400 lb. payload could be carried if the fuel load was reduced. All these possibilities, as well as the basic payload vs range graphs, are shown in Figs. 16 and 17.

2.8 Typical Mission

The results of sections 2.3 to 2.6 can be assembled in such a way as to make up a mission profile that might be typical of those required. The total fuel weight is 960 lb. but 96 lb. will be set aside as a reserve. 164 pounds are allowed for scouting at the outer end of the mission. This might be said to correspond to a half an hour's endurance but this, of course, depends on the engine handling during this time. It allows both engines to run at 20% power for half an hour or for one engine to run at maximum continuous power. Zero fuel and zero range descents are assumed.

At the top of Fig. 18 a schematic mission profile is shown. Below are plotted the radii of action possible for three different cruising altitudes, 20,000 ft., 2.8 Typical Mission (cont'd)

10,000 ft., and sea level. For each cruising altitude a number of speeds are possible; the top point represents the speed at maximum continuous power, the middle one the maximum range speed, and the bottom point the speed for maximum endurance. Broken lines indicate the radii of action possible during single engine operation (with one engine driving both propellers). There is insufficient power to maintain 20,000 ft. on one engine and at 10,000 ft. only limited performance is possible.

3.0 VTOL VERSIONS - 2 J85's

3.1 Circular Version

After the investigation of a large number of related designs (see Ref. 1) work was concentrated on the aircraft shown in Fig. 19. This design differs from most of its predecessors in having low mounted engines exhausting upward through the tip turbine and then directly to atmosphere.

It was planned that 40% of the fan air flow would be directed to the peripheral nozzles while the other 60% passed out the low loss central jet. The areas of these nozzles were fixed in each flight mode. That is, for hovering, the central jet area was 2300 sq. in. and the peripheral jet area was 1730 sq. in., while for forward flight the values became 1310 and 1400 sq. in. On this basis, and using a drag coefficient calculated according to standard aircraft methods, the thrust and drag vs Mach number curves at sea level and 20,000 ft. were produced. (Figs. 20 and 21). The corresponding engine fuel flows are found in Figs 22 and 23.

A fixed nozzle area for all forward flight conditions fails to ensure operation always at the optimum pressure ratio. The introduction of a variable peripheral jet area, as set out in Fig. 24, can produce the new sea level thrust vs Mach number curve of Fig. 25. Note the improved performance possible, particularly at higher speeds, with this simple innovation. The central jet is not required to adjust other than to go from hovering to forward flight configuration.

At a late stage in the design, wind tunnel tests indicated that the central jet should pass not more than 20% of the fan flow and 80% should go through the peripheral nozzles for control. Such a change decreases the thrust of the vehicle because there are higher losses in the passage to the periphery than in the short central jet passage. The variation of hovering thrust over the entire range of flow division is shown in Fig. 26. Note that the change from 60% to the central jet to 20% to the central jet lowers the thrust from 11,400 to 10,730, a 6% reduction.

This is a gross thrust reduction and so for forward flight the net thrust, which is the gross thrust minus the momentum drag will fall even more. (The momentum drag is independent of the flow division but increases with speed).

Performance is presented on the basis of Figs. 20 and 21. In order that the reduced central jet does not invalidate these curves we must assume that the full advantage of adjusting the peripheral nozzle area with speed will be realized. Then Figs. 20 and 21 can be taken as approximately correct. A more comprehensive study of these problems will be found in the Propulsive System Analysis, Ref. 2.

3.2 Winged Version

The additional short, swept wings from each side of the Avrocar type discus (Fig. 27) was suggested as an alternative to gyro stabilization for longitudinal stability (see Ref. 5). The increased aspect ratio allows increased range and endurance. The improved low speed qualities will lower the speed for conversion from vertical take-off to horizontal flight and will shorten the ground run when a ground effect take-off is to be used. There will be, as a result of the enlarged wing area, an increase in weight and a reduction in the top speed.

No significant thrust change is expected from the new layout; the hovering system is unchanged and in forward flight the extra duct length will compensate for the better nozzle positioning. Then Figs. 20 and 21 are modified to include the drag curves of the winged version and reappear as Figs. 28 and 29.

In subsequent sections both versions will be considered.

3.3 Performance in the Ground Effect

Although at normal operating weights this aircraft can take-off and climb vertically, support by the ground cushion is seen as enabling overload take-offs after curshion-borne ground run and economical ground level scouting. A weight of 8,500 lb. has been taken for this section, a weight that might correspond to fuel remaining at the outer end of a typical mission. (See section 3.8 of this report). Obviously, for the ground run of an overload take-off the ground clearance would be less than Figs. 30 and 31 indicate.

Figs. 30 and 31 show the height and speed possible for various engine power settings and various horizontal components of the fan thrust. Note that even when there is no horizontal component of thrust from the central and peripheral nozzles there can be a sizeable forward speed from the residual thrust of the turbine exhaust. The total thrust is $\frac{T}{L}$ x 8,500 lb. + residual thrust, where $\frac{T}{L}$, the horizontal thrust over the weight supported, comes from Fig. 5 of Ref. 3. $\frac{T}{L}$ is a function of the shift of the focus of the peripheral jets or the inclination of the vehicle.

Note that 95 and 100 percent rpm are not included in Fig. 30 because this would be enough to get the vehicle out of the ground effect entirely. Fig. 31 shows how much more efficient single engine operation can be. There would be a slight problem of assymetric residual thrust in this case.

These graphs are applicable to either the circular or the winged version at 8,500 lb. The lifting jet system of each design is the same and they both have the same circular base area inside the peripheral nozzles.

3.4 Take-Off Performance

Fig. 20 shows the nozzle thrust at zero speed to be 11,300 lb. Once clear of the ground the flow from the peripheral nozzles must flow in toward the center to form a single jet. ('Tree Trunk' flow). Because of this phenomena and the attendant losses the free air thrust may be 10% less than the nozzle thrust or only 10,200 lbs. This is sufficient thrust to provide VTOL performance for vehicles weighing up to 9,700 lb. (The criterion for adequate VTOL performance is assumed to be that the thrust = 1.05 x weight). Since the circular version weighs 8,230 lb and the winged version 8,820 lb. both these aircraft, at their normal maximum all up weight can enjoy vertical take-off and climb out under standard atmospheric conditions.

The effect of operation at 5,000 ft. will be about a 10% thrust loss. A similar loss will occur when operating in a Mil.Std. hot day (103°F). Then the circular version, at 8,230 lb., is capable of vertical take-offs at altitudes up to 7,500 ft. on standard days, or 2,500 ft. on hot days. The winged version, at 8,820 lb., is capable of vertical take-offs at altitudes up to 4,500 ft. on standard days.

To counter losses due to unfavourable environment and to allow operation at weights in excess of 10,000 lb. this aircraft, like the Astazou powered model, will be capable of taking off obliquely after a short run in the ground cushion. The distance to clear a 50 ft. obstacle under standard atmospheric conditions for an 11,000 lb. circular aircraft, after a run at 4 ft. off the ground, is 280 ft. The winged version requires 150 ft. At 14,000 lb., and with only 3 ft. ground clearance, the circular version requires 1,390 ft. and the winged version requires 510 ft.

The dominant drag term during the low speed climb out is the induced drag which varies inversely with the aspect ratio and the lift coefficient squared. This is why the winged version with greater aspect ratio and wing area is so superior during ground effect take-offs.

Landing distance is not normally critical. In the first place if the landing is conducted after some of the fuel has been consumed it can probably be made vertically. However, ground effect landings can normally be made in considerably less distance than that required for take-off. The jet flap allows low speeds with high induced drag, the fan flow provides momentum drag, and once in the ground effect the peripheral nozzles can be directed to give reverse thrust.

3.5 Climb Performance

Climbs from sea level to 20,000 ft. were calculated for two weights, each of which was assumed constant throughout the climb. These weights were 11,000 lb., an overload case requiring GETOL, and 8,500 lb., the approximate design all up weight and also the weight at the outer end of a mission begun at 11,000 lb.

3.5 Climb Performance (cont'd)

Fig. 32 shows the assumed data for the climb, based upon optimum climb speeds at sea level and 20,000 ft., for military power. A step by step integration under these curves yields the following information:

		Climb to 10,000 ft.		Climb to 20,000 ft.	
		11,000 lb.	8,500 lb.	11,000 lb.	8,500 lb.
Time taken	(min.)	1.36	.93	3.95	2.43
Fuel consumed	(lb.)	98	67	244	152
Distance covered	(n.m.)	6	3	17	9

This section refers only to the circular version. The winged version climbs a little more steeply with less horizontal speed but the difference is not significant.

3.6 Cruise Performance

The specific air range for the circular and the winged versions, one and two engines operating, sea level and 20,000 ft., is shown in Figs. 33 to 36. The single weight chosen, 8,500 lb. (8,800 lb. for the winged version) is about the average weight expected during a typical mission. At higher weights the nautical air miles per pound will be less, and vice versa, although this effect is significant mainly at low speeds.

Single engine operation is seen to be superior under all conditions. However, before it can be recommended, studies should be made of the effect of the trim drag required to balance the assymetric residual thrust and the drag of the intake to the shut down engine. (Fortunately the fan thrust remains symmetrical). Then there is the question of the speed and realiability of the aerial re-light system when a return to two engine operation is necessary.

The effect of the 'wings' added to the basic circular planform can now be assessed. They are seen to provide a greater specific air range but to have slightly reduced the top speed and the speed for maximum range. The winged version has the greatest advantage at very low speeds where the dominant drag term is the induced drag. This explains why the winged version will have a much shorter ground effect take-off than the lower aspect ratio circular version when both are over-loaded for VTOL.

3.7 Payload - Range

The basic design weight of the circular version is 8,230 lb. of which 3,360 lbs. is fuel and 1,200 lb. (second pilot and cargo) is payload. Then, allowing no fuel reserves, the payload vs range graphs for sea level and 20,000 ft., two and one engines operating, are shown on Fig. 37. Similar graphs for the winged version (weight = 8,820 lb., fuel = 3,360 lb., payload = 1,200 lb.) are given in Fig. 38. Note that these weights are well within the VTOL range.

3.7 Payload - Range (cont'd)

The maximum VTOL weight, as established in Section 3.4, is 9,700 lb. Although the basic design weights of 8,230 lb. and 8,820 lb. were set at an early stage it is expected that they will be increased to take advantage of the power available. Therefore, a curve of payload vs range at 9,700 lb. has been added to each set of curves in Figs. 37 and 38. No allowance has been made for a possible slight increase in structural weight to handle the extra weight, all of which has been made up of extra payload or fuel.

We also wish to consider overloaded cases because, given a short and reasonably clear field, sufficient structural strength and useable volume, these aircraft can fly at much higher weights. An increase in the payload will cause a very slight decrease in the range available from full internal tankage. However, if the extra weight goes to fuel, an estimate of the extra range (in nautical miles) at the optimum cruise speed is had by multiplying the weight of the extra fuel by the specific air range (NAMPP) from Figs. 33 to 36. For convenience these values of NAMPP are tabulated below:

	Avrocar (Ref. 6)		Circular Version		Winged Version	
	3 engines	2 engines	2 engines	1 engine	2 engines	1 engine
Sea Level	.097	.128	.104.	.146	.111	.161
20,000 ft.			.135	.169	.142	.211

For example, if the winged version were to take-off at 14,000 lb. how far could it carry its full 1200 lb. payload? Assume a two engined cruise at 20,000 ft. From Fig. 38 the range with 3,360 lb. fuel and full payload is 466 nautical miles. But the extra fuel to be carried is 14,000 - 8,820 = 5,180 lb. Multiplying this by .142 from the Table above we get 735 extra n.m. Then the still air range without reserves will be about 1200 nautical miles.

3.8 Typical Missions

At the top of Fig. 39 is the schematic profile of a typical sea level VTOL mission starting at the maximum VTO weight of 9,700 lb. The curves below show how the radius of action attainable varies with the cruising speed employed for the two configurations, single and twin engine operation. Three points are shown on each curve. They represent, from top to bottom, maximum speed, maximum range, and maximum endurance.

Fig. 40 shows the speed and radius of action for a mission where VTO has been exchanged for increased range and scouting duration.

Allowing for the fuel consumed in the climb it can be shown that the radius of action is about 10% greater at 10,000 ft. and 20% greater at 20,000 ft. for the missions considered. Note that for the same fuel weight the winged version would go further but at the same all up weight it carries less fuel due to its heavier structure.

3.9 Modified Intake Characteristics

Section 4.7 of Ref. 2, discusses the effect on the propulsive system of some early results of a wind tunnel intake test program. One configuration tested was shown to penalize the maximum thrust available at sea level as follows:

Mach Number	.2	.3	.4	.5	.6
Thrust remaining	98%	91%	78%	56%	22%

Developed configurations, using inlet guide vanes, will reduce this loss.

The effect of incorporating these test results would be very marked at high speeds. Maximum speeds would be reduced and high speed cruise would become less economical. At typical cruise speeds, say Mach Numbers of .3 and .4, the engine rpm would have to increase from about 78 and 86% to 82 and 92% at sea level and the corresponding fuel flows would increase by 11 and 35%. Then for the winged version, operating on both engines at sea level, the specific air range would be reduced from .110 to .099 nautical air miles per pound of fuel at M = .3. At M = .4 the reduction would be from .093 to .069 NAMPP. That is, there would be about a 10% reduction in maximum range at sea level (where the maximum range speed is about 200 knots) and a slightly more serious penalty at altitude. With the use of inlet guide vanes the maximum range should be as much as 95% of the values listed in this report.

4.0 CONCLUDING REMARKS

The GETOL aircraft powered by two Astazou II engines makes an attractive light vehicle for operating from unprepared surfaces, capable of speeds up to 240 knots. Since its specific fuel consumption is so superior to the Avrocar's it has a much greater range. Its take-off performance, 1,320 ft. after a run at 2 ft., is not as spritely as might be desired and extra horsepower could be used here to increase the ground clearance, shorten the run, or increase the useful load. In this connection it is worth noting that besides the Turbomeca - Continental Astazou there are the Boeing 520 and the Pratt and Whitney PT6 of similar power and weight, and so there will certainly be 'stretched' engines becoming available.

The J85 powered VTOL designs promise to be extremely useful. The versatility imparted by vertical take-off and the load carrying ability when operating with ground effect take-off as well as the very nimble flight performance counter the disadvantage of the high fuel consumption. Actually the specific air range is comparable with the Avrocar, but the much greater carrying capacity give it very superior payload and range. Since the addition of wing increases specific range and improves GETOL distances immensely, while only slightly limiting top speed and VTOL payload/range, they are advantageous from the performance standpoint.

5.0 LIST OF REFERENCES

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3.	AVRO 000/PROP/1	Research Program Leading to the Design of a U.S. Army GETOL Aircraft
4.	AVRO/ SPG/TR 308 D.B. Garland	Data Report for 1/20th Scale Avrocar Model Focussing Ring Control, Static Tests.
5.	AVRO/S & C/428 I. King	Canadian Government Program for the Avrocar Stability and Control
6.	AVRO/SPG/TR 254	The Avrocar Design

