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L.O. 9542A FILE BM49-7-12 PREPARED BY OEM CHECKED BY RJT RJ	NATIONAL AERONAUTICAL ESTABLISHMENT OTTAWA, CANADA LABORATORY MEMORANDUM SECTION Aerodynamics	NO. AE-46a PAGE 1 OF 42 COPY NO. 4 DATE 25 Nov.,1953

DECLASSIFED on August 29, 2016 by Steven Zan.

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SECURITY CLASSIFICATION ~~Secret~~

SUBJECT ASSESSMENT OF THE PERFORMANCE CHARACTERISTICS OF THE PROPOSED A.V. ROE C-105/1200 ALL-WEATHER SUPERSONIC FIGHTER AIRCRAFT.

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ISSUED TO Internal

SUMMARY

An assessment of the performance characteristics of the proposed A.V. Roe C-105/1200 All-weather Supersonic Fighter Aircraft has been undertaken by the Aerodynamics Laboratory of the National Aeronautical Establishment upon request from the Royal Canadian Air Force.

It is found that considerable differences exist between the present analysis and the A.V. Roe analysis with respect to the drag and performance characteristics of the aircraft. Contrary to the A.V. Roe design studies, and the conclusions reached by the Company on the basis of the recent subsonic and transonic wind tunnel test results, it is found that the aircraft fails to meet the R.C.A.F. specifications for minimum combat performance and combat radius of action. The differences between the two supersonic drag estimates account for the major differences in performance and extensive supersonic wind tunnel tests are probably required before these differences can be decisively resolved.

1.0 INTRODUCTION

The Aerodynamics Laboratory of the National Aeronautical Establishment has been requested by the Royal Canadian Air Force to check the performance estimates of the proposed C-105/1200 supersonic All-weather Interceptor Aircraft as given in A.V. Roe Report No. P/C-105/1. (Reference 3).

The present report deals with the general aerodynamic characteristics and performance of the C-105/1200 aircraft when powered by two Rolls Royce RB 106 engines fitted with afterburners (Reference 7). Complete performance estimates for the other two engine proposals (the Wright J67 and the Bristol B. OL. 4) are not available to this Laboratory at the present time, and detailed performance estimates pertaining to the aircraft when powered by these powerplants are thus not attempted. However, it is felt that the trends in agreement, or disagreement, between the N.A.E. and the A.V. Roe analyses are independent of the specific engines considered.

The R.C.A.F. Specification AIR 7-3 and the Operational Requirement OR 1/1-63 (References 1 and 2) are used as the standard against which the performance of the proposed aircraft is assessed. The main material supporting the present assessment and the detailed comparison with the A.V. Roe results are contained in the Appendices.

2.0 DRAG CHARACTERISTICS

In order to estimate the performance characteristics of a given aircraft, a detailed knowledge of the thrust and drag characteristics of the aircraft is imperative. The Rolls Royce RB 106 thrust characteristics, as presented in Reference 7, are assumed a priori throughout the present analysis. The drag estimates are, in general, derived from a large number of references representing experimental, empirical and theoretical considerations. It must be remembered, however, that transonic and supersonic estimates cannot, as yet, be made accurately.

2.1 Profile Drag

The present profile drag analysis (Figure 1) shows that the C-105/1200 aircraft is a relatively clean aerodynamic configuration. The subsonic aerodynamic cleanness is about the same as for the well known F-80 and F-86 subsonic fighters. The supersonic drag coefficient is less than twice the value of the subsonic drag coefficient. The total wave drag is hence of the same order as the total supersonic skin friction drag. The wave drag of the wing and the vertical tail is almost negligible as a result of the thin sections employed in the design. However, the wave drag of the fuselage is large and the total drag contribution of this part of the aircraft is consequently more than fifty percent of the total supersonic profile drag. A reduction of 20 percent in fuselage maximum cross-sectional area would thus reduce the total supersonic profile drag by almost 15 percent if the effect resulting from changing the fineness ratio is included.