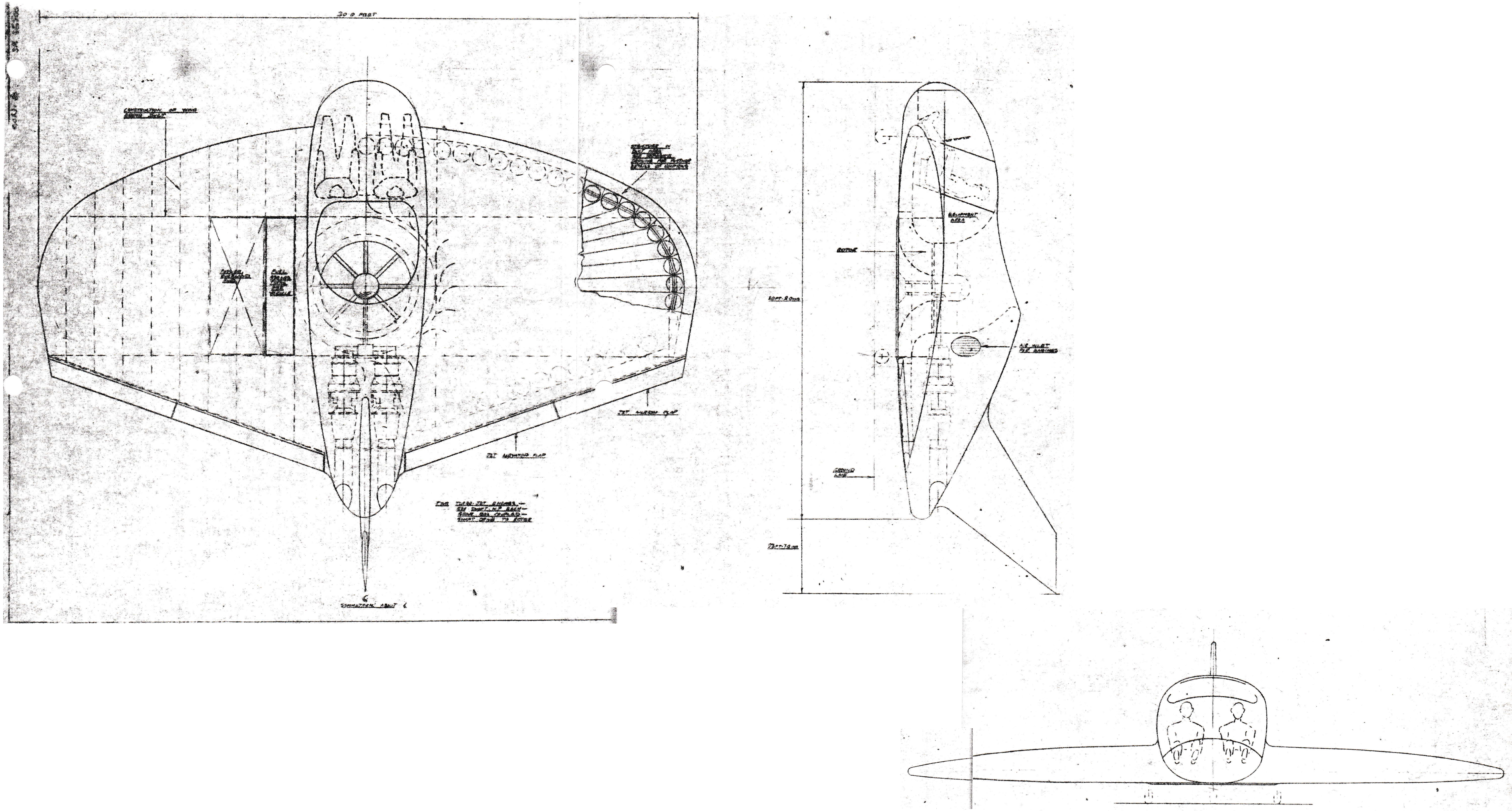


FIG. I

FIG GENERAL ARRANGEMENT

16

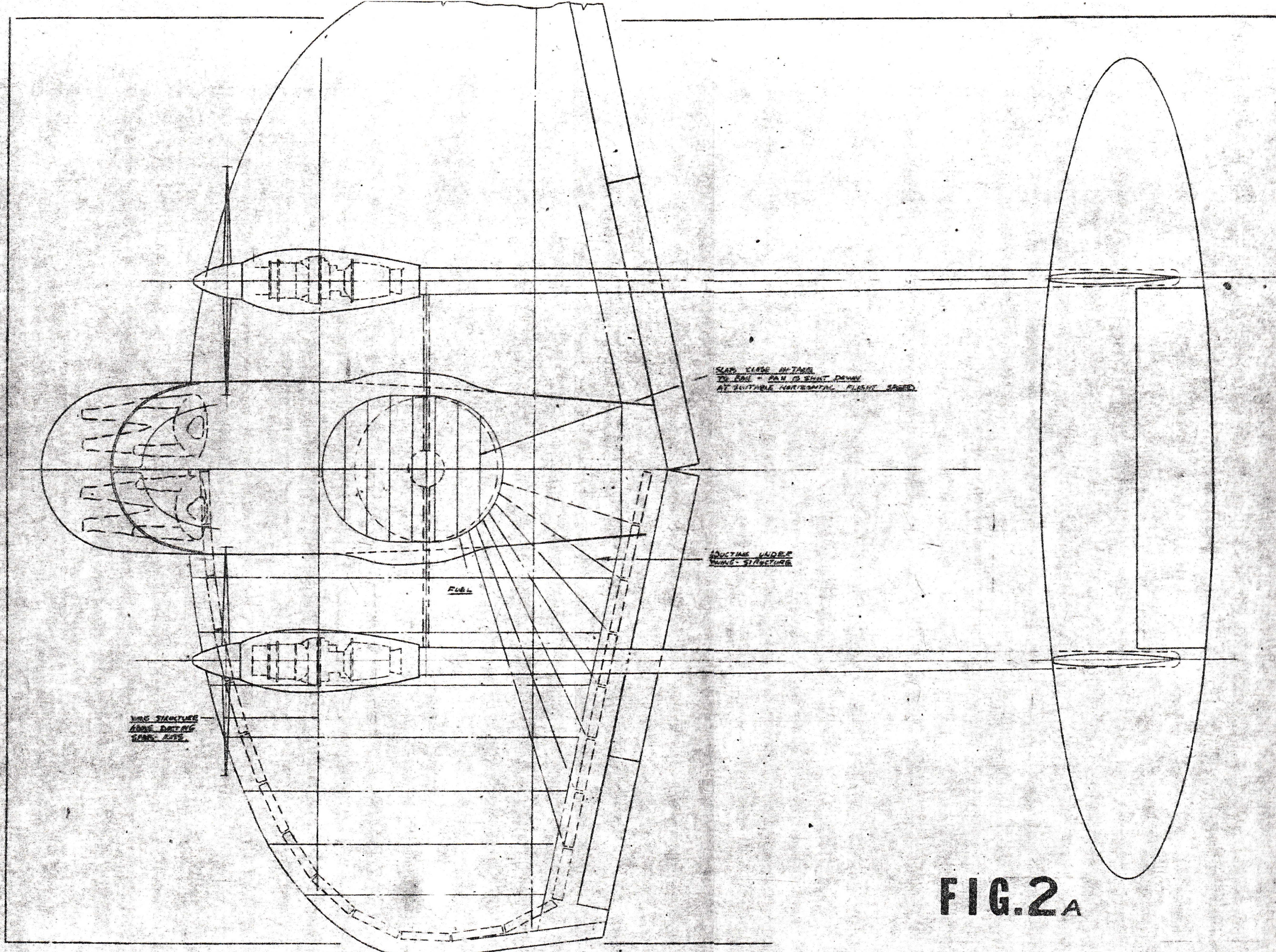
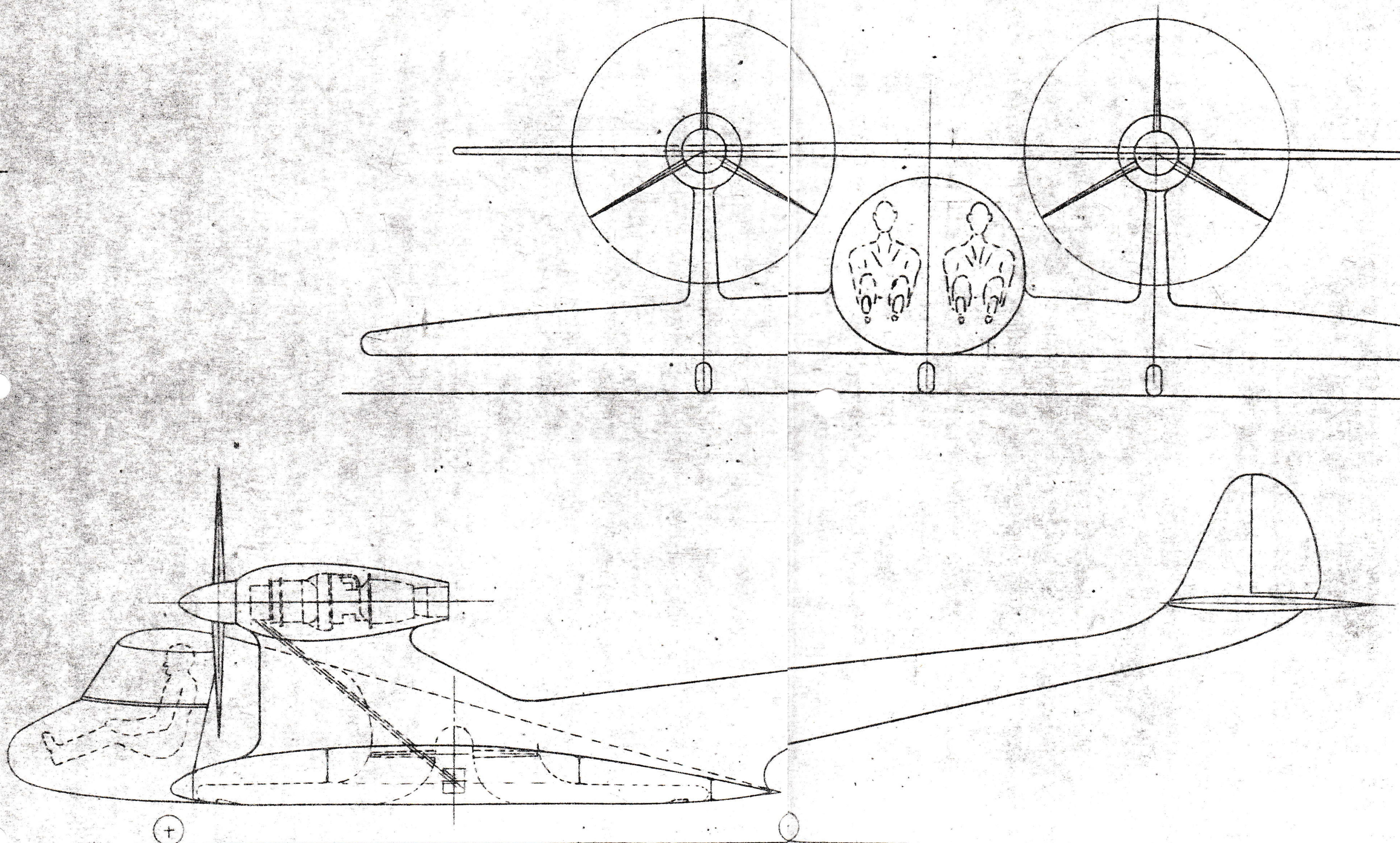

FAN IS SHUT DOWN
AT SUITABLE HORIZONTAL FLIGHT SPEED
DUCTING UNDER
WING-STRUCTURE
FUEL

FIG.2A



SPAN – 30'-0"
O/A LENGTH – 36'-3"
O/A HEIGHT – 15'-0"
WING AREA – 353 SQ.FT.
WING ASPECT RATIO – 2.50
T/C WING – 10%
TAILPLANE SPAN – 25'-3"
TAILPLANE AREA – 106 SQ.FT.
TAILPLANE ASPECT RATIO – 6
T/C TAILPLANE – 8%
2 ASTAZOU II ENGINES – 522 S.H.P. EACH
FUEL – 550 LB.
PRESSURE RATIO–FAN – 1.07
CARGO CAPACITY – 200 LB
GROSS WEIGHT – 6[illegible] LB

FIG. 2 B

THRUST & DRAG VS MACH NUMBER

SEA LEVEL

TWO ASTAZOU II ENGINES

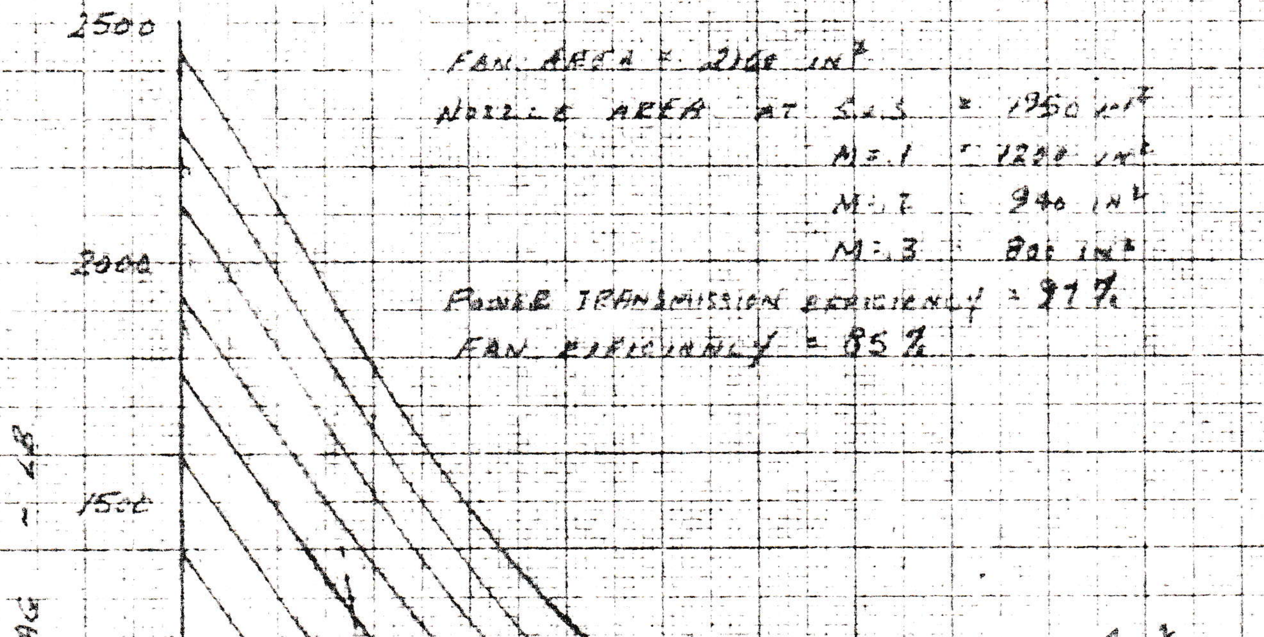


FIG 3

THRUST & DRAG VS MACH NUMBER

10,000 FT.

TWO ASTAZOU II ENGINES

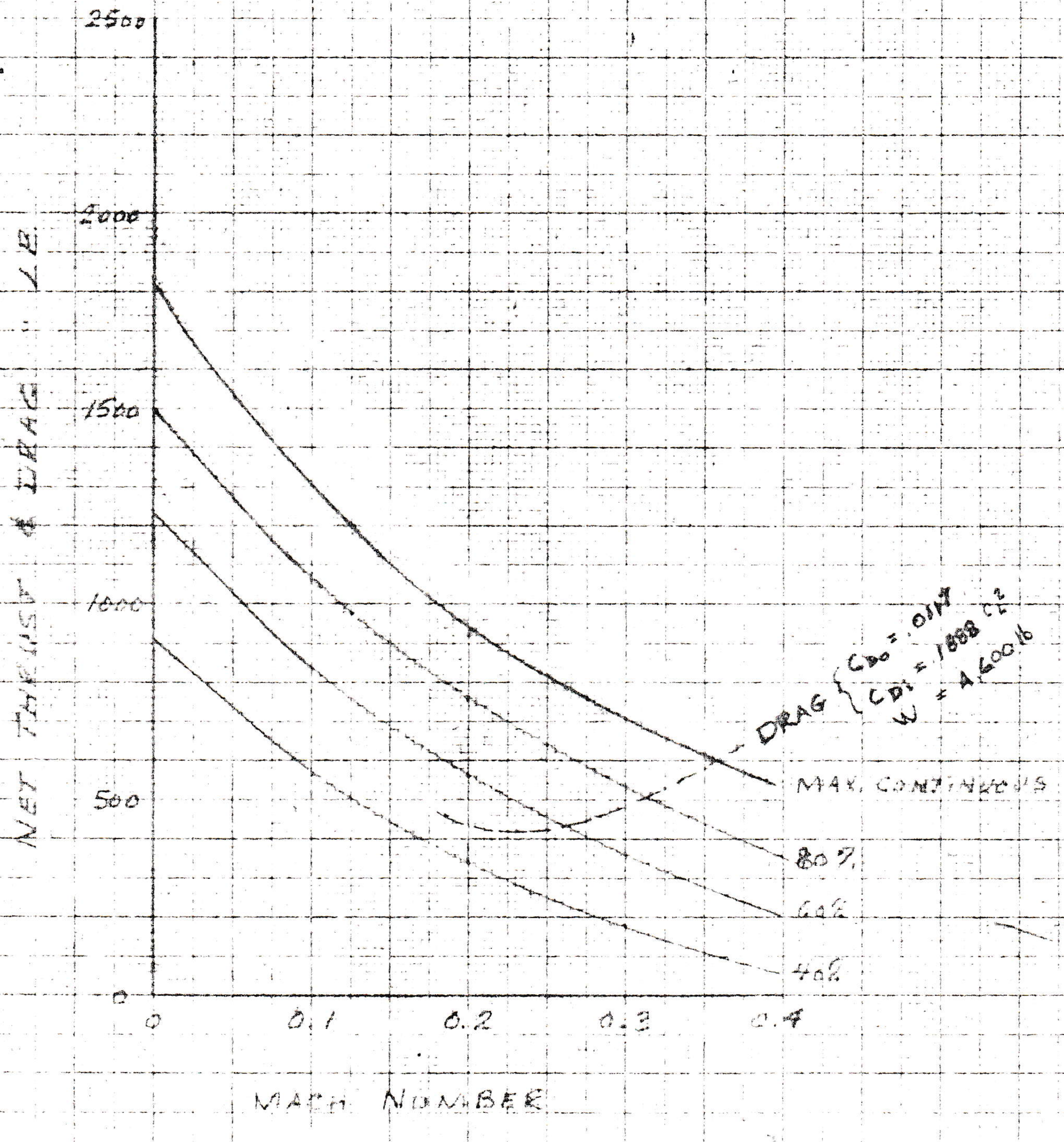


FIG 4

2 ASTAZOU II

FUEL CONSUMPTION VS ENGINE POWER

BASED ON TWO ENGINES OPERATING AT SOME % OF MAX CONTINUOUS POWER

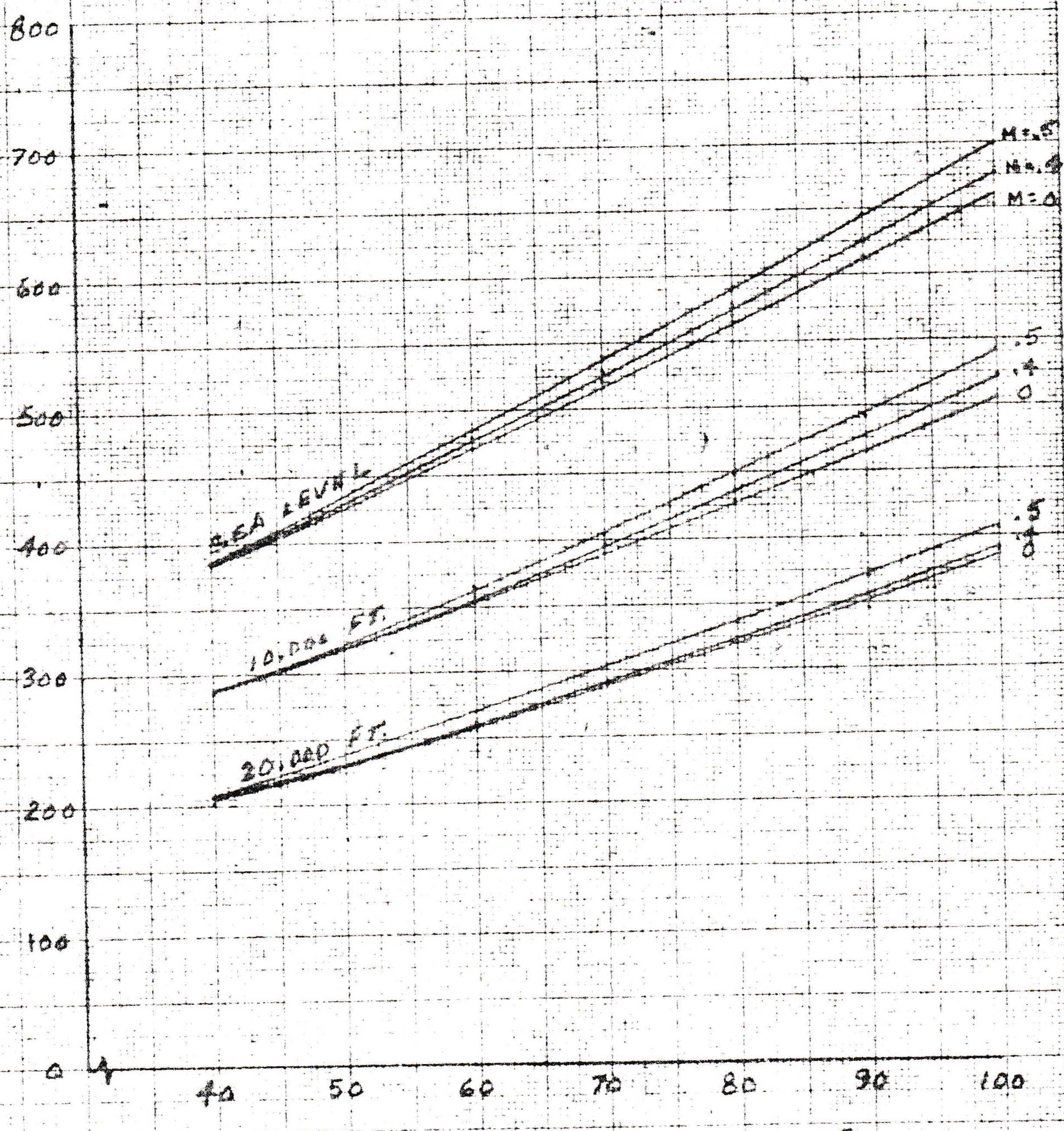
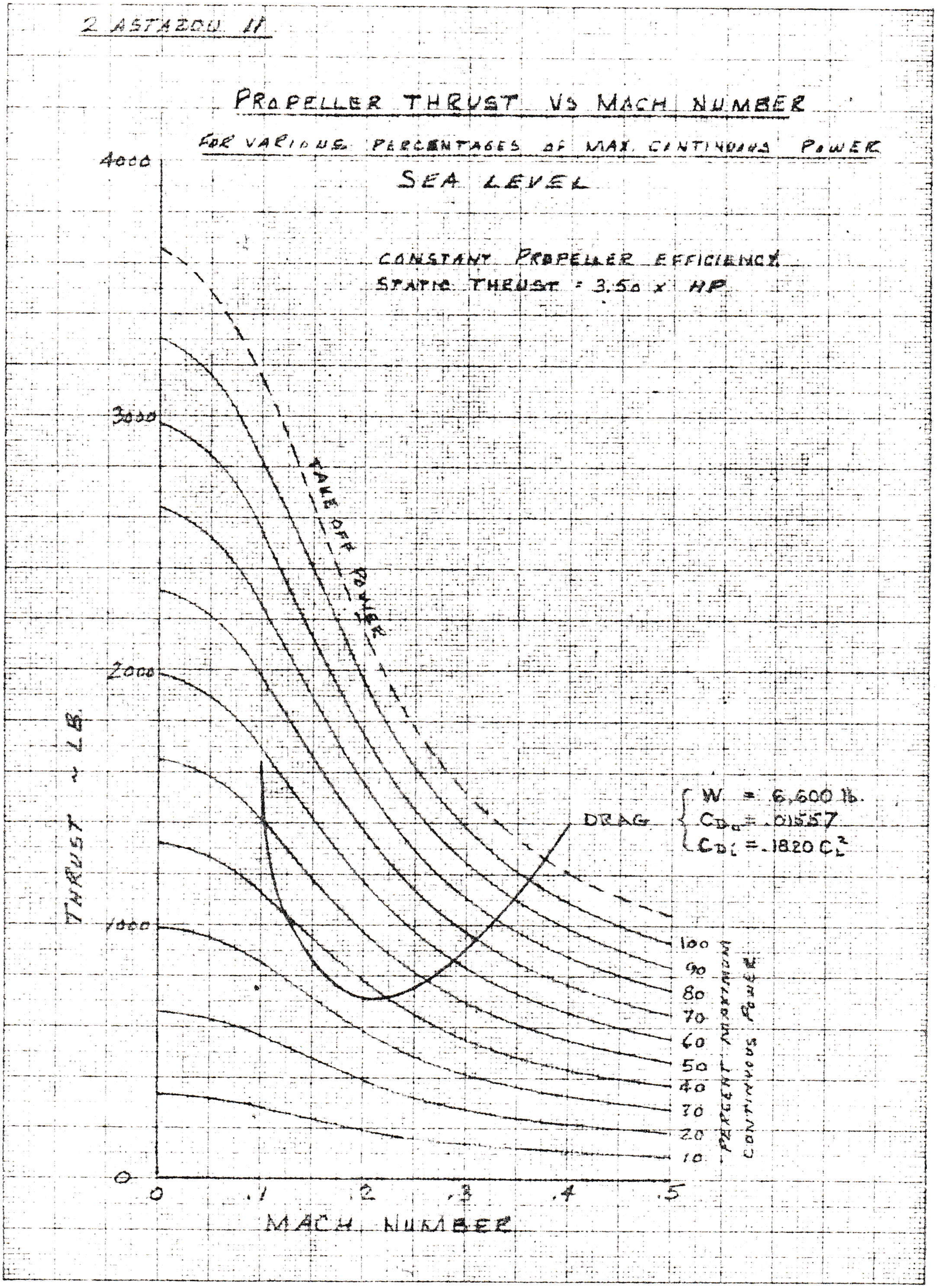


FIG 5



2 ASTAZOU II
PROPELLER THRUST VS MACH NUMBER
FOR VARIOUS PERCENTAGES OF MAX. CONTINUOUS POWER
SEA LEVEL
CONSTANT PROPELLER EFFICIENCY
STATIC THRUST = 3.50 x HP
TAKE OFF POWER
DRAG
W = 6,600 lb.
C_{D_0} = .01557
C_{D_i} = .1820 C_L^2
THRUST ~ LB.
4000
3000
2000
1000
0
0
.1
.2
.3
.4
.5
MACH NUMBER
100
90
80
70
60
50
40
30
20
10
PERCENT MAXIMUM CONTINUOUS POWER