The comparison between the present profile drag estimate and previous estimates for the C-104/1 and the C-104/2 aircraft (Figure 2) indicates that the present estimate is in good agreement with both the N.A.E. and the A.V. Roe estimates for the C-104/1. The A.V. Roe estimate for the C-104/2, considered by the Company to apply to the C-105/1200 as well, is considerably lower than all the others. In particular, the A.V. Roe estimate is almost 25 percent less than the present one at a Mach number of 1.5. Detailed considerations, as described in Appendix A, conveys the impression that the A.V. Roe estimate is optimistic and inconsistent with the Company's own profile drag estimate for the C-104/1 aircraft.

The recent Cornell wind tunnel tests of the C-105 model (References 8, 9 and 10) fail to indicate the actual profile drag of the aircraft even at subsonic and transonic speeds. It is pointed out in Reference 10 that the values derived therein may be subject to considerable error.

2.2 Drag Due to Lift

The present drag due to lift analysis is in excellent agreement with the Cornell test results at subsonic and transonic speeds (Figure 4). The A.V. Roe estimate is more conservative than the present one at transonic and supersonic Mach numbers.

The total drag due to lift for the C-105/1200 aircraft is large as a result of the low value of aspect ratio. However, it is interesting to note that a considerable reduction in the value of the drag due lift factor occurs just before the drag rise Mach number is reached. It will be recalled that if the profile drag coefficient and the drag due to lift factor are considered constant, the value of the minimum drag is the same at all altitudes provided the aircraft is at the minimum drag speed. In this case, however, an optimum value of the minimum drag will exist when the aircraft is at the altitude where the Mach number for the minimum drag is about M = 0.95. Since the thrust required is an absolute minimum, it is probable that this condition will also define the optimum cruising altitude and speed.

2.3 Trim Drag

The present trim drag analysis is based on the Cornell test results in the subsonic and transonic Mach number range. The values of the various coefficients at supersonic speeds are obtained by extrapolation using the A.V. Roe estimates and/or available experimental evidence as a guide (Refer to Figures 5 to 9).

The effects of negative wing camber on the trim drag are discussed in detail in Appendix A. The trimmed lift coefficient for zero elevator drag for the C-105/1200 aircraft is found to be 0.0486 at a Mach number of 1.5. This corresponds to level flight at combat weight at 57,000 feet altitude.

The net elevator effectiveness is considerably smaller than the elevator pitching moment effectiveness at supersonic speeds as a result of the short elevator moment arm and the large value of the static margin. The elevator drag is inversely proportional to the net elevator effectiveness and is hence extremely sensitive to the value of the elevator pitching moment effectiveness at supersonic speeds. It is shown in Appendix A that this value at M = 1.5 cannot be predicted with absolute certainty to a higher accuracy than ±25 percent at the present time. (Refer to Figure 7). The resulting extreme values of elevator drag thus differ by a factor of five. The values of the elevator pitching moment effectiveness assumed in the present analysis appear somewhat optimistic at supersonic speeds compared with the values obtained from experimental data. However, the A.V. Roe estimates are considerably more optimistic than the present ones at low supersonic Mach numbers and represent the extreme upper limit at M = 1.5.

3.0 COMBAT PERFORMANCE

It was pointed out in the previous section that considerable difference exists between the N.A.E. and the A.V. Roe drag estimates at supersonic speeds. Since the most important differences are unlikely to be resolved before complete supersonic wind tunnel test results are available, the present analysis attempts to clarify the effects of these differences on the combat performance.

The present analysis indicates that the C-105/1200 fighter aircraft fails to meet the minimum combat performance as specified by the R.C.A.F. If we define the relative combat effectiveness of a fighter, with respect to this specification, as 100 percent for a combat load factor of 2 and zero percent for a combat load factor of unity at combat height and speed, the relative combat effectiveness is 85 percent (Figure 10). However, the load factor increases with Mach numbers above M = 1.1, and the optimum value occurs at a Mach number of about 1.9 and is slightly greater than 2.