FIG 6

2 ASTAZOU II

PROPELLER THRUST Vs MACH NUMBER

FOR VARIOUS PERCENTAGES OF MAX. CONTINUOUS POWER

10,000 FT

CONSTANT PROPELLER EFFICIENCY

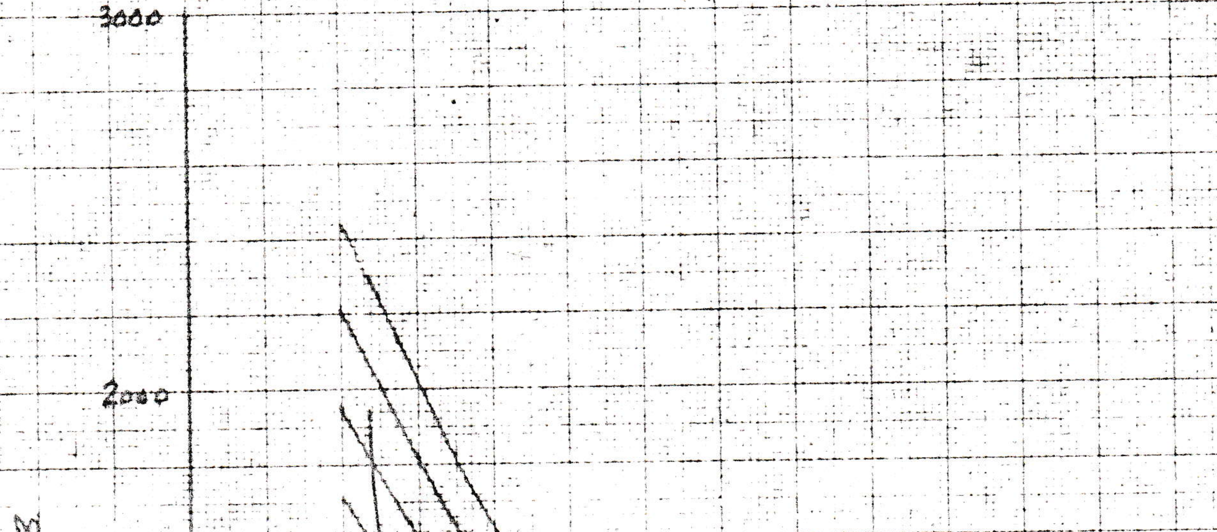


FIG 7

2 ASTAZOU II

PROPELLER THRUST VS MACH NUMBER

FOR VARIOUS PERCENTAGES OF MAX CONTINUOUS POWER

20,000 FT

CONSTANT PROPELLER EFFICIENCY

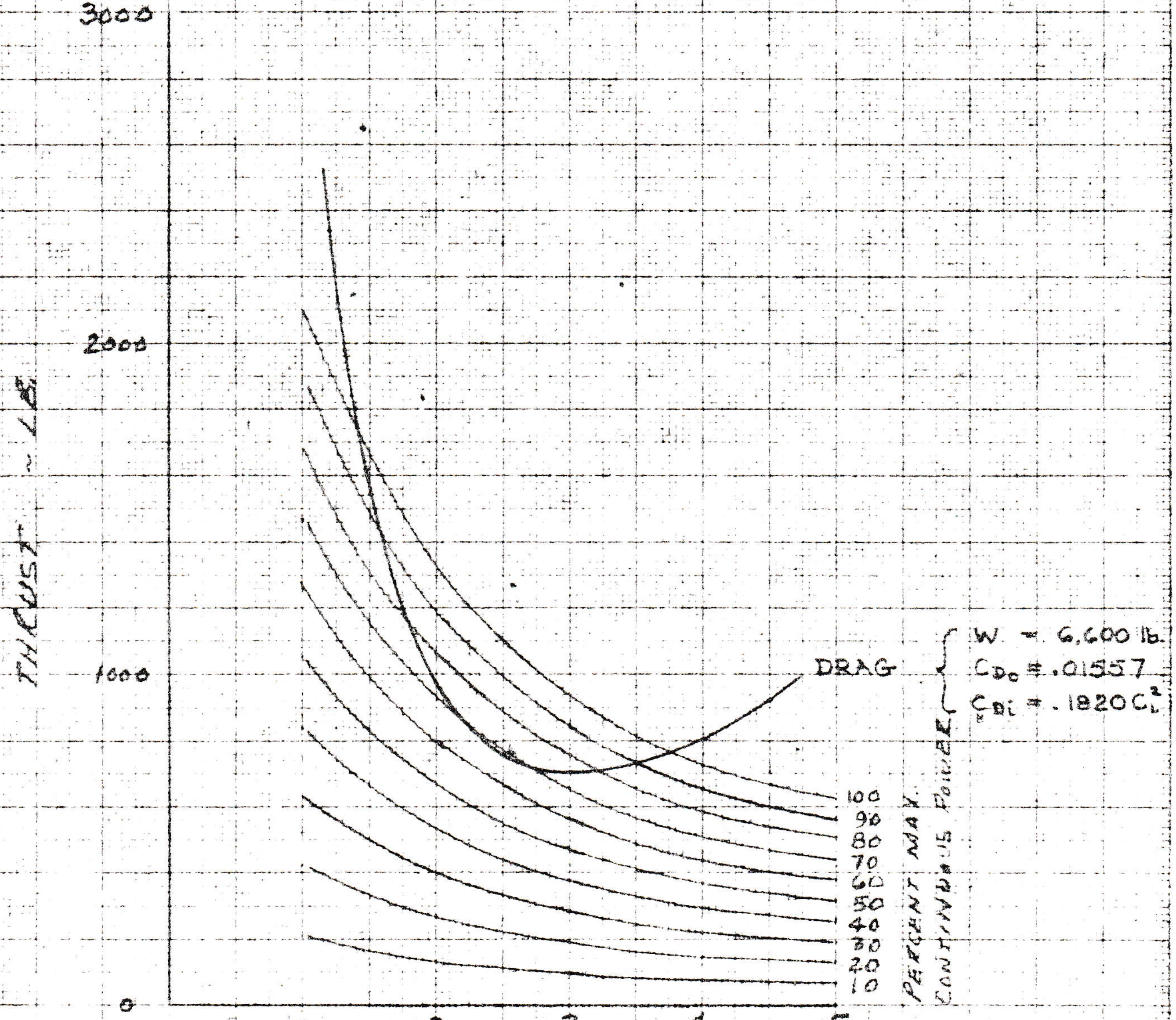


FIG 8

K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

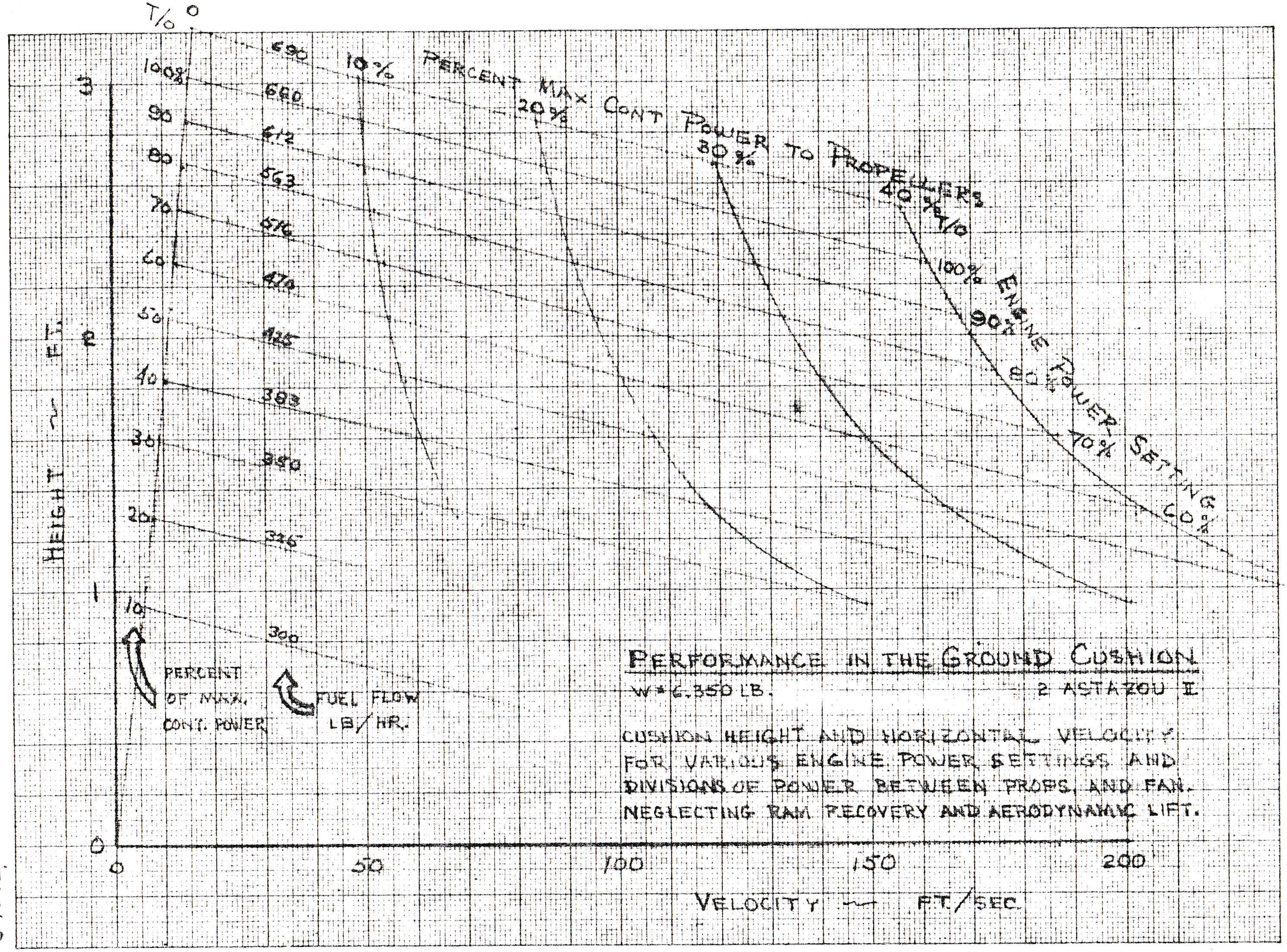


FIG 9

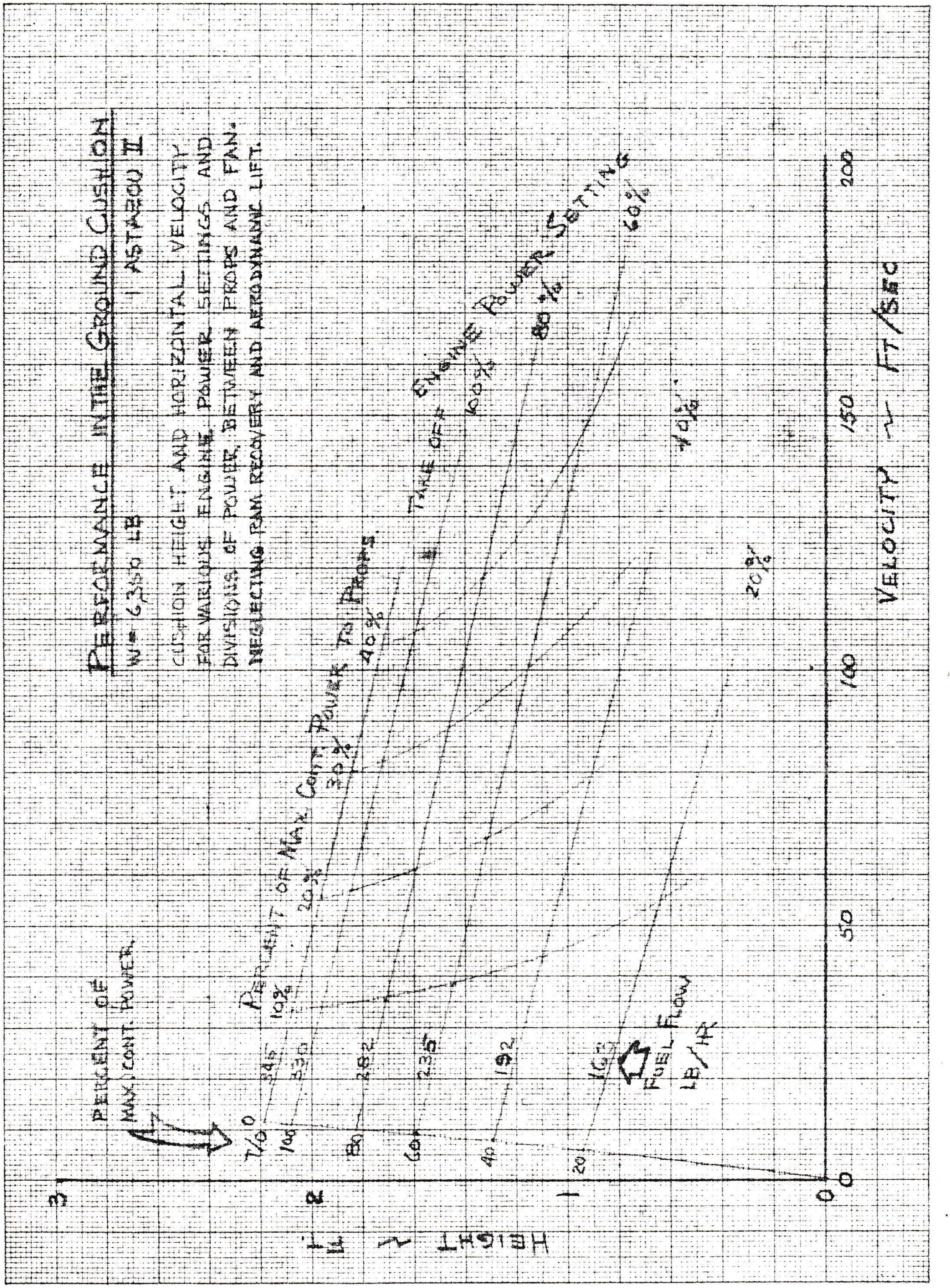
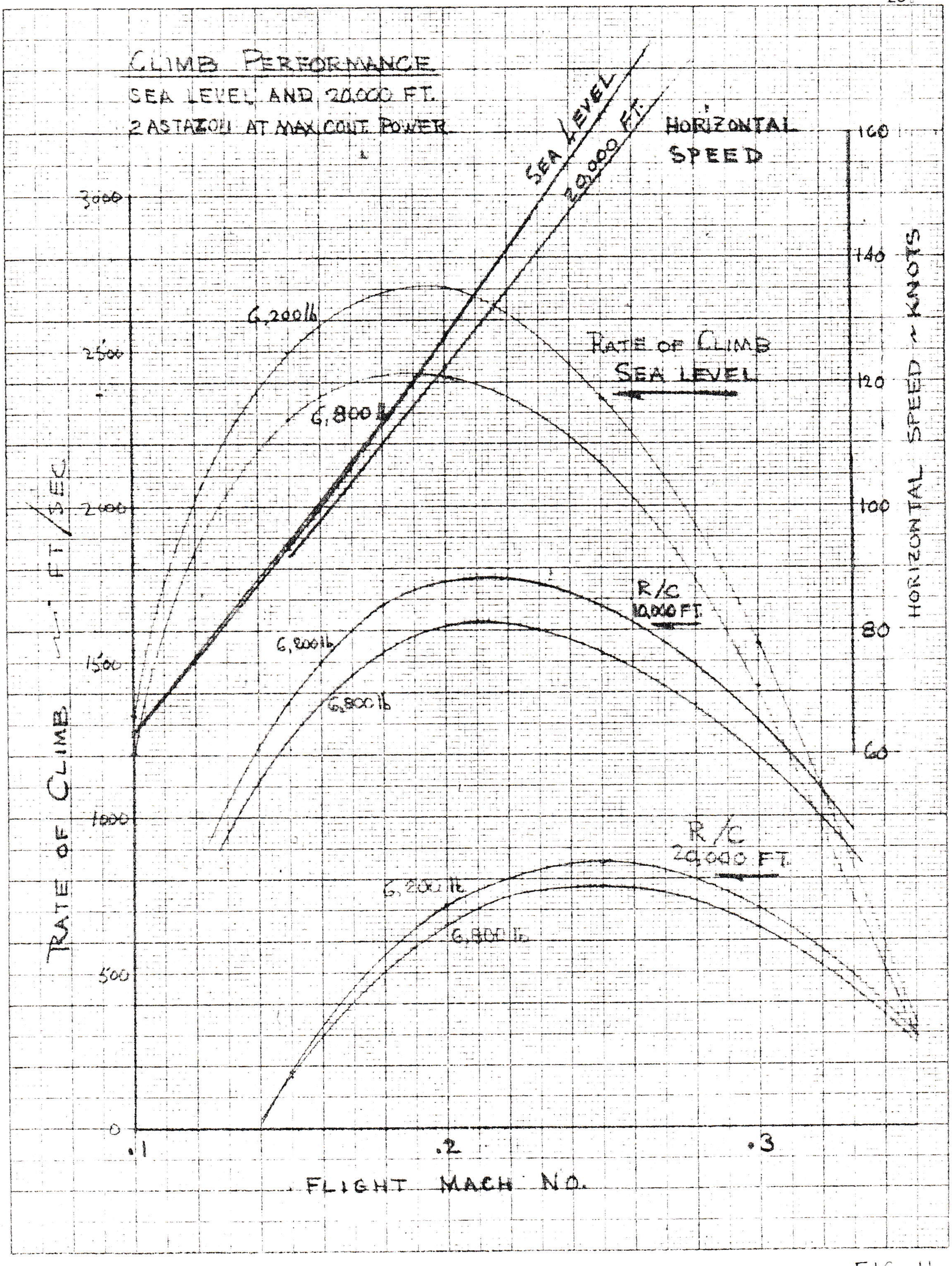

PERFORMANCE IN THE GROUND CUSHION
W = 6350 LB
ASTAZOU II
CUSHION HEIGHT AND HORIZONTAL VELOCITY FOR VARIOUS ENGINE POWER SETTINGS AND DIVISIONS OF POWER BETWEEN PROPS AND FAN.
NEGLECTING RAM RECOVERY AND AERODYNAMIC LIFT.
PERCENT OF MAX. CONT. POWER
T/O
100
80
60
40
20
345
330
282
235
192
165
FUEL FLOW LB/HR
PERCENT OF MAX CONT. POWER TO PROPS
0
10%
20%
30%
40%
TAKE OFF ENGINE POWER SETTING
100%
80%
60%
40%
20%
HEIGHT ~ FT
0
1
2
3
VELOCITY ~ FT/SEC
0
50
100
150
200

FIG 10



K·E 10 X 10 TO THE ½ INCH 359-12 KEUFFEL & ESSER CO.

FIG. 11

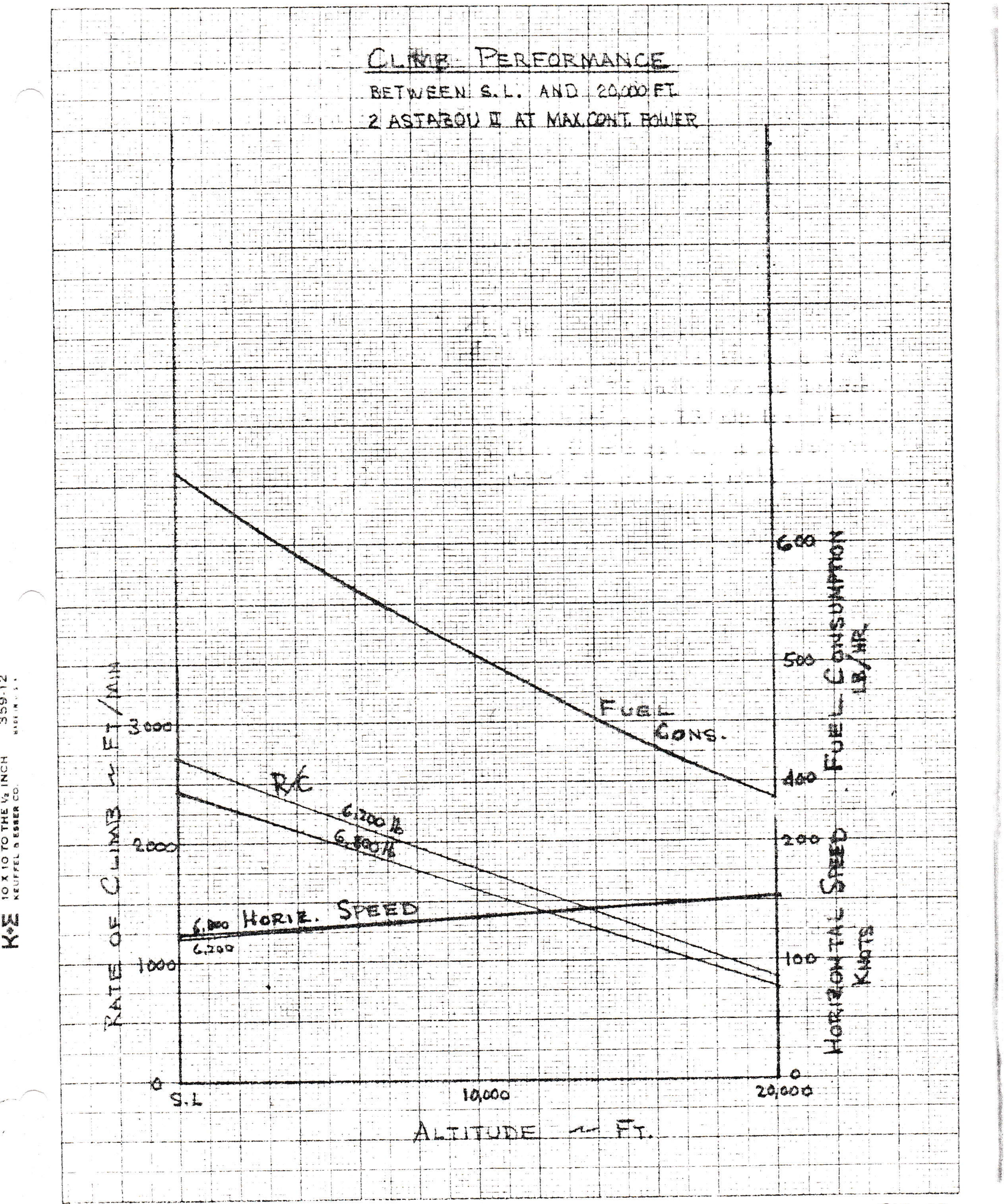

CLIMB PERFORMANCE
BETWEEN S.L. AND 20,000 FT.
2 ASTAZOU II AT MAX. CONT. POWER
FUEL CONS.
R/C
6,200 lb
6,800 lb
6,800 HORIZ. SPEED
6,200
RATE OF CLIMB ~ FT/MIN
3000
2000
1000
0
S.L
10,000
20,000
ALTITUDE ~ FT.
600
500
400
200
100
0
FUEL CONSUMPTION LB/HR.
HORIZONTAL SPEED KNOTS
K+E 10 X 10 TO THE ½ INCH 359-12
KEUFFEL & ESSER CO.

FIG 12

HN 524 2 ASTAZOU II

SPECIFIC AIR RANGE

SEA LEVEL

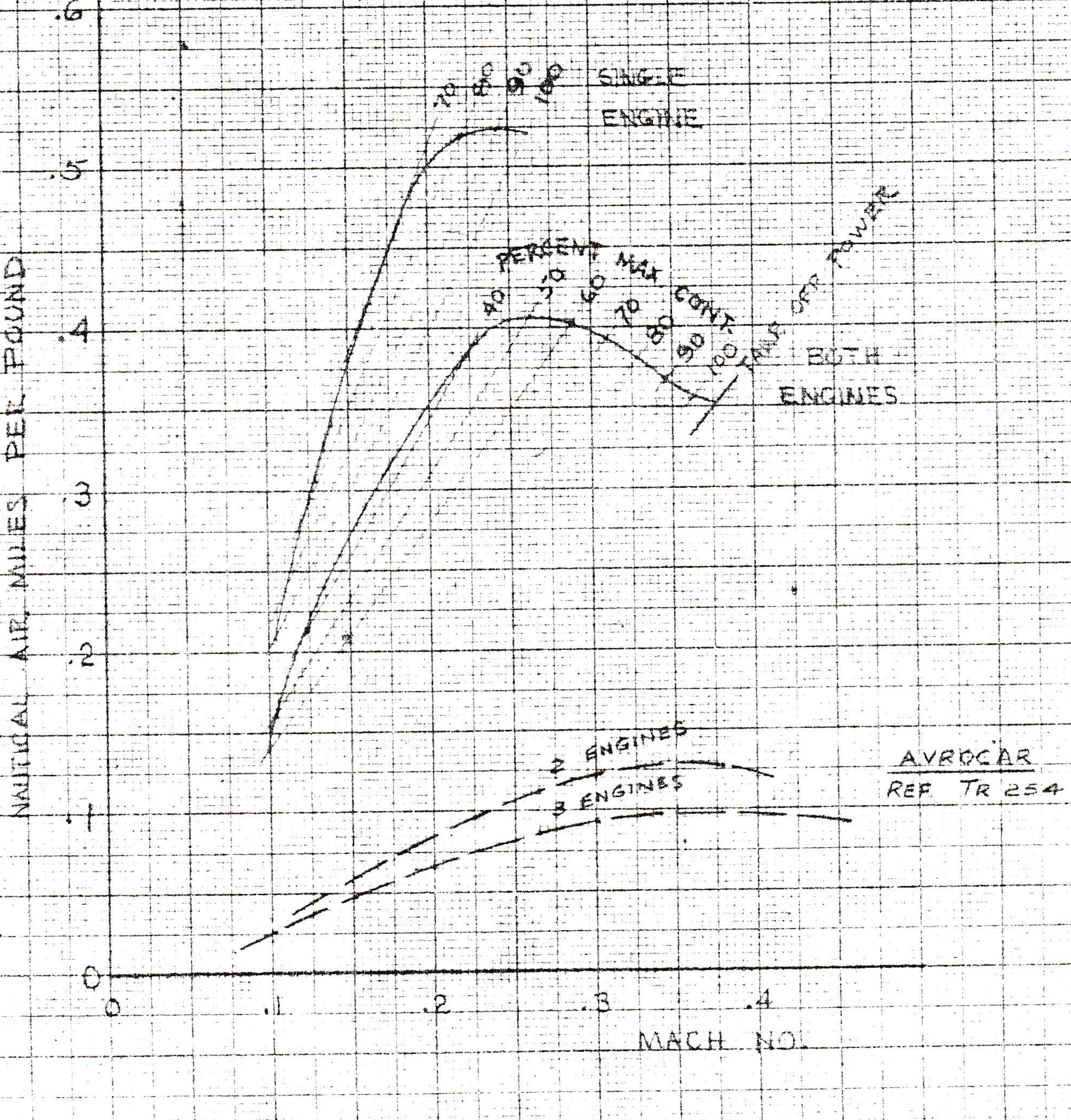
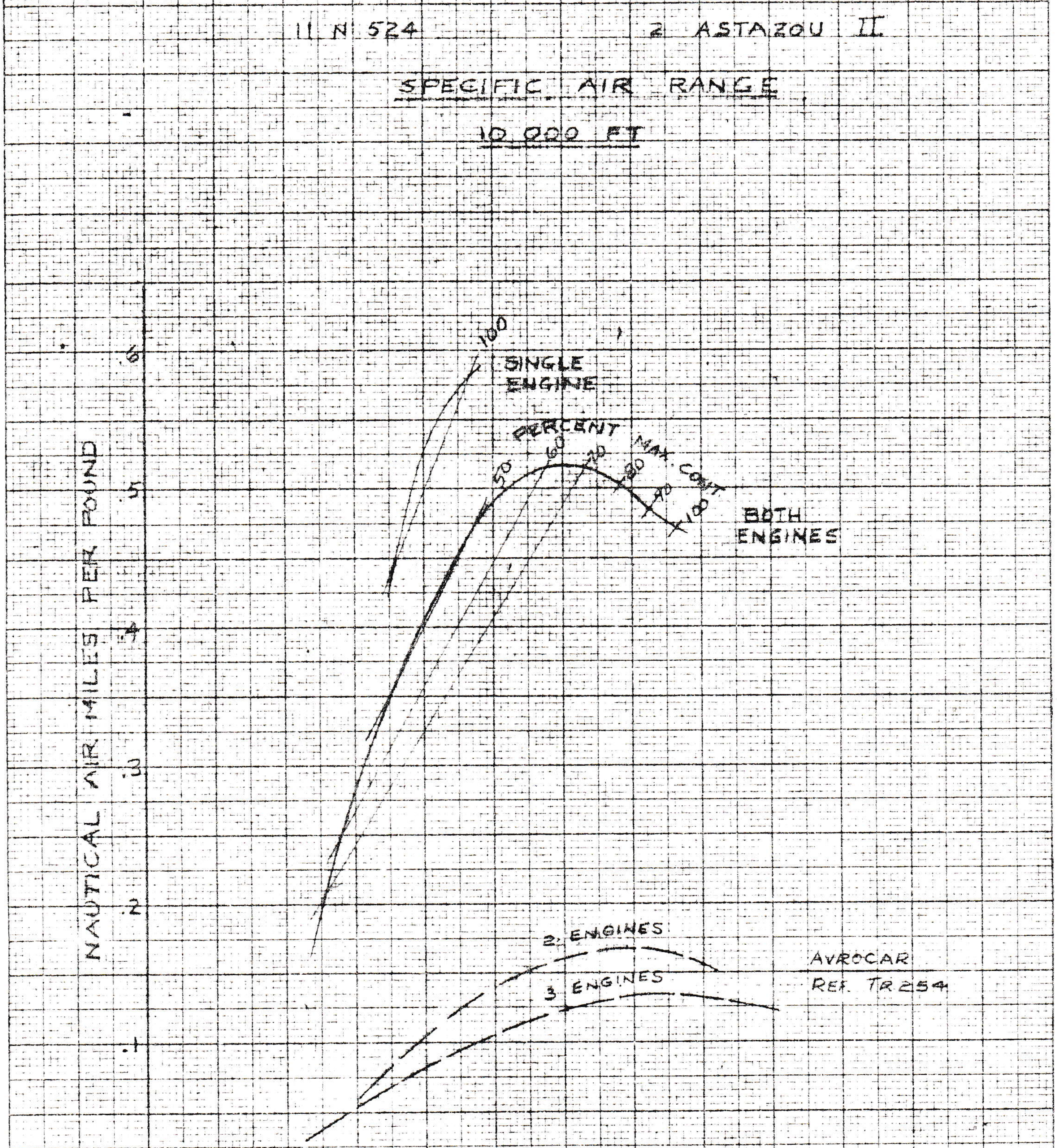


FIG 13



10 X 10 TO THE ½ INCH 359.12 KEUFFEL & ESSER CO.

FIG 14

II N 524 2 ASTAZOU II

SPECIFIC AIR RANGE

20,000 FT.

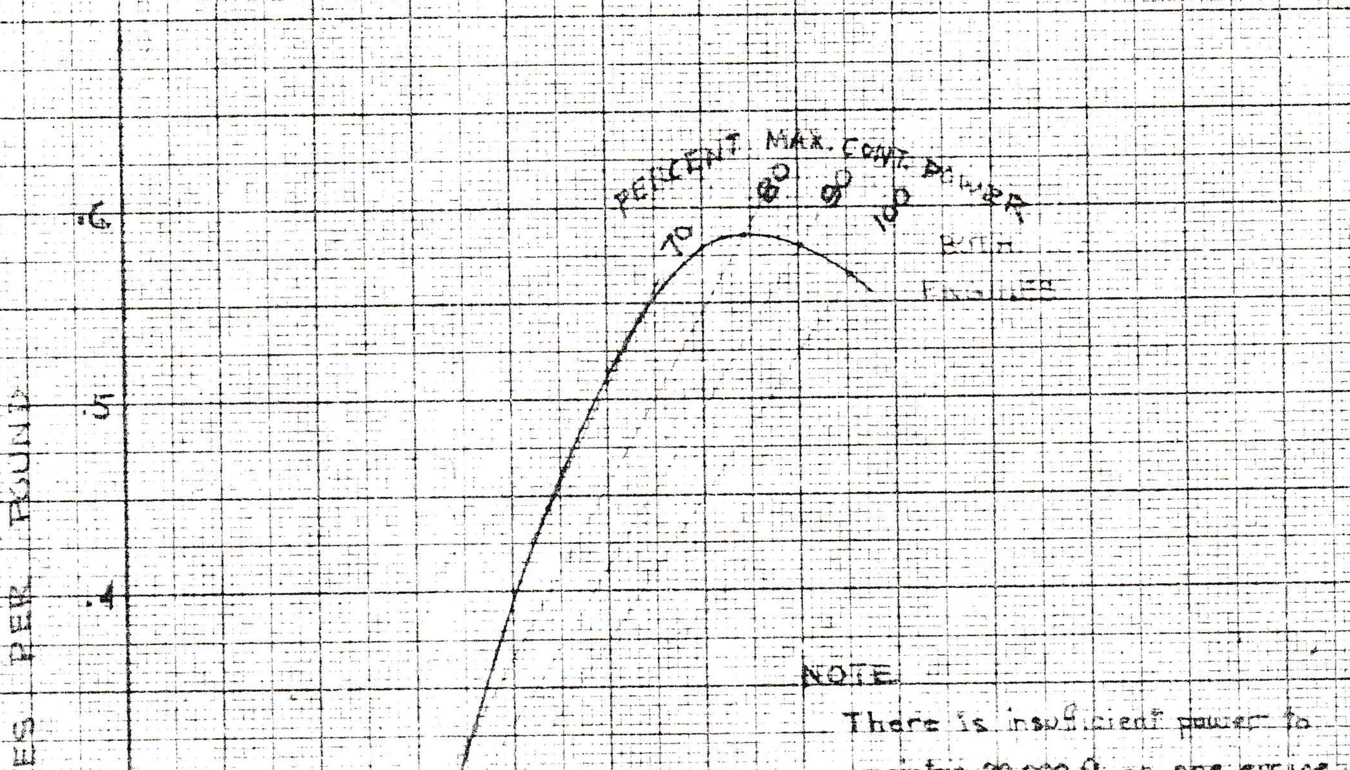


FIG 15

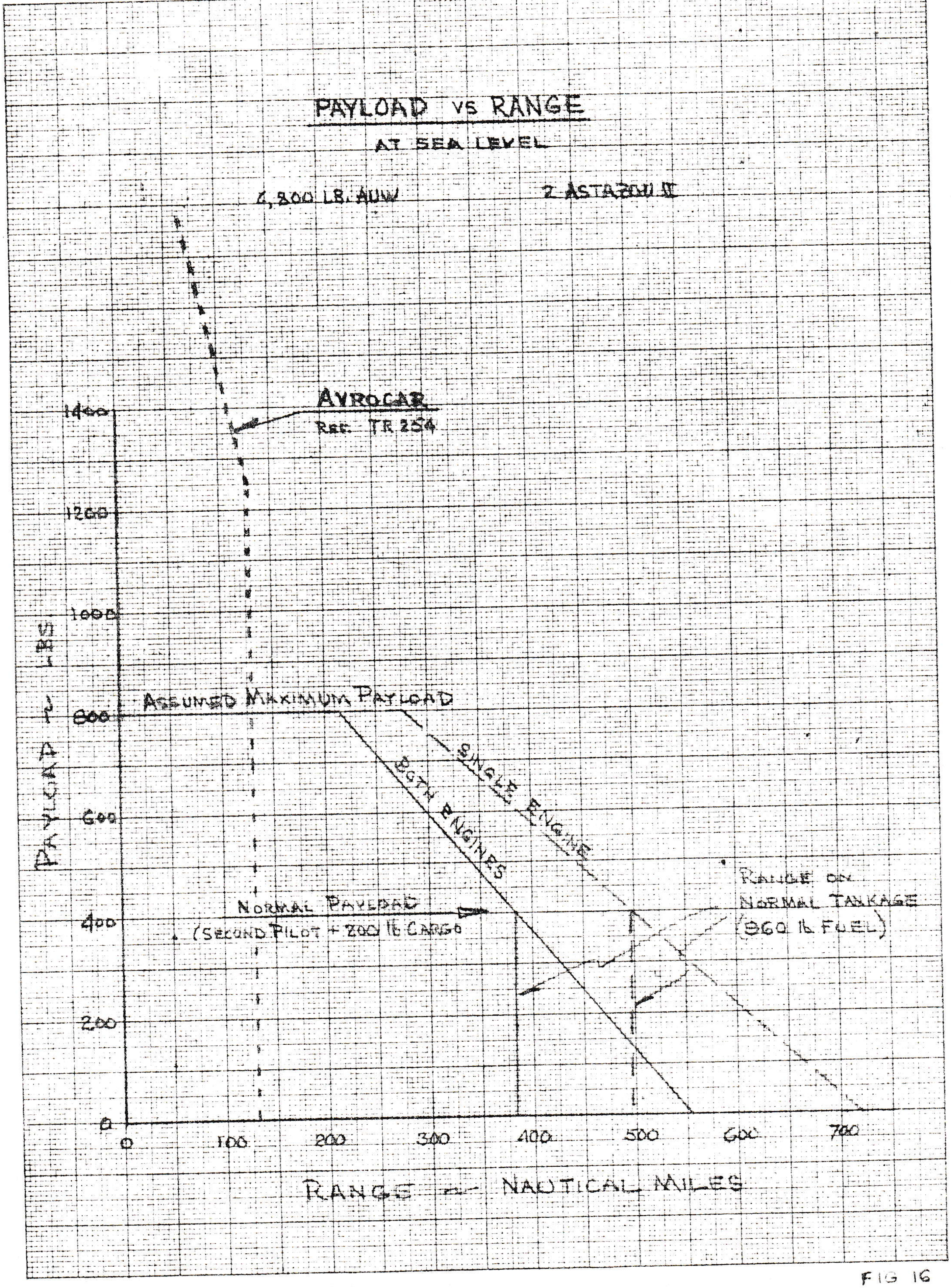


FIG 16

K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

PAYLOAD VS RANGE

AT ALTITUDE

6,000 LB AUW   2 ASTAZOU II

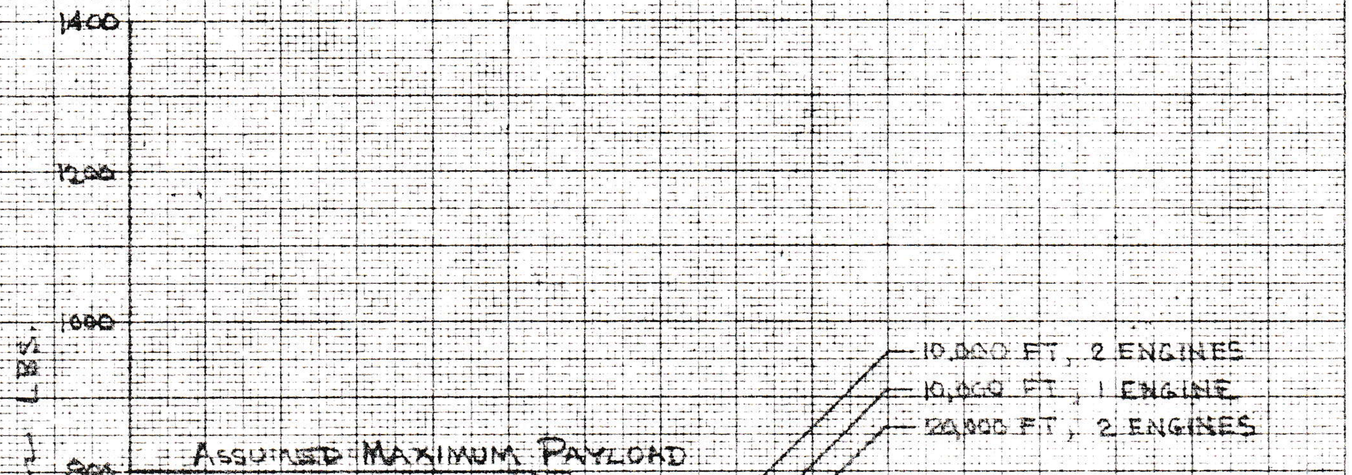


FIG 17

2 ASTAZOU

TYPICAL MISSION

USING 864 LB FUEL

N.B. TIMES SHOWN ARE FOR OUTWARD TRIP ONLY

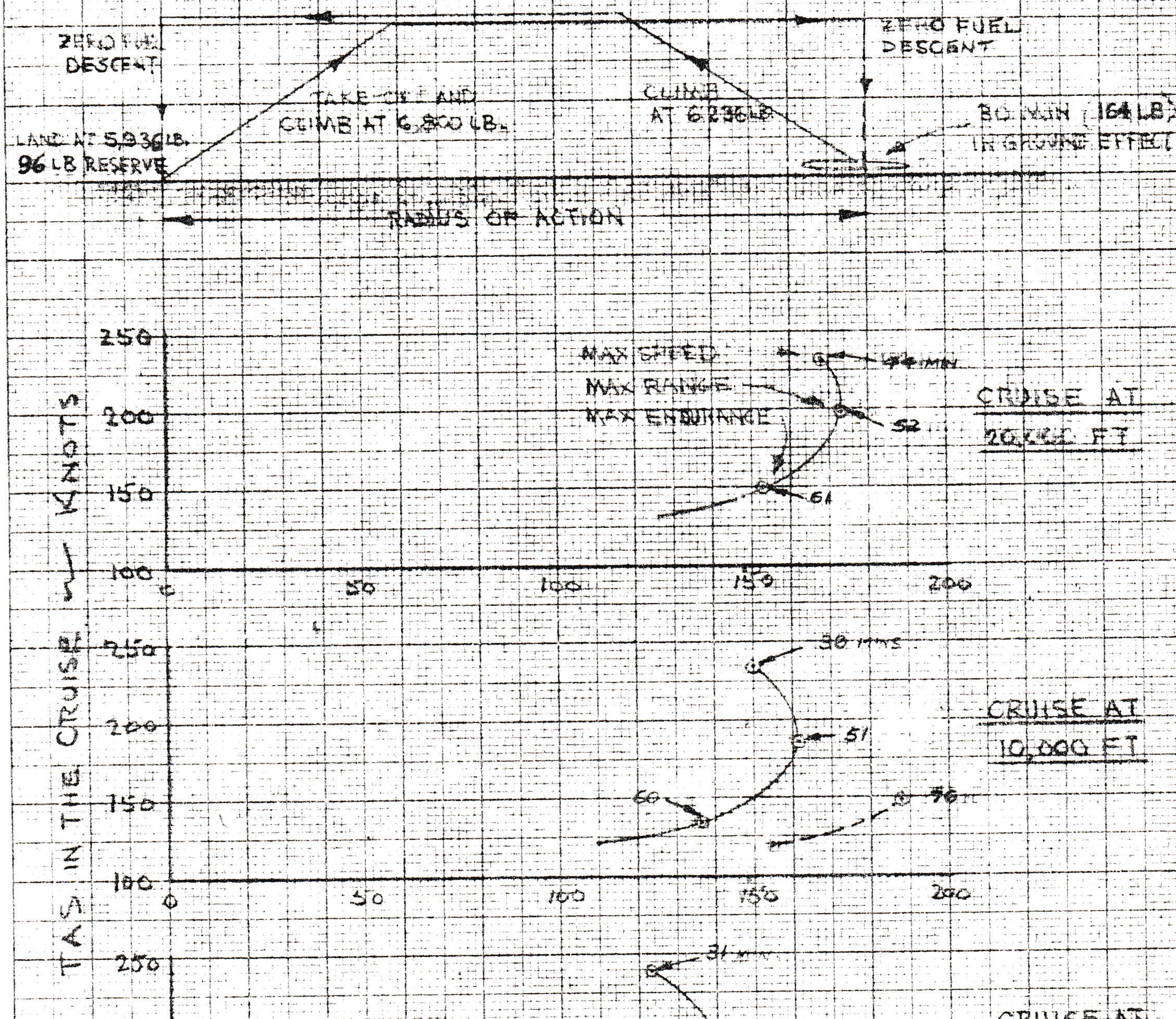
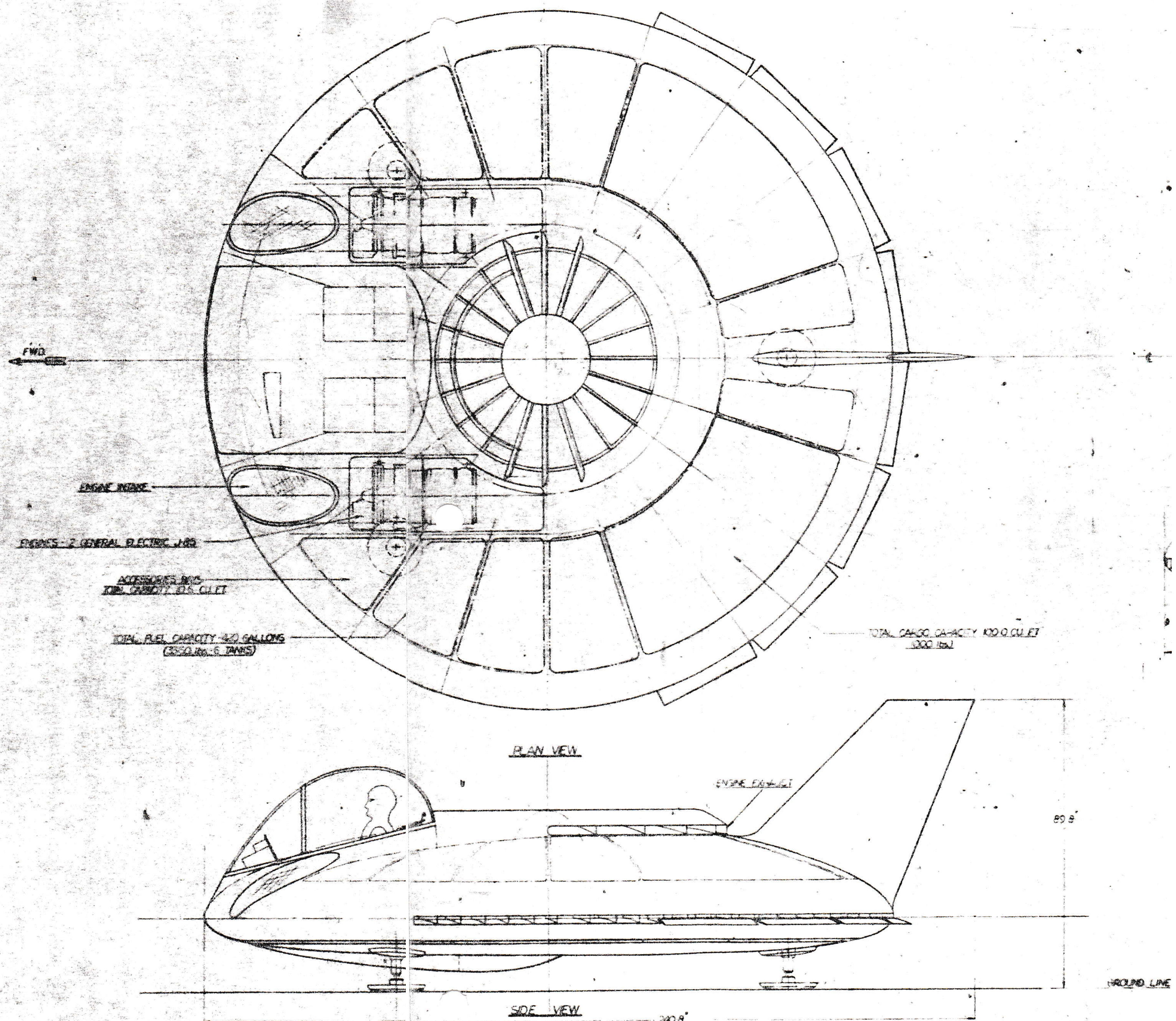