Both the minimum radius of sustained, level turn and the minimum time to complete a 360 degree level turn increase steadily with Mach number above M = 0.95 in spite of the increase in load factor (Figure 12). The aircraft thus fails to meet the indirectly specified values of combat radius of turn and time to turn at Mach numbers equal or greater than the specified combat value.

Reference 3 gives some engine characteristics for the two alternative powerplants. It is found that the relative combat effectiveness of the aircraft is 62 percent with the Wright J.67 engines and 42 percent with the Bristol B.OL.4 engines if these engine data are used.

The effect of variations in the value of the profile drag coefficient on the maximum load factor in sustained, level turn is significant throughout the supersonic Mach number range. The relative combat effectiveness is found to be 103 percent if the estimated N.A.E. profile drag is replaced by the A.V. Roe estimate.

A substantial gain in combat load factor is obtained by the use of negative wing camber. The optimum load factor without camber occurs at the drag rise Mach number and the relative combat effectiveness is only 60 percent. Since the relative combat effectiveness is 137 percent if zero trim drag is assumed, it may appear advantageous to employ more negative wing camber. The camber required to give (theoretically) zero trim drag in a maximum rate steady turn at combat height and speed is found to be minus 5 percent. The combat load factor for zero trim drag is 2.37 g. (Figure 10).

However, this amount of camber on a thin delta wing appears unacceptable for several reasons and would change the basic drag characteristics appreciably so that the benefit of zero trim drag would, in effect, be greatly reduced.

It is found that the combat load factor is extremely sensitive to the value of elevator pitching moment effectiveness (Figure 11). The relative combat effectiveness varies between 41 percent and 94 percent within the range of elevator pitching moment effectiveness values considered possible by extrapolating the Cornell test results to the combat Mach number. A relative combat effectiveness of 68 percent is obtained on the basis of the experimental elevator pitching moment effectiveness data obtained from Reference 31.

If the A.V. Roe estimates of profile drag and elevator pitching moment effectiveness are used simultaneously, the resulting combat load factor compares well with the value presented in Reference 3. The main reasons for the difference in combat performance between the A.V. Roe analysis and the present analysis are thus due to the differing estimates of the profile drag and the elevator pitching moment effectiveness.

4.0 COMBAT CLIMB, ACCELERATION TIME AND CEILING

The present analysis indicates that the C-105/1200 fighter aircraft is well within the R.C.A.F. specification with respect to time to combat height and speed. No attempt has been made in the present analysis to find the minimum time to height, but it appears certain that this value will be between 2.5 and 3 minutes. The rates of climb at combat speed are considerably higher than at the drag rise speed above 12,000 feet altitude (Figure 13) so that the acceleration from M = 0.95 to M = 1.5 should probably be performed close to this altitude if minimum time to height is desired.

The flight plan assumed in the present analysis is based on consideration of minimum effective fuel to height with the engine operating at maximum reheated thrust. The latter condition assures that the time to height is reasonably short. Detailed considerations, described in Appendix C, indicate that a drag rise Mach number climb from sea level to 36,000 feet and acceleration from M = 0.95 to M = 1.5 at the tropopause followed by a constant combat Mach number climb to 50,000 feet yields closely the minimum effective fuel consumption to combat height and speed with the engines operating at maximum reheated thrust. The resulting time, from a position of rest at sea level to combat height and speed (Figure 15), is found to be 3.35 minutes and is thus well within the specified value of 6 minutes. The horizontal distance covered during this flight plan (Figure 16) is found to be 32 nautical miles and the total fuel consumption (Figure 17) is 4,777 pounds, including the fuel used during taxi and warm-up.