FIG 18


FWD.
ENGINE INTAKE
ENGINES - 2 GENERAL ELECTRIC J-85
ACCESSORIES BOX
TOTAL FUEL CAPACITY 420 GALLONS
(3550 lbs. 6 TANKS)
TOTAL CARGO CAPACITY 100.0 CU. FT.
PLAN VIEW
ENGINE EXHAUST
GROUND LINE
SIDE VIEW
240.8"

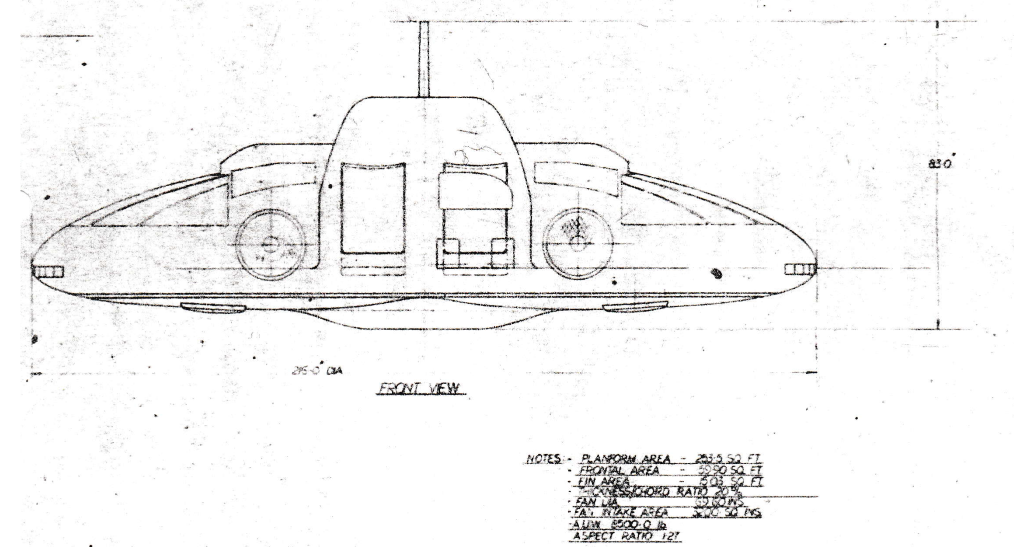
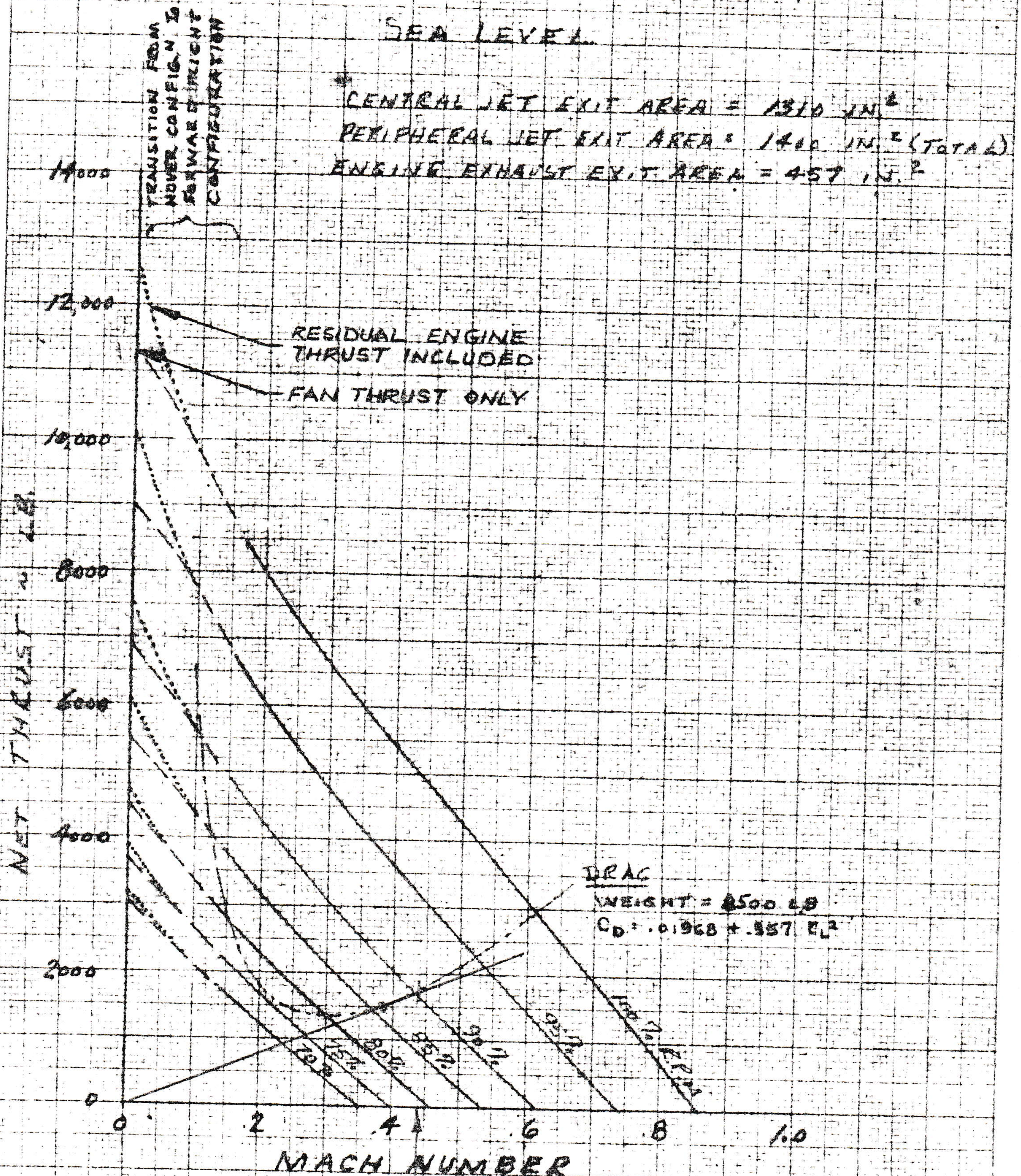

25'-0" DIA
FRONT VIEW
NOTES: - PLANFORM AREA
- FRONTAL AREA
- FIN AREA
- FAN INTAKE AREA
ASPECT RATIO

FIG.19

NET THRUST V/S MACH NUMBER



V_D = 220 kts (see wts. Rept)

FIG 20

TWO J-85'S

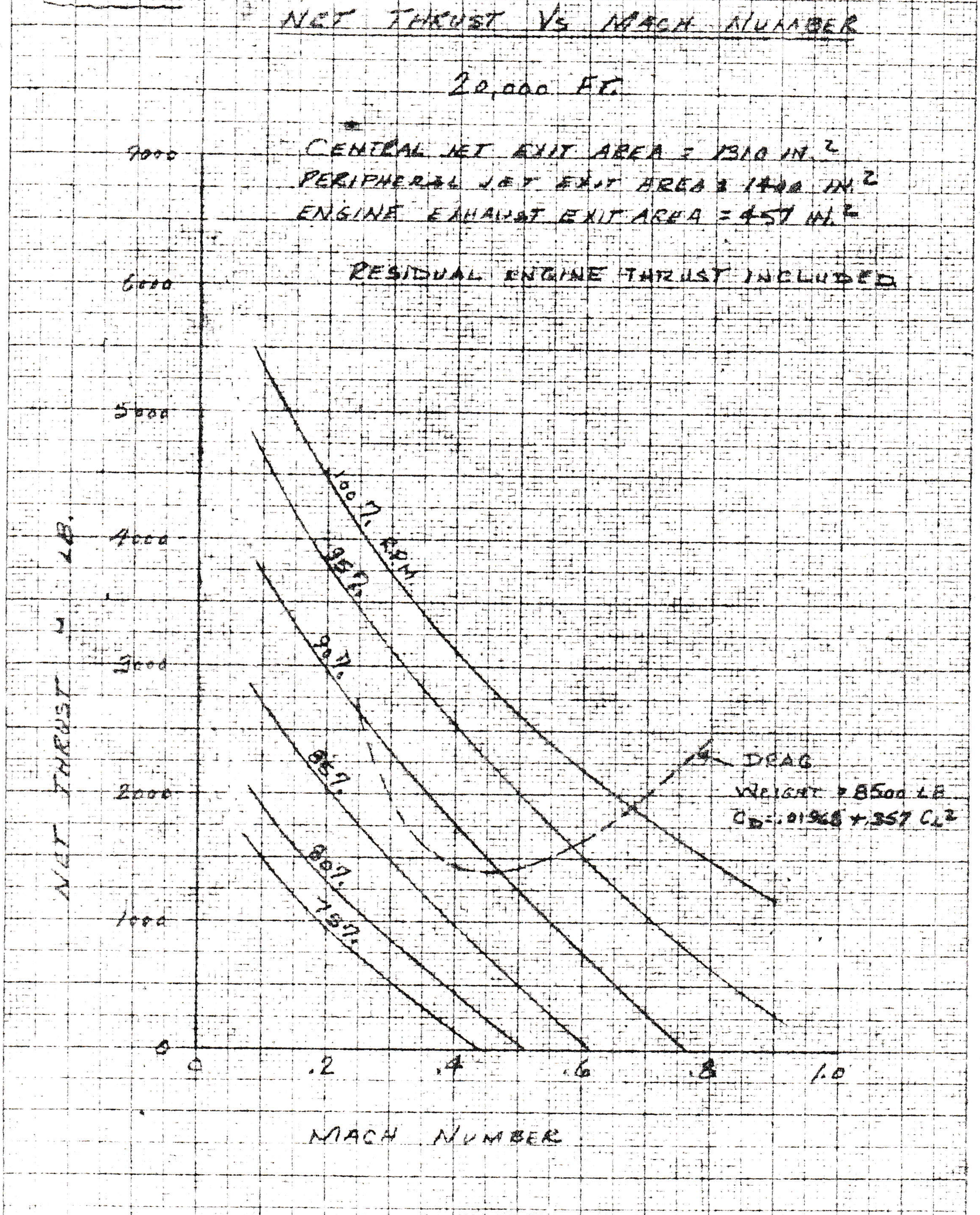


FIG 21

J-85

FUEL FLOW PER ENGINE VS MACH NUMBER

SEA LEVEL

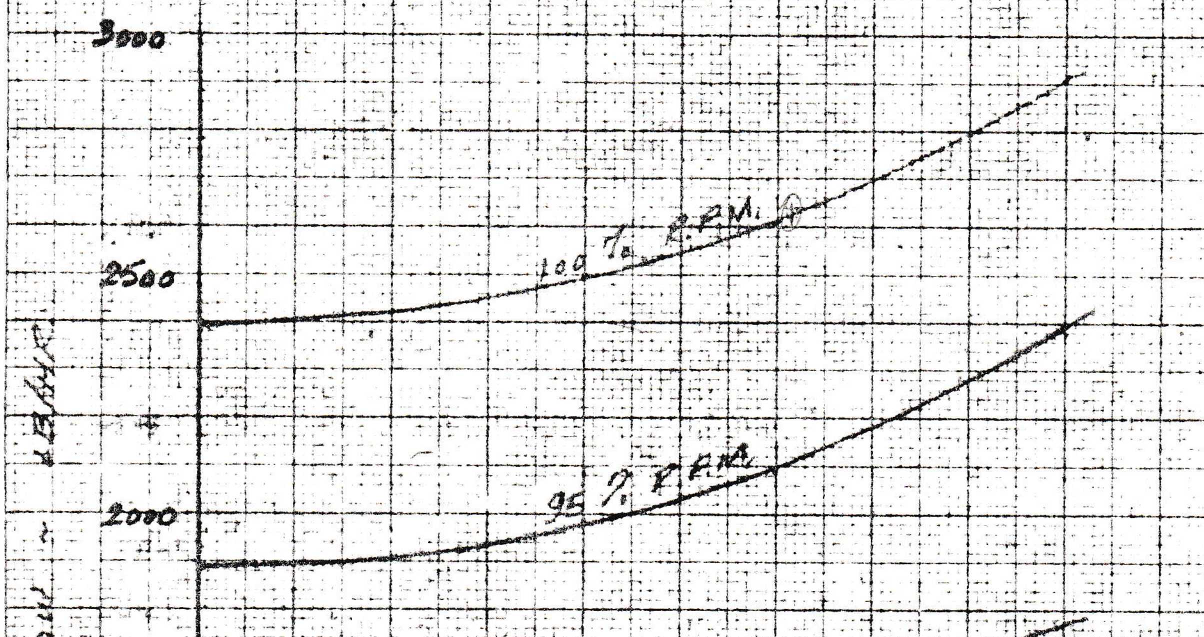
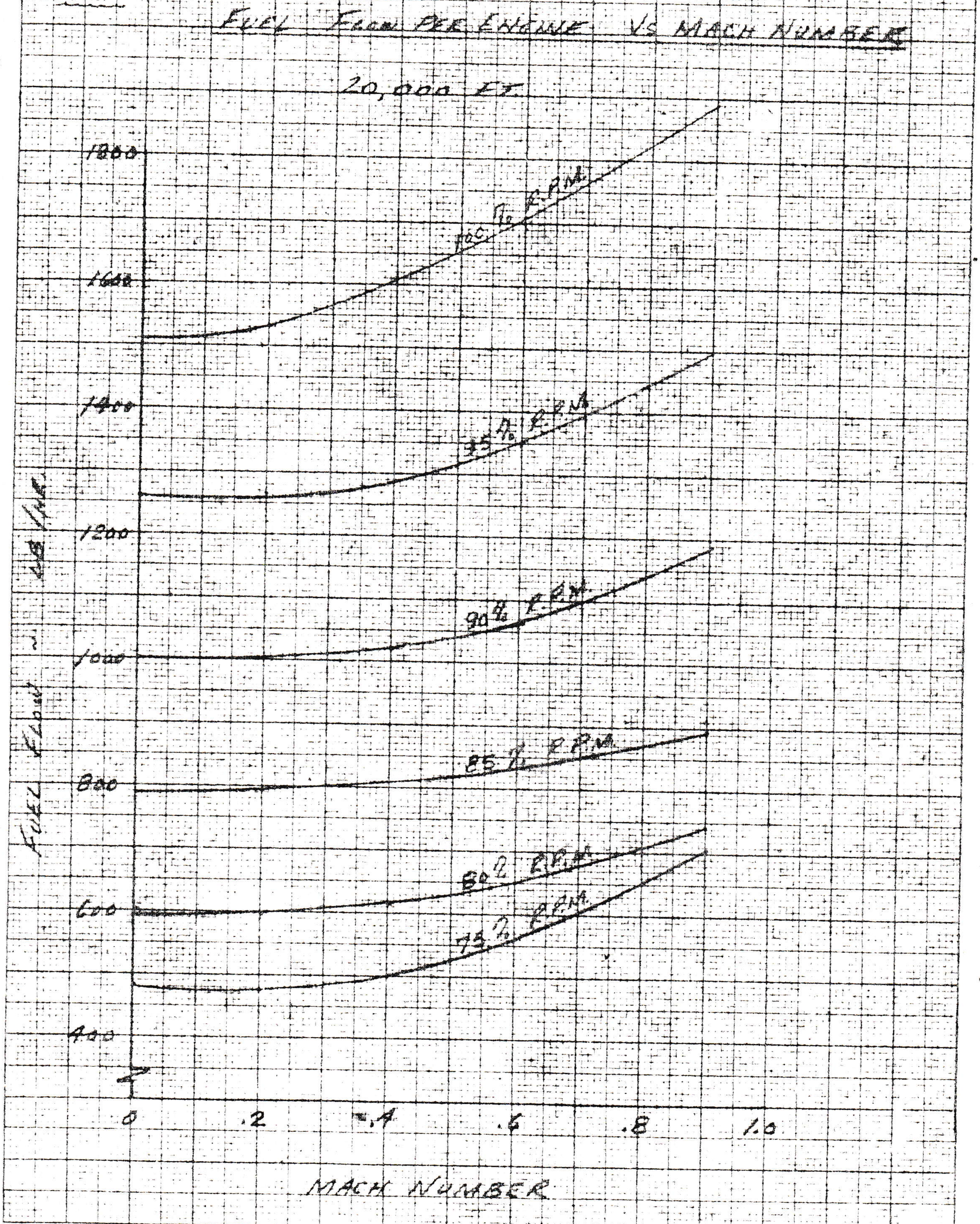


FIG 22

J-85



K•E 10 X 10 TO THE ½ INCH 359-12 KEUFFEL & ESSER CO. MADE IN U.S.A.