The difference between the calculated rates of climb for the N.A.E. and the A.V. Roe profile drag estimates is negligible for M = 0.95 within the troposphere. However, the rates of climb pertaining to the A.V. Roe profile drag analysis at M = 1.5 above the tropopause are considerably higher than the values obtained with the N.A.E. estimate. Corresponding differences are obtained with respect to time to height, acceleration time, horizontal distance and fuel consumption during climb. (Refer to Figures 13 and 15 to 17).

The time to combat height and speed presented in Table V of Reference 3 is in reasonably good agreement with the values obtained in the present analysis. However, the value of horizontal distance covered during the climb procedure given in Reference 3 is considerably larger than the various values obtained herein. Since the flight plan assumed by Reference 3 is unknown to the Aerodynamics Laboratory, no explanation of these differences can be offered at present. The fuel to combat height and speed quoted in Reference 3 is less than the value obtained in the present analysis with the N.A.E. profile drag estimate, but compares well with the value obtained for the A.V. Roe estimate.

The maximum value of combat ceiling is found to be almost 65,000 feet at a Mach number of 1.9 in the present analysis (Figure 18). This is well above the minimum value at 60,000 feet specified by the R.C.A.F. The present estimates are in reasonably good agreement with the values presented in Reference 3 considering the differences in the N.A.E. and the A.V. Roe drag estimates.

5.0 COMBAT RADIUS OF ACTION

The various phases of the combat radius of action are described in detail in Appendices C and D. Considerable attention has been given to the problem of finding optimum solutions, within the specifications, for the various phases involved. Only two minor items are somewhat indeterminate, namely the fuel consumed during the taxi and warm-up phase and during the post-combat descent from 50,000 feet to sea level. The value obtained for the first item agrees closely with the value obtained by A.V. Roe and is thus not controversial. The values obtained for the second item are, although not directly comparable, less than one half of the value presented in Reference 3 and may thus be disputed as being optimistic.

The values obtained must, in general, be considered as the most favourable ones that can be logically presented on the basis of the present estimates of the engine and aerodynamic characteristics of the aircraft.

It will be seen from Table I that the C-105/1200 aircraft does not meet the minimum combat radius of action specified by the R.C.A.F. according to the present analysis. The radius of action is found to be only 142 nautical miles with the present fuel capacity of 12,900 pounds. The extra fuel needed to meet the specification is over 1500 pounds on the assumption of an aircraft gross weight of 48,400 pounds. If, however, any additional fuel weight must be considered additive to the present value of aircraft gross weight, considerably more fuel is required and the combat performance will suffer correspondingly.

The combat radius of action obtained in the present analysis with the A.V. Roe profile drag estimate is 186 nautical miles and the additional fuel required is 300 pounds assuming an aircraft gross weight of 48,400 pounds.

By comparing the fuel consumption values obtained in the present analysis with the values presented in Reference 3, it is noted that considerably higher fuel consumptions are obtained in the present analysis for the pre-combat and the combat phases of action. The differences with respect to the pre-combat phase are basically due to the differences in the drag estimates and the large differences obtained in the horizontal distance covered during the climb and acceleration phase in the two analyses. The combat fuel consumption value presented in Reference 3 is considerably lower than the value obtained by assuming maximum reheated thrust for 5 minutes at combat height and speed.

The fuel consumption values obtained for the post-combat phase in the present assessment are lower than the value obtained from the A.V. Roe analysis in spite of the differences in drag estimates and the favourable assumption made in Reference 3 that 64 miles of the return radius can be covered during the descent to sea level. This results partly from the effort made in the present analysis to find optimum solutions, and partly from the favourable suppositions made with respect to the fuel consumed during the descent to sea level.

6.0 ADDITIONAL PERFORMANCE ITEMS

Preliminary considerations indicate that similar differences to those obtained in the previous sections exist between the present analysis and the A.V. Roe analysis with respect to the cruising radius of action. The overload range and the take-off and landing performance of the C-105/1200 aircraft have not been investigated in the present analysis.