FIG 23

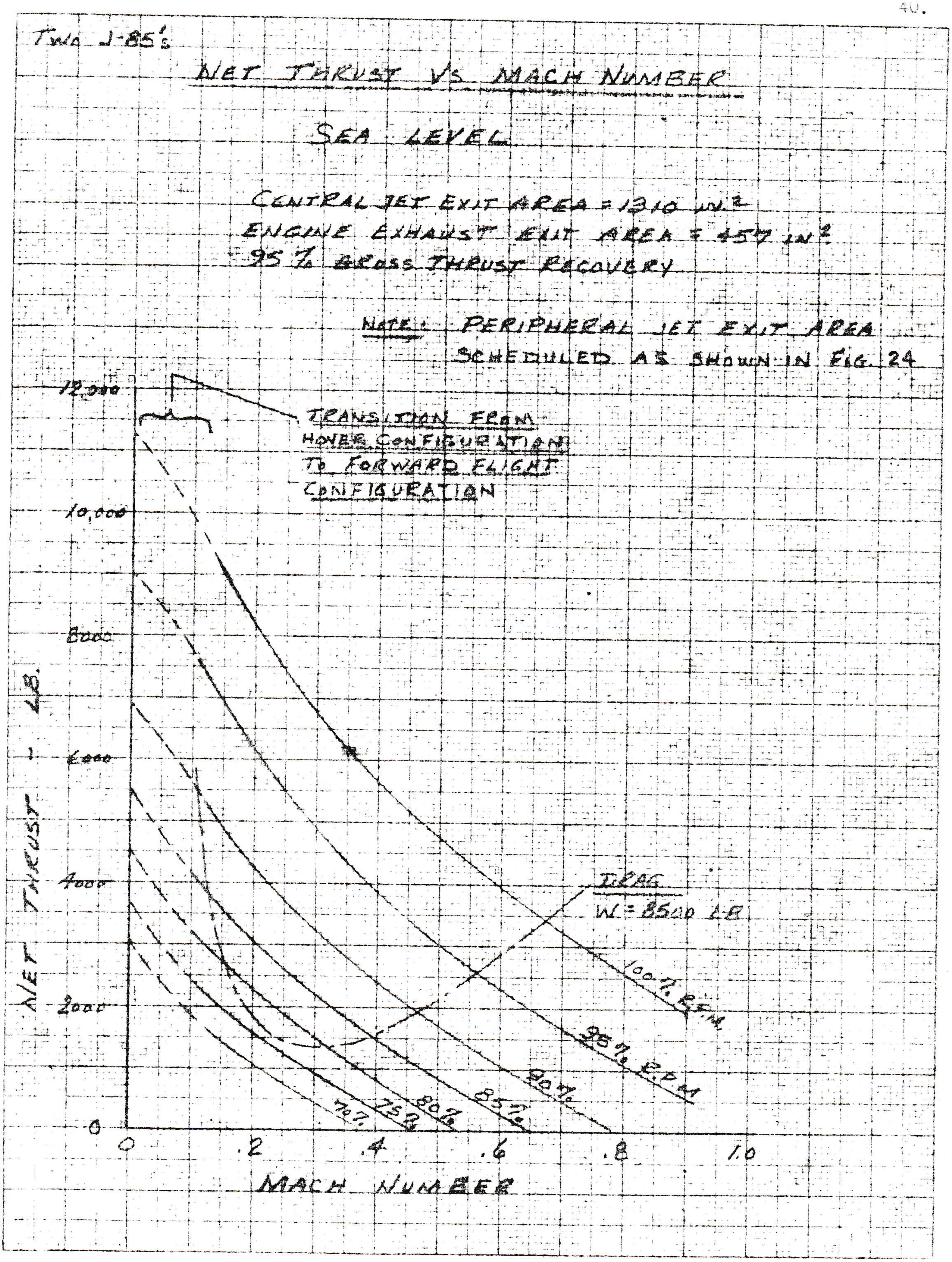
PERIPHERAL JET EXIT AREA

VARIATION WITH MACH NUMBER

CENTRAL JET EXIT AREA = 1810 IN.^{2}

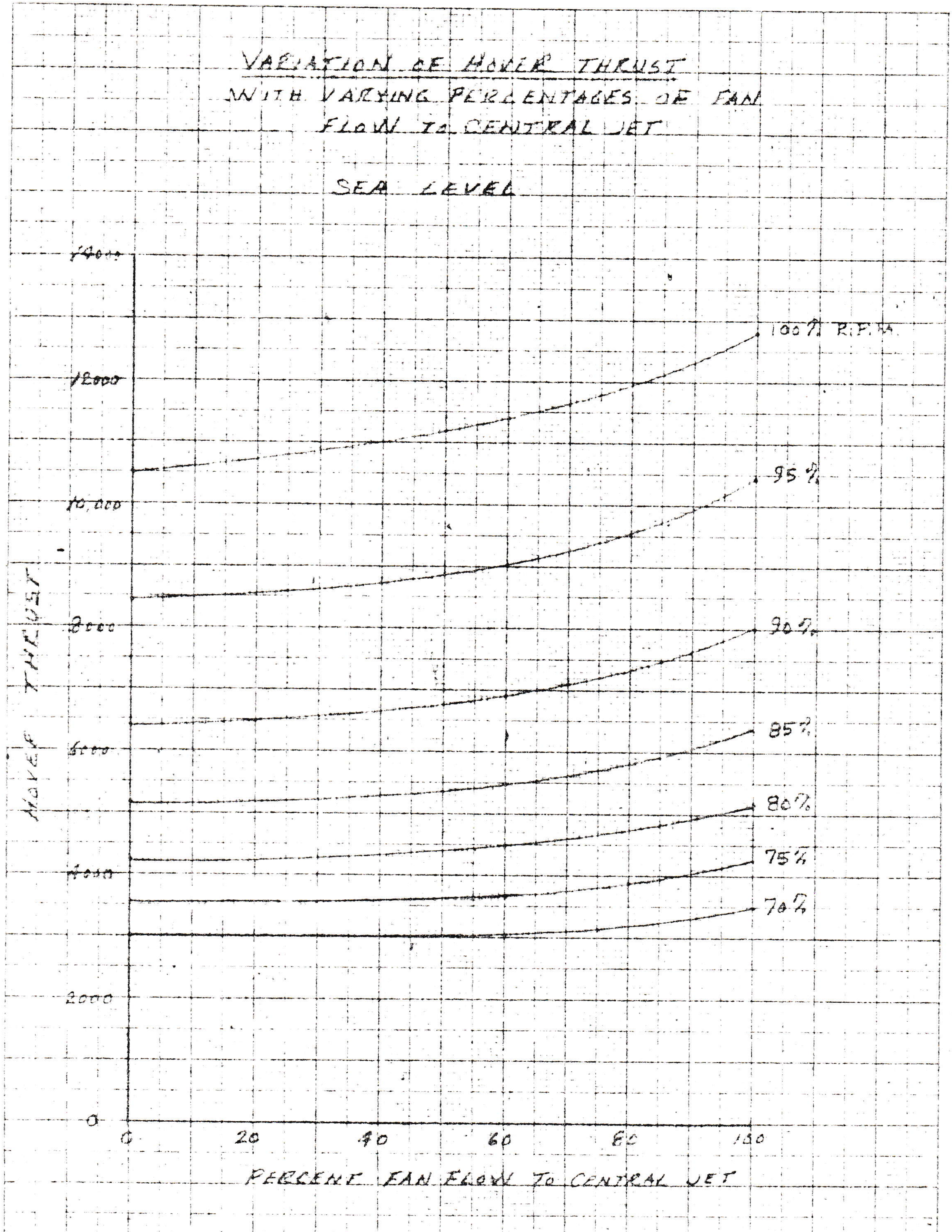
PERIPHERAL JET EXIT AREA VARIED TO GIVE A FAN PRESSURE = 1.16 AT SEA LEVEL WITH 1062 R.P.M.

FIG 24



TWO J-85's
NET THRUST VS MACH NUMBER
SEA LEVEL
CENTRAL JET EXIT AREA = 1310 IN²
ENGINE EXHAUST EXIT AREA = 457 IN²
95% GROSS THRUST RECOVERY
NOTE: PERIPHERAL JET EXIT AREA SCHEDULED AS SHOWN IN FIG. 24
TRANSITION FROM HOVER CONFIGURATION TO FORWARD FLIGHT CONFIGURATION
12,000
10,000
8000
6000
4000
2000
0
NET THRUST - LB.
DRAG W = 8500 LB
100% RPM
95% RPM
90%
85%
80%
75%
70%
0
.2
.4
.6
.8
1.0
MACH NUMBER

FIG 25



VARIATION OF HOVER THRUST
WITH VARYING PERCENTAGES OF FAN
FLOW TO CENTRAL JET
SEA LEVEL
HOVER THRUST
14000
12000
10,000
8000
6000
4000
2000
0
0
20
40
60
80
100
PERCENT FAN FLOW TO CENTRAL JET
100% R.P.M.
95%
90%
85%
80%
75%
70%

FIG 26

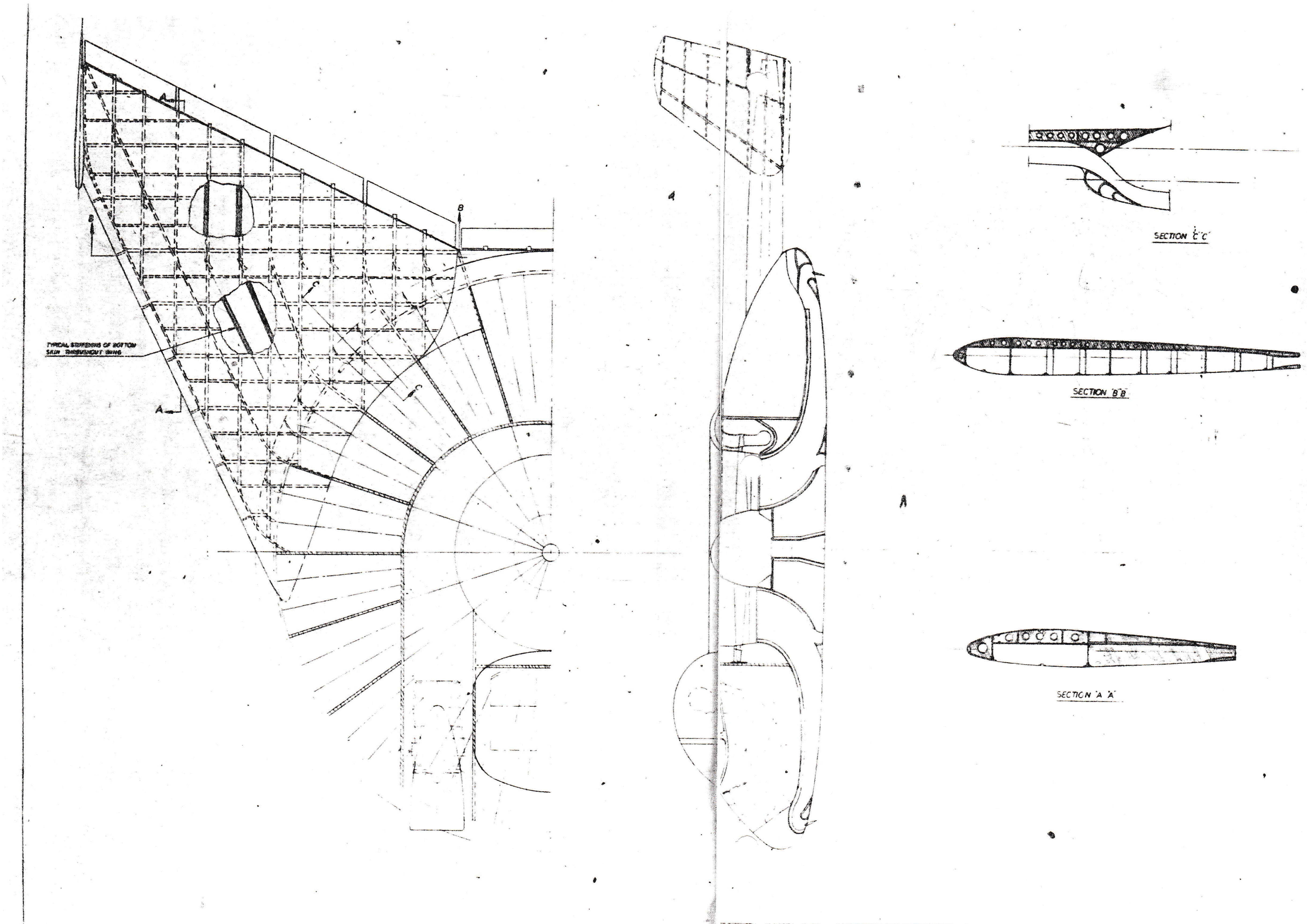

TYPICAL STIFFENING OF BOTTOM SKIN THROUGHOUT WING
SECTION 'C'C'
SECTION 'B'B'
SECTION 'A'A'

FIG.2

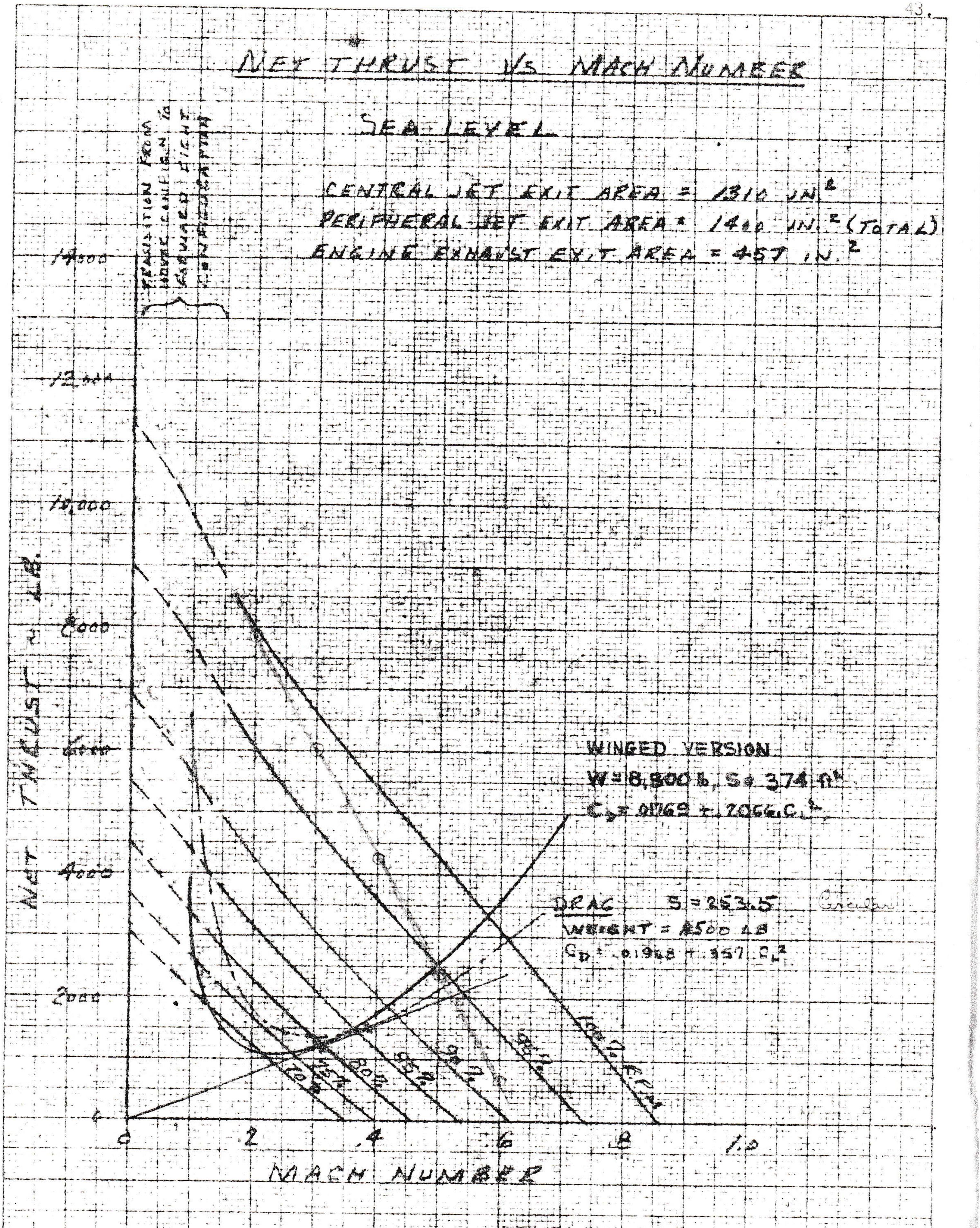

NET THRUST VS MACH NUMBER
SEA LEVEL
CENTRAL JET EXIT AREA = 1310 IN.²
PERIPHERAL JET EXIT AREA = 1400 IN.² (TOTAL)
ENGINE EXHAUST EXIT AREA = 457 IN.²
TRANSITION FROM HOVER CONFIGURATION TO FORWARD FLIGHT CONFIGURATION
14000
12000
10,000
8000
6000
4000
2000
0
NET THRUST ~ LB.
WINGED VERSION
W = 8,800 lb, S = 374 ft²
C_D = .0169 + .2066 C_L^2
DRAG S = 253.5 Circular
WEIGHT = 8500 LB
C_D = .01948 + .357 C_L^2
100% RPM
95%
90%
85%
80%
75%
70%
0
.2
.4
.6
.8
1.0
MACH NUMBER

FIG 28

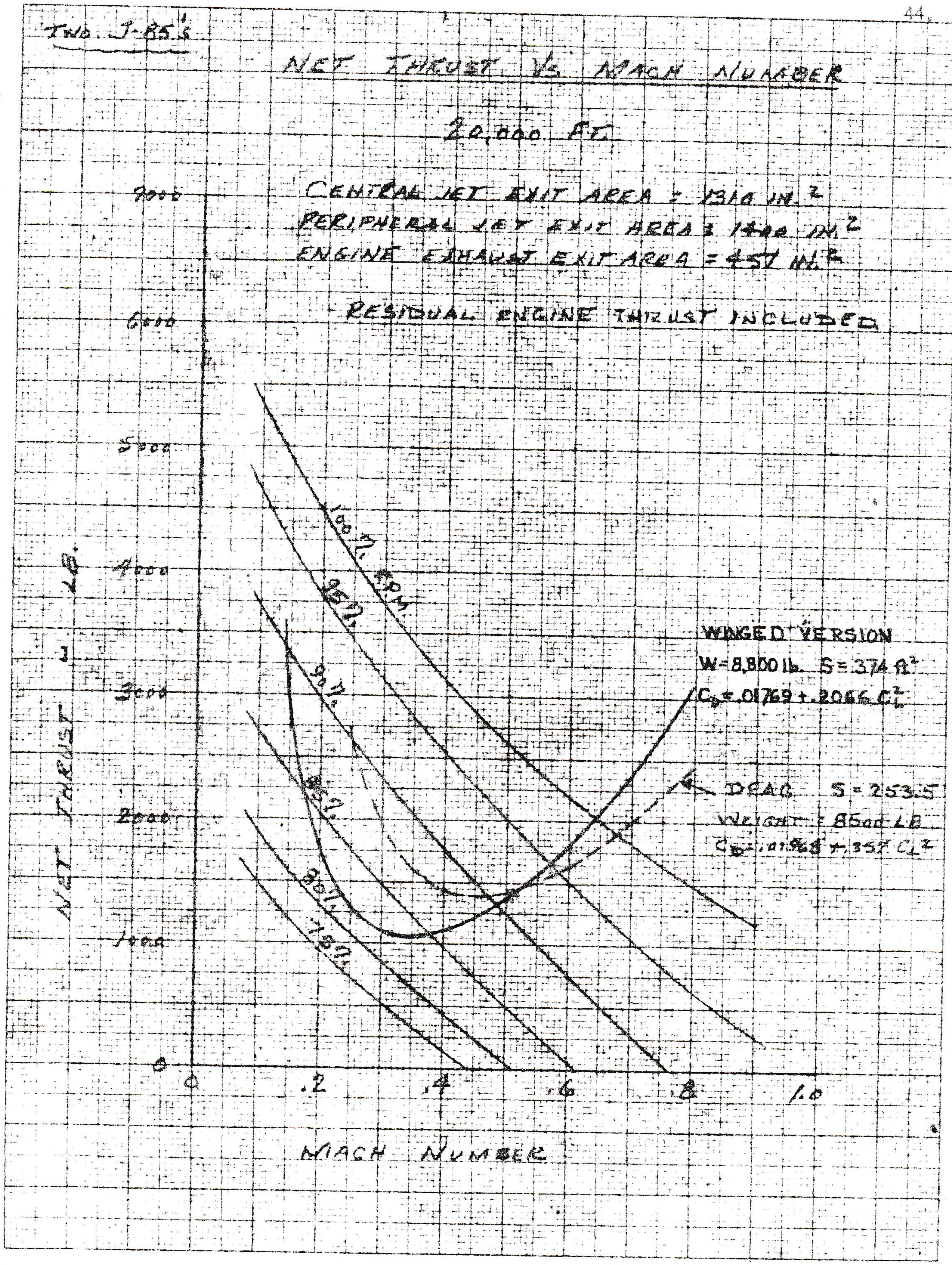

TWO J-85's
NET THRUST VS MACH NUMBER
20,000 FT.
CENTRAL JET EXIT AREA = 1310 IN.²
PERIPHERAL JET EXIT AREA = 1400 IN.²
ENGINE EXHAUST EXIT AREA = 457 IN.²
RESIDUAL ENGINE THRUST INCLUDED
NET THRUST LB.
MACH NUMBER
100 % RPM
95 %
90 %
85 %
80 %
78 %
WINGED VERSION
W = 8,800 lb. S = 374 ft²
C_D = .01769 + .2066 C_L^2
DRAG S = 253.5
WEIGHT = 8500 LB
C_D = .01568 + .357 C_L^2
0
1000
2000
3000
4000
5000
6000
9000
0
.2
.4
.6
.8
1.0
10 X 10 TO THE ½ INCH 358-12
KEUFFEL & ESSER CO.

FIG 29

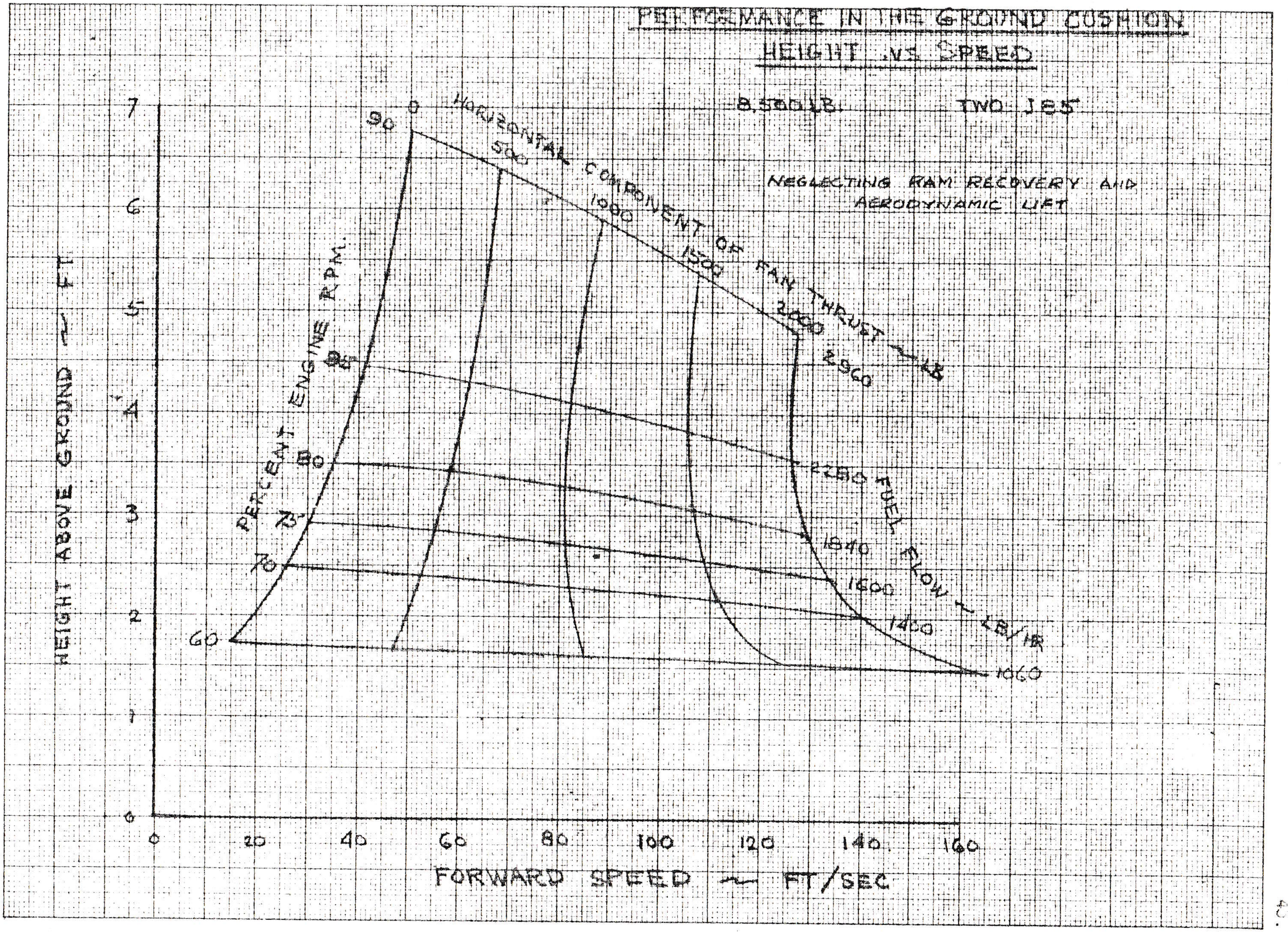

PERFORMANCE IN THE GROUND CUSHION
HEIGHT VS SPEED
8,500 LB.
TWO J85
NEGLECTING RAM RECOVERY AND
AERODYNAMIC LIFT
HORIZONTAL COMPONENT OF FAN THRUST ~ LB
0
500
1000
1500
2000
2960
FUEL FLOW ~ LB/HR
2280
1840
1600
1400
1060
PERCENT ENGINE RPM.
90
85
80
75
70
60
HEIGHT ABOVE GROUND ~ FT
0
1
2
3
4
5
6
7
FORWARD SPEED ~ FT/SEC
0
20
40
60
80
100
120
140
160

FIG 30

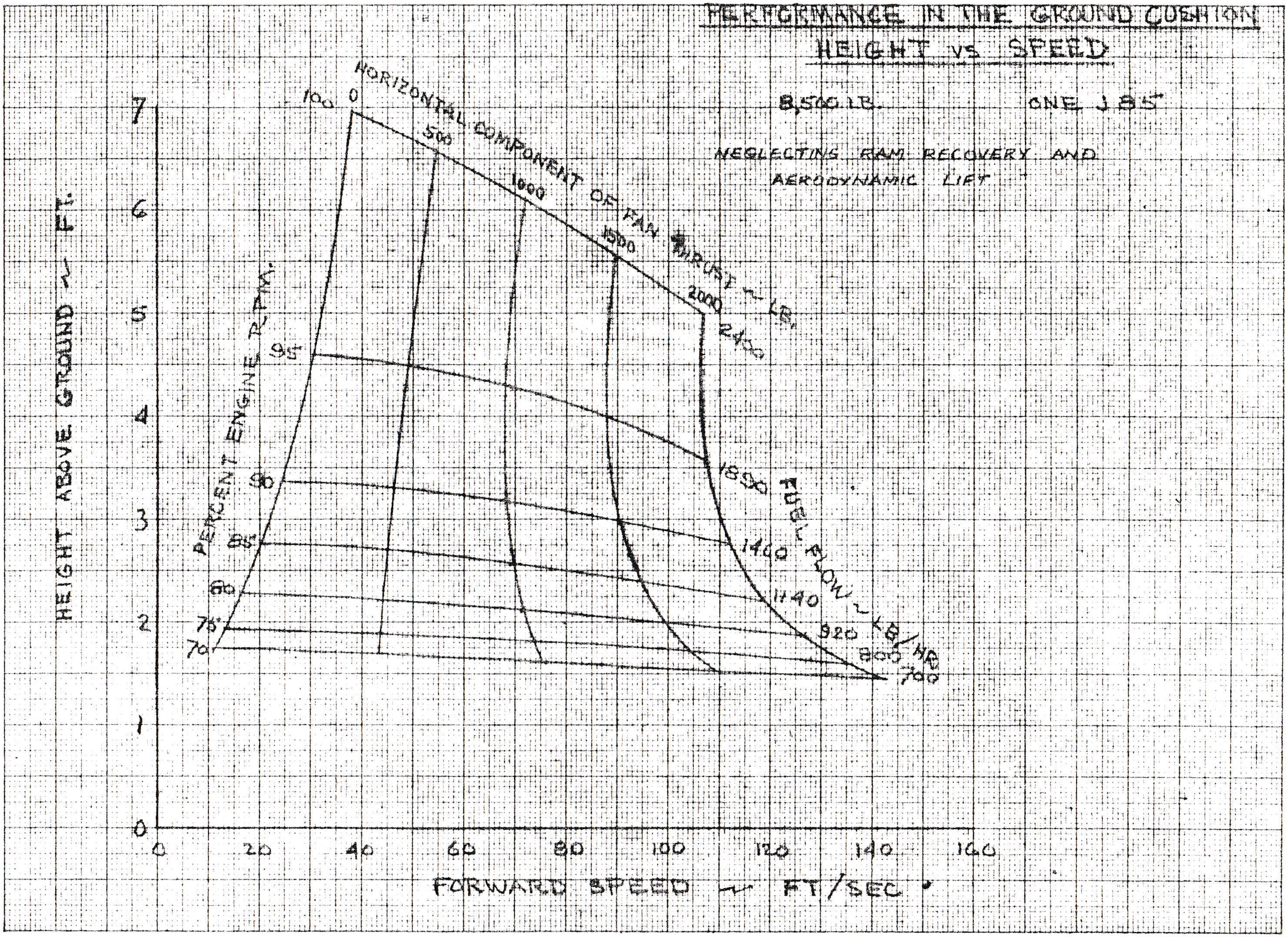

PERFORMANCE IN THE GROUND CUSHION
HEIGHT vs SPEED
8,500 LB.
ONE J 85
NEGLECTING RAM RECOVERY AND
AERODYNAMIC LIFT
HORIZONTAL COMPONENT OF FAN THRUST ~ LB.
0
500
1000
1500
2000
2400
PERCENT ENGINE R.P.M.
100
95
90
85
80
75
70
FUEL FLOW ~ LB/HR
1850
1460
1140
920
800
700
HEIGHT ABOVE GROUND ~ FT.
0
1
2
3
4
5
6
7
FORWARD SPEED ~ FT/SEC
0
20
40
60
80
100
120
140
160

FIG 31

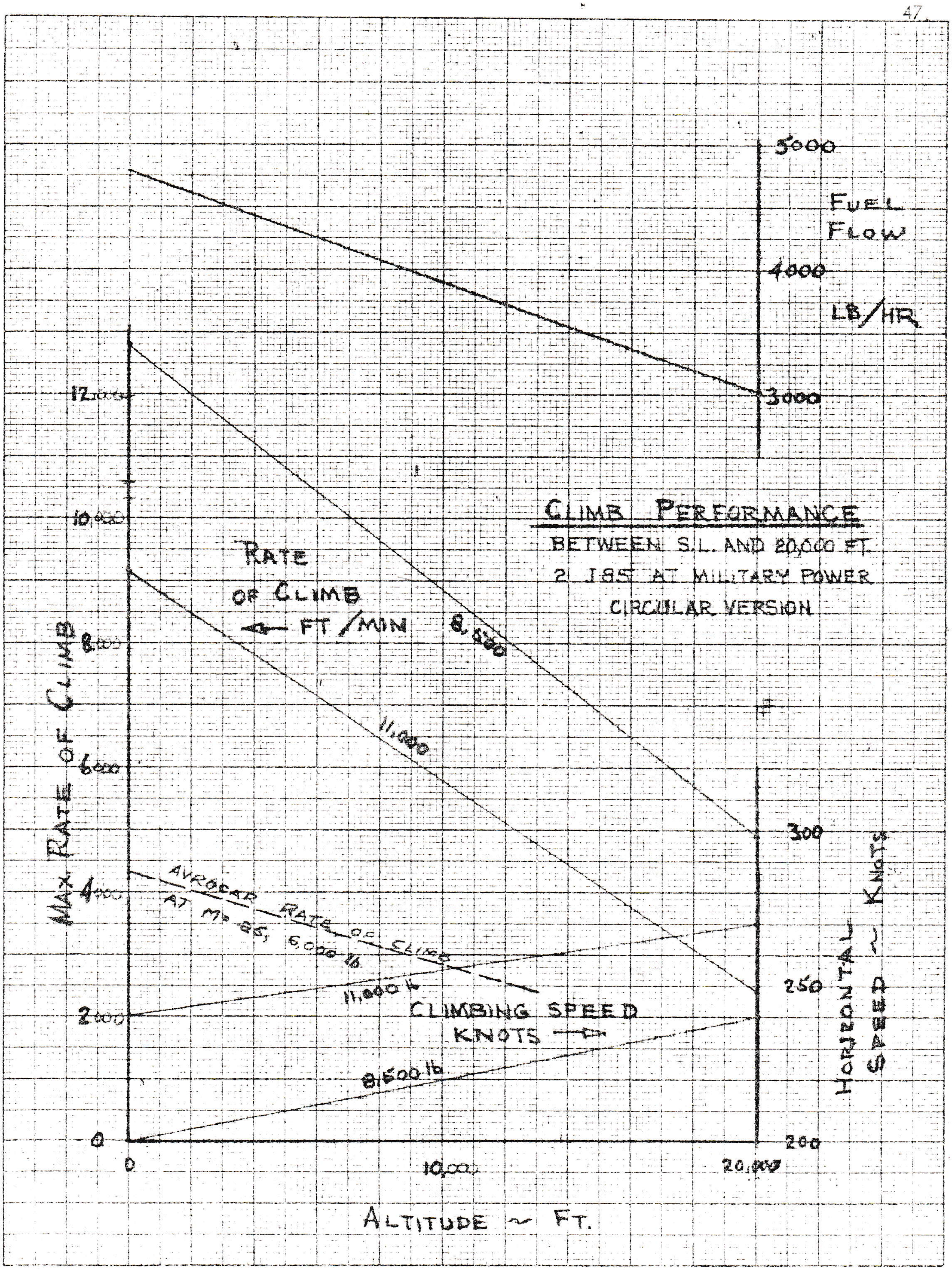

CLIMB PERFORMANCE
BETWEEN S.L. AND 20,000 FT.
2 J85 AT MILITARY POWER
CIRCULAR VERSION
FUEL FLOW LB/HR
5000
4000
3000
RATE OF CLIMB ← FT/MIN
8,500
11,000
AVROCAR RATE OF CLIMB AT M=.85, 6,000 lb
11,000 lb
8,500 lb
CLIMBING SPEED KNOTS →
MAX RATE OF CLIMB
12,000
10,000
8,000
6,000
4,000
2,000
0
HORIZONTAL SPEED ~ KNOTS
300
250
200
0
10,000
20,000
ALTITUDE ~ FT.

FIG 32

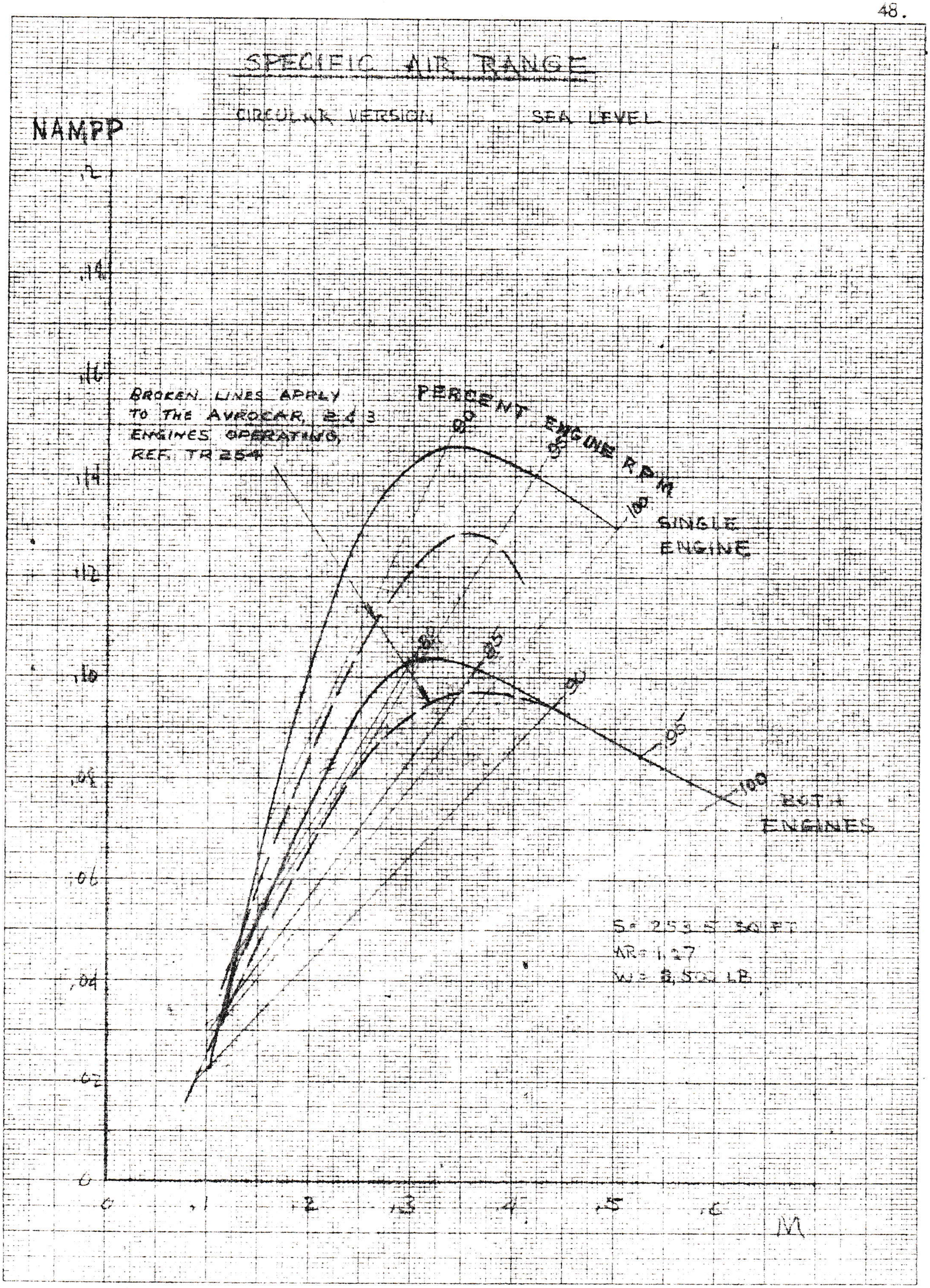
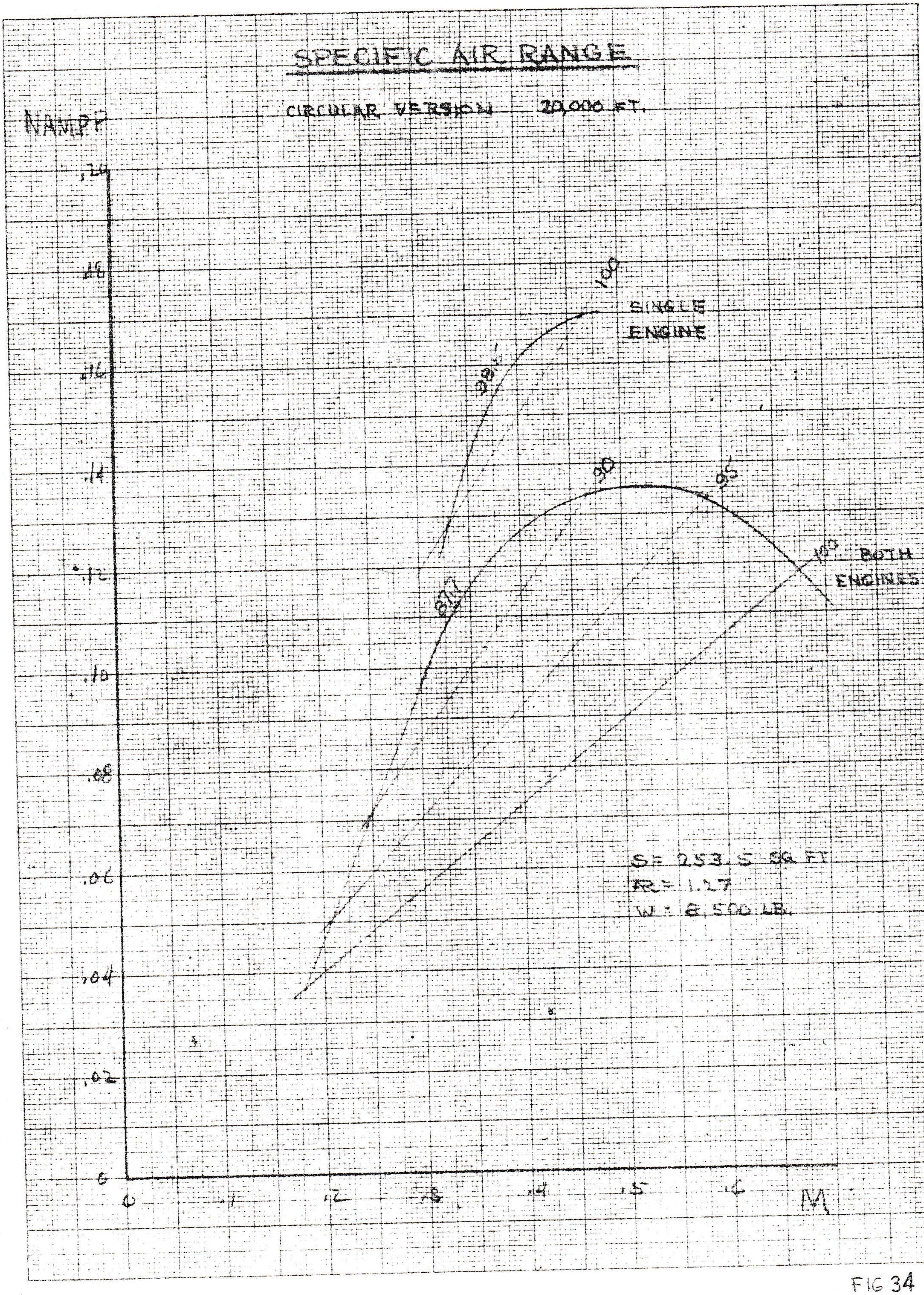

SPECIFIC AIR RANGE
CIRCULAR VERSION
SEA LEVEL
NAMPP
.12
.14
.16
.14
.12
.10
.08
.06
.04
.02
0
BROKEN LINES APPLY TO THE AVROCAR, 2 & 3 ENGINES OPERATING, REF. TR 254
PERCENT ENGINE RPM
90
95
100
SINGLE ENGINE
80
85
90
95
100
BOTH ENGINES
S = 253.5 SQ FT
AR = 1.27
W = 5,500 LB
0
.1
.2
.3
.4
.5
.6
M
10 X 10 TO THE CM 359-14
KEUFFEL & ESSER CO.

FIG 33



10 X 10 TO THE C.M. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

FIG 34

SPECIFIC AIR RANGE

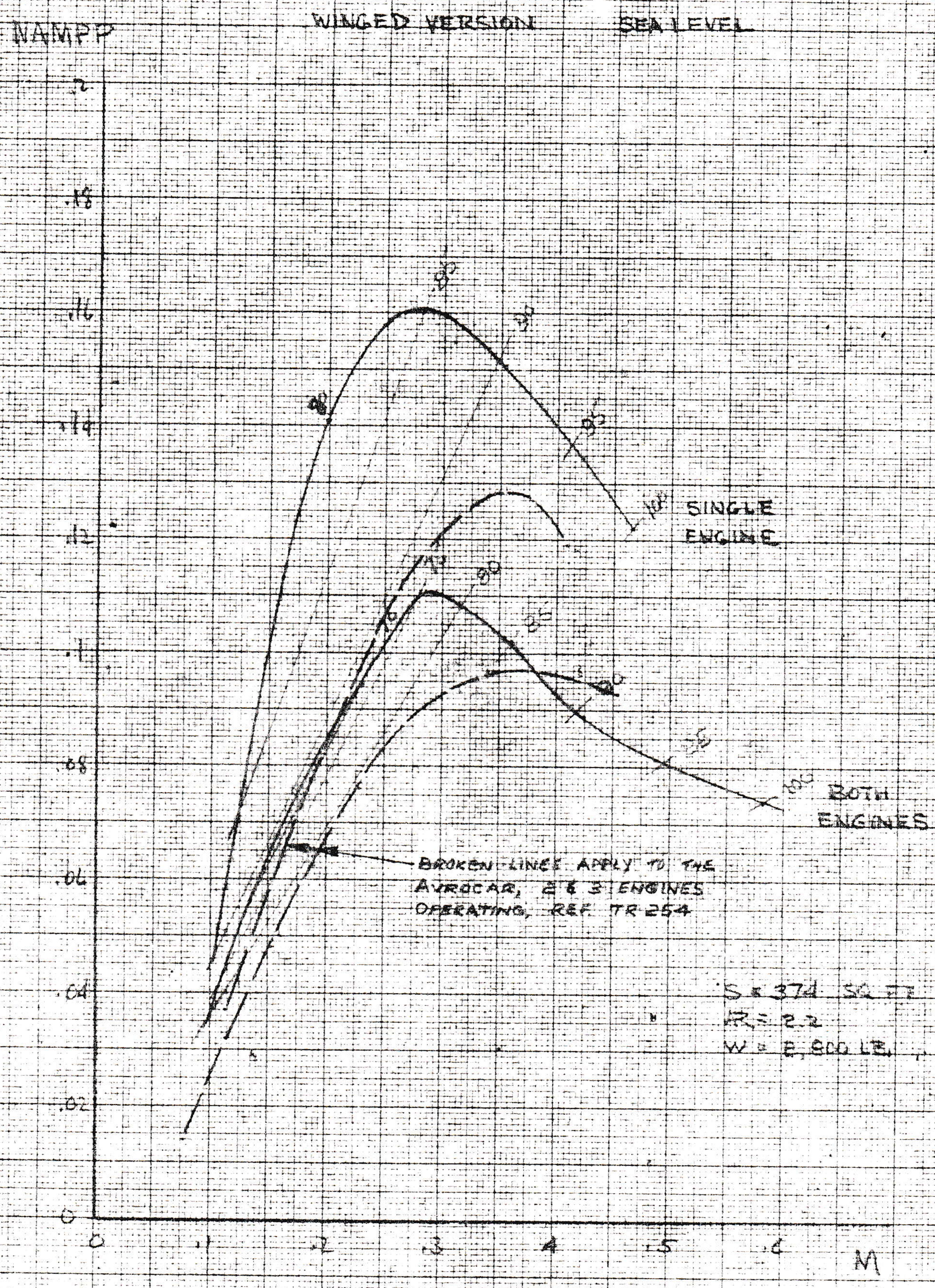


FIG 35

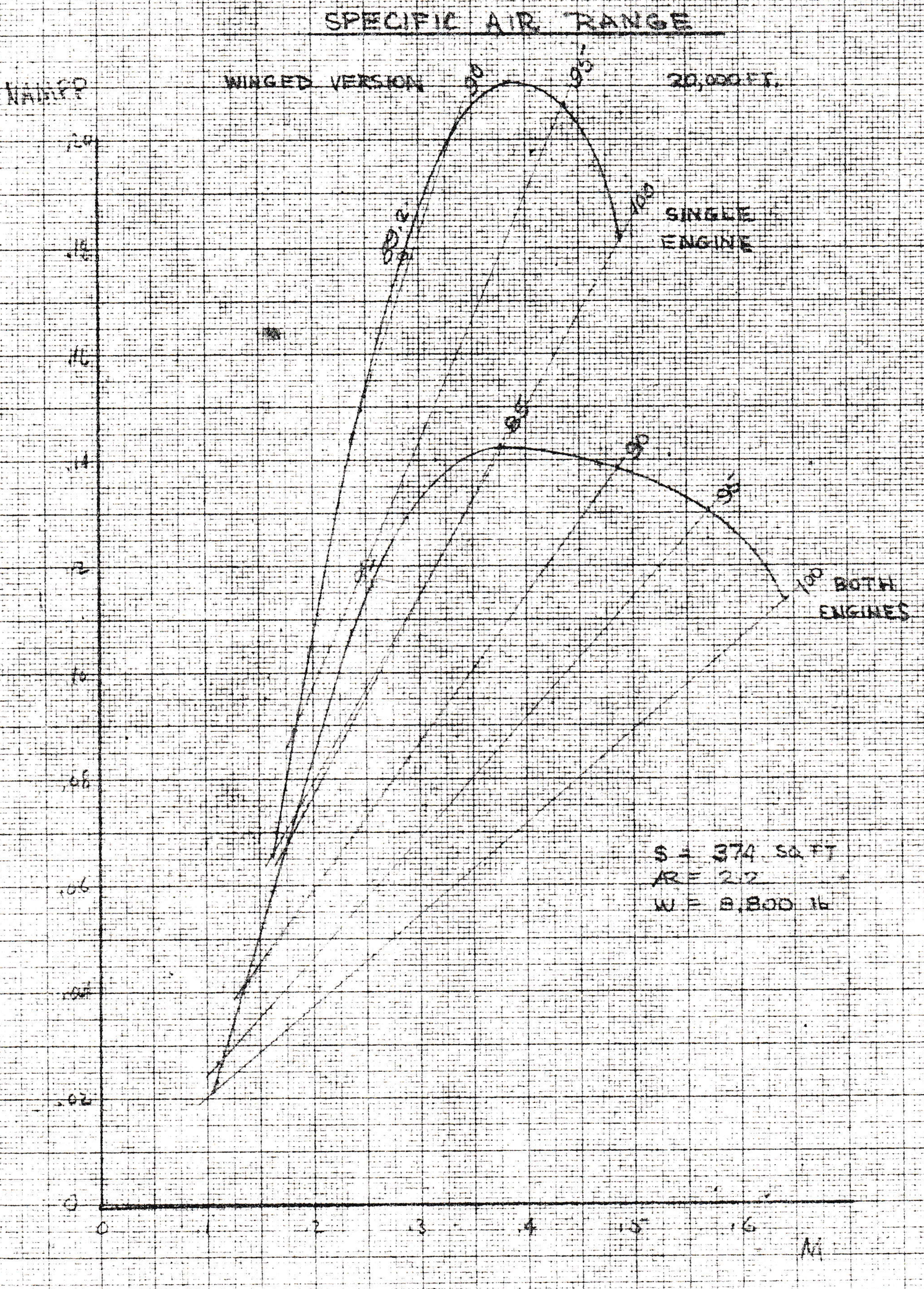

SPECIFIC AIR RANGE
WINGED VERSION
20,000 FT.
NAMI/PP
SINGLE ENGINE
BOTH ENGINES
S = 374 SQ FT
AR = 2.2
W = 8,800 lb
M
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0
.1
.2
.3
.4
.5
.6