7.0 CONCLUSIONS

An assessment of the aerodynamic characteristics and performance of the proposed A.V. Roe C-105/1200 Supersonic, All-weather Interceptor Aircraft has been made by the Aerodynamics Laboratory at the National Aeronautical Establishment. The following conclusions can be drawn from this investigation:

(a) Considerable differences exist between the present analysis and the A.V. Roe analysis with respect to the drag characteristics of the aircraft. The net effect of these differences is, in general, to give more conservative values of drag in the present assessment than obtained by the Company, particularly at supersonic speeds. Accurate supersonic wind tunnel test results are probably required before these differences can be definitely resolved.

(b) The aircraft fails to meet the minimum combat performance specified by the R.C.A.F., on the basis of the N.A.E. drag estimates. The combat load factor is found to be quite sensitive to the value of aircraft profile drag and wing camber and extremely sensitive to the value of elevator pitching moment effectiveness. With the present drag estimates, and the estimated performance of the Rolls Royce R.B. 106 engines, the combat load factor for the C-105/1200 is 1.85. The corresponding value for the Wright J.67 engine characteristics is 1.62 and for the Bristol B.O.L.4 engines 1.42. Experimental evidence indicates, however, that the assumed value of elevator pitching moment effectiveness at combat speed may be optimistic and the combat load factor on the basis of these experimental results is only 1.68 with the R.B. 106 engines.

(c) The C-105/1200 fighter aircraft is well within the R.C.A.F. specification with respect to time to combat height and speed. It appears that the minimum time to height will be between 2.5 and 3 minutes with the R.B. 106 engines.

(d) The aircraft does not appear to meet the R.C.A.F. specification for minimum combat radius of action. The radius of action is found to be only 142 nautical miles with the present fuel capacity of

12,900 pounds for the R.B. 106 engines and the present drag estimates. The additional fuel needed to meet the specification of 200 nautical miles combat radius of action is over 1,500 pounds even if the present aircraft gross weight is assumed.

(e) The remaining performance items required to meet the R.C.A.F. performance specifications have not been investigated in the present analysis.

COMBAT RA

Aircraft: C-105/1200
Engines: Two RB 106
Gross Weight: 48,400 pounds.
*1,312 pounds ammunition fired

Phase of Action	Horizontal Distance - nautical miles			N.A.
	N.A.E. Analysis		A.V. Roe Analysis	
	N.A.E. C_{D_o}	A.V. Roe C_{D_o}		N.A.E.
Taxi and Warm-up	-	-	-	4.0
Combat Climb and Acceleration	32.0	27.9	39	3.34
Combat Cruise	(168.0) 109.7	(172.1) 158.0	161	(11.7) 7.64
Combat*	-	-	-	5.0
Pre-combat and Combat	(200) 141.7	(200) 185.9	200	(24.04) 19.98
Return to Base	(200) 141.7	(200) 185.9	136	(22.0) 15.6
Loiter	-	-	-	15.0
Descent to Sea Level	-	-	64	6.0
Loiter Reserve	-	-	-	5.0
Post-combat	(200) 141.7	(200) 185.9	200	(48.0) 41.6
Total	(400) 283.4	(400) 371.8	400	(72.04) 61.48

COMBAT RADIUS OF ACTION

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Note: The values in brackets pertain to the required radius of action of 200 n.m. The other values pertain to the available fuel weight of 12,900 pounds.

-	Time - minutes			Fuel Consumption - pounds		
	N.A.E. Analysis			N.A.E. Analysis		
.V. Roe Analysis	N.A.E. C_{D_o}	A.V. Roe C_{D_o}	A.V. Roe Analysis	N.A.E. C_{D_o}	A.V. Roe C_{D_o}	A.V. Roe Analysis
-	4.0	4.0	4.0	663	663	660
39	3.34	3.01	3.2	4,114	3,757	3,740
161	(11.7) 7.64	(12.0) 11.0	11.2	(3,180) 2,103	(2,425) 2,230	2,230
-	5.0	5.0	5.0	3,265	3,265	3,050
200	(24.04) 19.98	(24.01) 23.01	23.4	(11,222) 10,145	(10,110) 9,915	9,680
136	(22.0) 15.6	(22.0) 20.45	15.6	(1,570) 1,113	(1,498) 1,393	1,100
-	15.0	15.0	15.0	885	850	875
64	6.0	6.0	7.8	320	320	710
-	5.0	5.0	5.0	437	422	535
200	(48.0) 41.6	(48.0) 46.45	43.4	(3,212) 2,755	(3,090) 2,985	3,220
400	(72.04) 61.48	(72.01) 69.46	66.8	(14,434) 12,900	(13,200) 12,900	12,900

NOTATION

General:

M	Mach number.
V	forward speed, feet per second.
ρ	air density, slugs per cu. ft.
q	dynamic pressure, lb. per sq. ft. ($q = \frac{1}{2}\rho V^2$).
S	total wing area, sq. ft.
W	aircraft weight, lb.
T	thrust
C_D	total drag coefficient.
C_{D_o}	profile drag coefficient.
C_L	lift coefficient in trimmed flight.
$dC_D/dC_L{}^2$	drag due to lift factor.
α	angle of attack, degrees.

Trim drag:

δ	elevator deflection, degrees.
$(C_{M_o})_{\delta=0}$	zero lift pitching moment coefficient for zero elevator deflection.
$(dC_M/dC_L)_{\delta=0}$	static margin for zero elevator deflection.
$(C_{M_\delta})_{\alpha=\text{const.}}$	elevator pitching moment effectiveness for constant angle of attack.
$(C_{L_\delta})_{\alpha=\text{const.}}$	elevator lift effectiveness for constant angle of attack.
$(dC_D/d\delta^2)_{C_L=0}$	elevator drag factor for zero lift.
C_{D_t}	drag coefficient due to trim.

Performance:

n	maximum load factor in steady, level turn, g's.
R	minimum radius of steady, level turn, nautical miles.
T_t	minimum time to complete 360 degree level turn, minutes.
R/C	rate of climb, ft./minute.
θ	angle of climb, degrees.

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APPENDIX A

DRAG CHARACTERISTICS

The drag estimates are, in general, based on the data of a large number of references and the values obtained thus represent average values resulting from experimental, empirical and theoretical considerations.

Profile Drag

The estimated variations of the profile drag contributions of the various parts of the C-105/1200 aircraft, with flight Mach number, are shown in Figure 1. It is notable that the total supersonic drag coefficient is less than twice the value of the total subsonic drag coefficient everywhere. The fuselage contribution, being more than one half the total drag coefficient, is by far the largest one at supersonic Mach numbers.

Figure 2 shows the comparison of four estimates of total drag coefficients for similar aircraft configurations. It will be noted that both estimates for the C-104/1 aircraft agree reasonably well with the present estimate for the C-105/1200 aircraft at subsonic and transonic Mach numbers. The present estimate for the C-105/1200 is almost 10 percent higher than the A.V. Roe estimate for the C-104/1 and $3\frac{1}{2}$ percent higher than the N.A.E. estimate for the C-104/1 aircraft at supersonic Mach numbers. The A.V. Roe estimate for the C-104/2 aircraft is considerably lower than all the other estimates. (It is understood that the A.V. Roe C-104/2 profile drag and drag efficiency estimates are considered by the Company to apply directly to the C-105/1200 as well. (Compare Figures 21 and 22 of Reference 5 with Figures 4.1 and 4.2 of Reference 10.)) In particular, at $M = 1.5$, the A.V. Roe estimate is almost 25 percent less than the present estimate. This difference is rather large and has, as shown later, a significant effect on the performance of the aircraft.