FIG 36

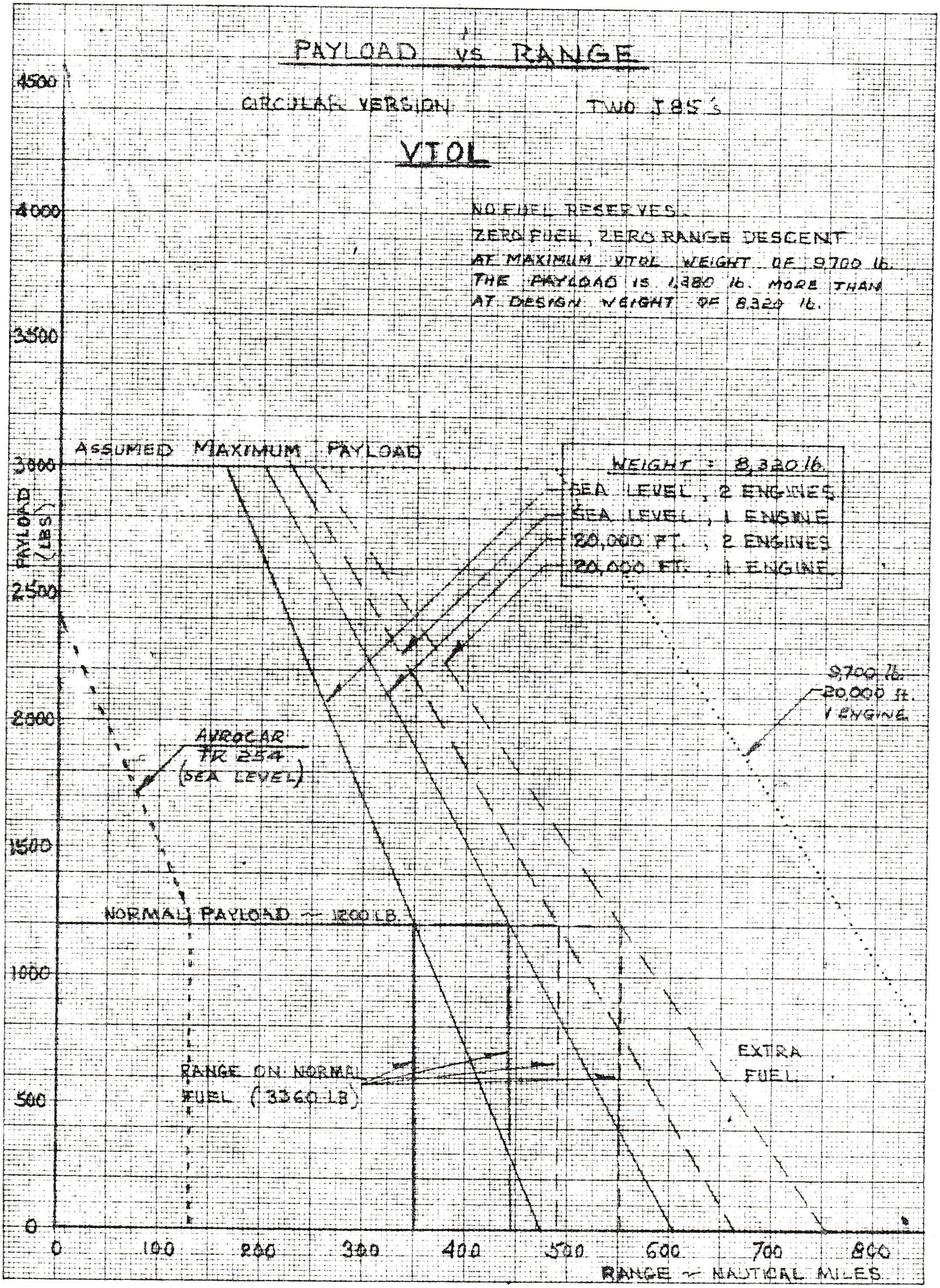

PAYLOAD vs RANGE
CIRCULAR VERSION
TWO J85's
VTOL
NO FUEL RESERVES.
ZERO FUEL, ZERO RANGE DESCENT
AT MAXIMUM VTOL WEIGHT OF 9,700 lb.
THE PAYLOAD IS 1,380 lb. MORE THAN
AT DESIGN WEIGHT OF 8,320 lb.
ASSUMED MAXIMUM PAYLOAD
WEIGHT = 8,320 lb.
SEA LEVEL, 2 ENGINES
SEA LEVEL, 1 ENGINE
20,000 FT., 2 ENGINES
20,000 FT., 1 ENGINE
9,700 lb.
20,000 ft.
1 ENGINE
AVROCAR
TR 254
(SEA LEVEL)
NORMAL PAYLOAD ~ 1200 LB.
RANGE ON NORMAL
FUEL (3360 LB)
EXTRA
FUEL
PAYLOAD (LBS)
RANGE ~ NAUTICAL MILES
4500
4000
3500
3000
2500
2000
1500
1000
500
0
100
200
300
400
500
600
700
800

FIG 37

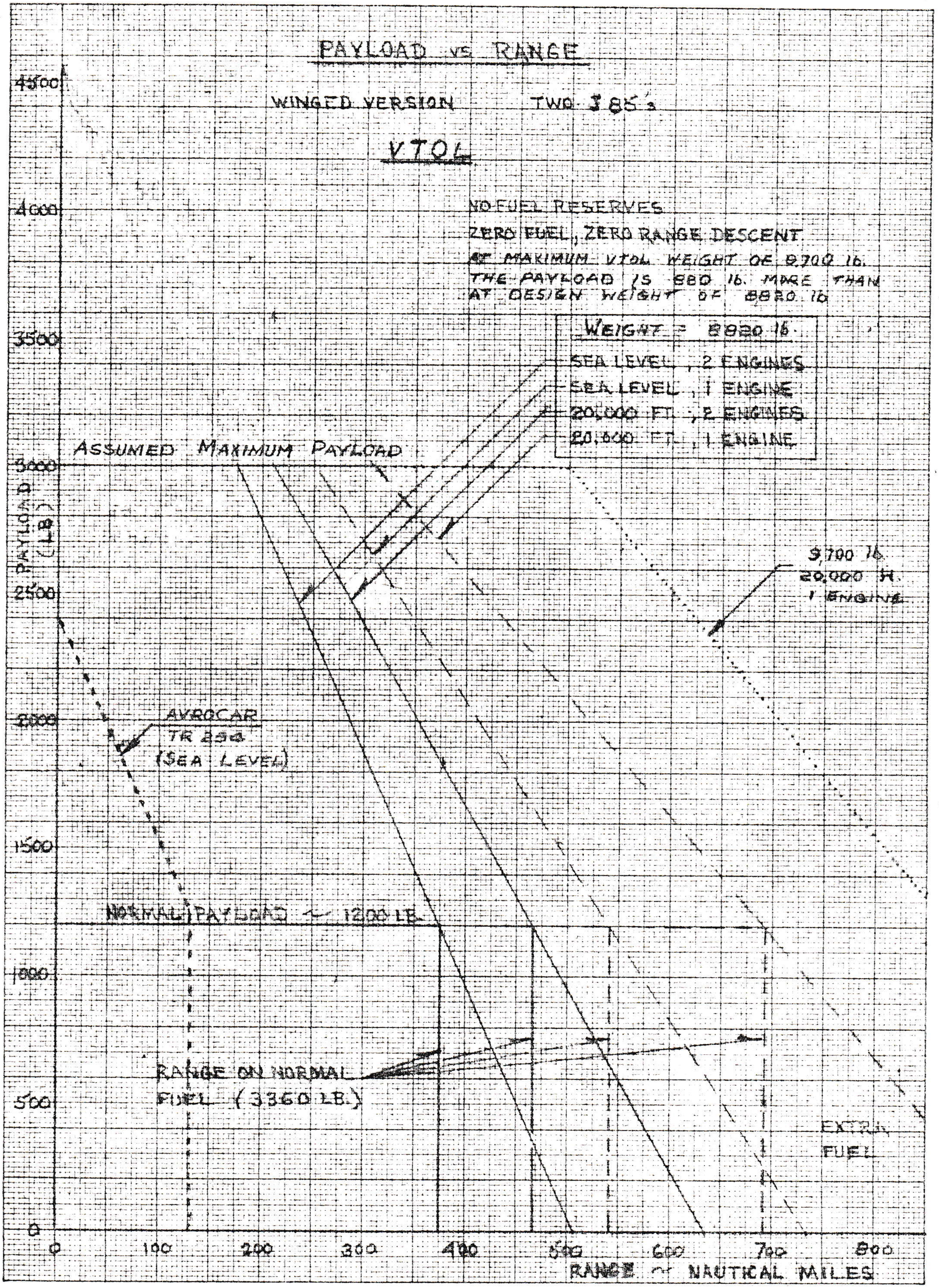

PAYLOAD vs RANGE
WINGED VERSION TWO J85's
VTOL
NO FUEL RESERVES
ZERO FUEL, ZERO RANGE DESCENT
AT MAXIMUM VTOL WEIGHT OF 9700 lb.
THE PAYLOAD IS 880 lb. MORE THAN
AT DESIGN WEIGHT OF 8820 lb
WEIGHT = 8820 lb
SEA LEVEL, 2 ENGINES
SEA LEVEL, 1 ENGINE
20,000 FT, 2 ENGINES
20,000 FT, 1 ENGINE
ASSUMED MAXIMUM PAYLOAD
9,700 lb
20,000 ft
1 ENGINE
AVROCAR
TR 254
(SEA LEVEL)
NORMAL PAYLOAD – 1200 LB.
RANGE ON NORMAL
FUEL (3360 LB.)
EXTRA
FUEL
PAYLOAD (LB.)
RANGE – NAUTICAL MILES
4500
4000
3500
3000
2500
2000
1500
1000
500
0
100
200
300
400
500
600
700
800

FIG 38

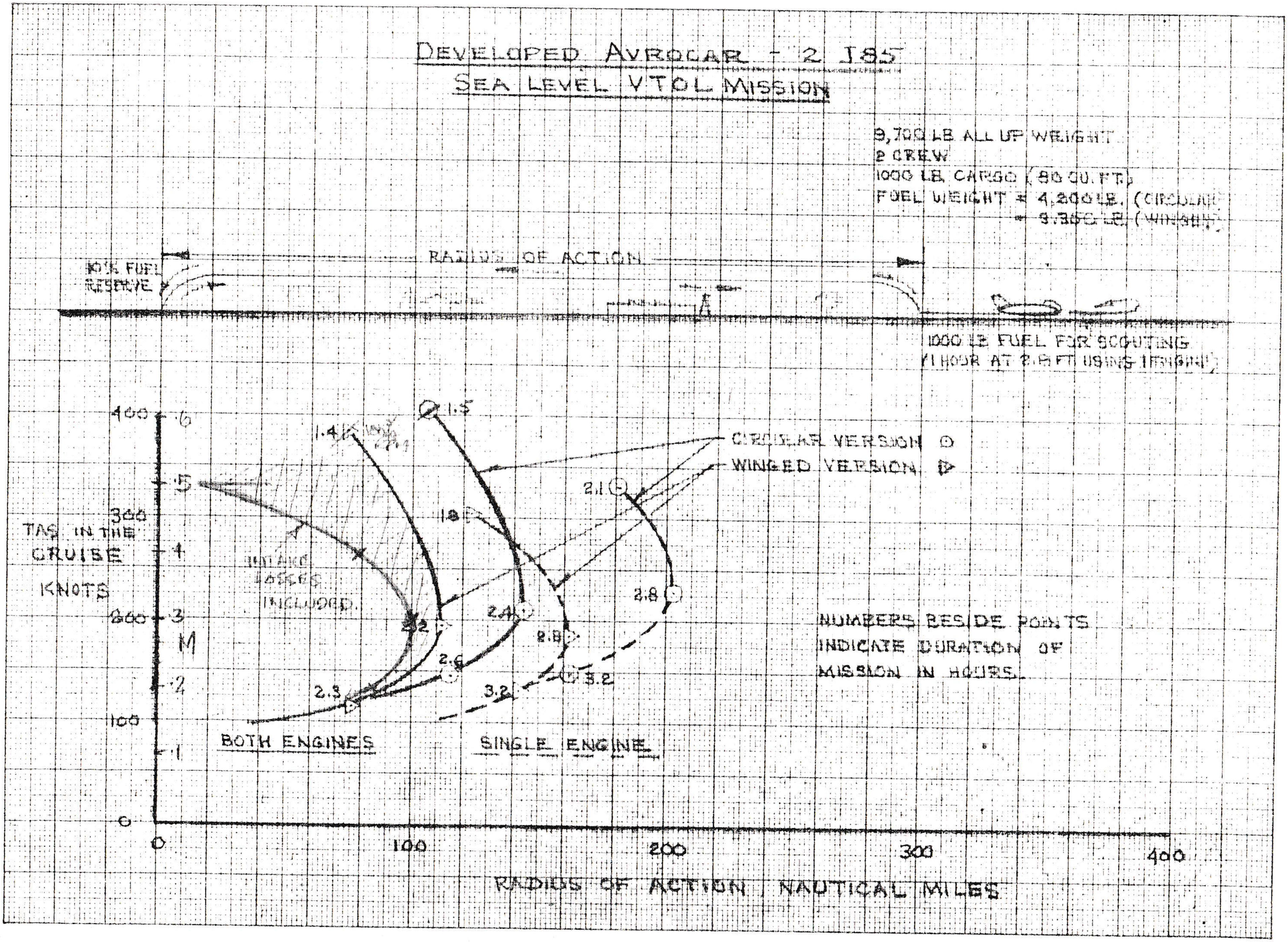

DEVELOPED AVROCAR - 2 J85
SEA LEVEL VTOL MISSION
9,700 LB ALL UP WEIGHT
2 CREW
1000 LB. CARGO (80 CU. FT.)
FUEL WEIGHT = 4,200 LB. (CIRCULAR)
= 3,350 LB. (WINGED)
RADIUS OF ACTION
10% FUEL RESERVE
1000 LB FUEL FOR SCOUTING
(1 HOUR AT 8 FT USING 1 ENGINE)
CIRCULAR VERSION
WINGED VERSION
INTAKE LOSSES INCLUDED
NUMBERS BESIDE POINTS INDICATE DURATION OF MISSION IN HOURS.
TAS IN THE CRUISE KNOTS
M
BOTH ENGINES
SINGLE ENGINE
RADIUS OF ACTION NAUTICAL MILES
0
100
200
300
400
1.4
1.5
1.8
2.1
2.2
2.3
2.4
2.6
2.8
2.8
3.2
3.2

FIG 39

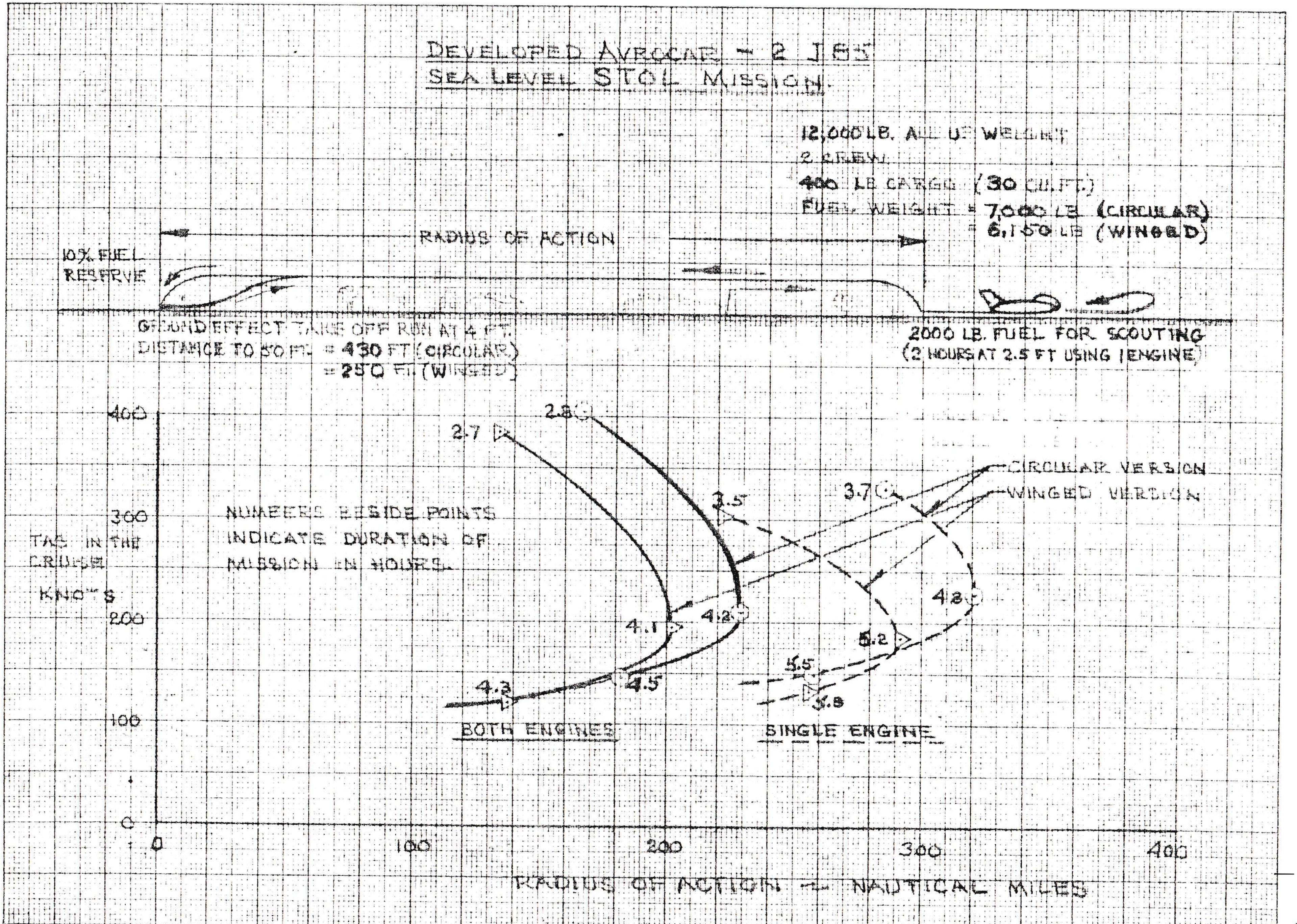

DEVELOPED AVROCAR - 2 J85
SEA LEVEL STOL MISSION.
12,000 LB. ALL UP WEIGHT
2 CREW.
400 LB CARGO (30 CU.FT.)
FUEL WEIGHT = 7,000 LB (CIRCULAR)
= 6,150 LB (WINGED)
RADIUS OF ACTION
10% FUEL RESERVE
GROUND EFFECT TAKE OFF RUN AT 4 FT.
DISTANCE TO 50 FT. = 430 FT (CIRCULAR)
= 250 FT (WINGED)
2000 LB. FUEL FOR SCOUTING
(2 HOURS AT 2.5 FT USING 1 ENGINE)
NUMBERS BESIDE POINTS INDICATE DURATION OF MISSION IN HOURS.
TAS IN THE CRUISE KNOTS
CIRCULAR VERSION
WINGED VERSION
2.7
2.8
3.5
3.7
4.1
4.2
4.3
4.5
4.8
5.2
5.5
5.8
BOTH ENGINES
SINGLE ENGINE
0
100
200
300
400
RADIUS OF ACTION - NAUTICAL MILES

FIG 40