Figure 3 indicates that the main difference between the two estimates is due to differences in estimating the drag contributions of the basic components, wing, body and vertical tail. A closer comparison of Figure 1 of the present analysis and Figure 21 of Reference 5 will reveal perfect agreement with respect to the vertical tail contribution, 20 percent difference with respect to the wing contribution and 27.5 percent difference with respect to the fuselage contribution at M = 1.5.

It appears that the wing drag contribution estimated by A.V. Roe was based primarily on the data of Reference 17. However, it will be noted that the estimated wing drag contribution at M = 1.5 is 25 percent less than the mean of the two wing drag values shown on Figure 5c of Reference 17. The mean Reynolds number correction for these data, as estimated in the present analysis, was minus 6 percent. The corrected wing drag value from Reference 17 thus agrees with the value obtained in the present analysis.

The largest difference between the two profile drag estimates is that due to the fuselage contribution. In the present analysis it is assumed that the fusealge drag will be the same as that of a streamlined body of revolution of the same maximum cross-sectional area and length. This means that the increased drag due to irregular shape (intake shocks etc.) is assumed to be cancelled by the reduction in drag resulting from the flow through the engine ducts. Although this assumption is rather crude, it is backed up by some experimental evidence. References 24 and 25 show that the drag of a fuselage with air inlets is closely the same as that of a streamlined body of revolution with the same cross-sectional area provided the mass flow ratio is close to unity. The additional drag at lower mass flow ratios is taken into account by a representative spillage drag term. The base drag is considered zero as a result of the jet exhaust.

Comparison of the geometric characteristics of the C-105/1200 fuselage with the C-104/1 fuselage reveals that the ratio of fuselage maximum cross-sectional area to wing area is 2.2 percent less for the C-105/1200 than for the C-104/1 whereas the fineness ratio for the C-105/1200 is 7.78 versus 9.13 for the C-104/1. Since the C-105/1200 total wing area is closely twice the C-104/1 wing area and the former needs twice the inlet mass flow of the latter, one might expect the effect of the intakes to be the same for the two aircraft. The main differences in fuselage drags to be expected in this case are thus due to the differences in the ratio of fuselage area to wing area, fineness ratio and Reynolds number. From Reference 20 it will be found that the change in drag by decreasing the fineness ratio from 9.13 to 7.78 at $M = 1.5$ is plus 13 percent. The estimated Reynolds number correction at this Mach number is minus 2.3 percent. The fuselage drag coefficient for the C-105/1200 can thus be derived by correcting the A.V. Roe estimate for the C-104/1 fuselage for the above factors (at $M = 1.5$):

$$\underline{C_{D_{fus.\,105}}} = C_{D_{fus.\,104/1}}(1 - 0.022 + 0.130 - 0.023)$$

$$= 1.085\ C_{D_{fus.\,104/1}}$$

$$= 1.085 \times 0.00885 = \underline{0.00960}$$

(Refer to Figure 21 of Reference 4)

It will be noted that this compares well with the value obtained in the present analysis.

The recent Cornell wind tunnel tests of the C-105 model (References 8, 9 and 10) fail, unfortunately, to indicate the actual profile drag of the aircraft even at subsonic and transonic speeds. The actual values obtained are considerably larger than all the estimates (Reference 8, Figure 3.1 or 3.8) as a result of the internal duct drag and the base drag due to the sting interference. A correction is applied to account for these effects in Figure 4.1 of Reference 10. It is interesting to note that the estimated correction is larger than the total fuselage drag contribution as estimated by A.V. Roe for Mach numbers below 1.00 and above 1.10. For example, the correction is 160 percent at $M = 0.90$ and 130 percent at $M = 1.23$ of the total estimated fuselage drag. However, Reference 10 points out that the corrections may be subject to considerable error.

Drag Due to Lift

The estimated variation of the total drag due to lift with flight Mach numbers is shown in Figure 4. Considering that this estimate was made prior to the completion of the Cornell wind tunnel tests, the agreement with the test results at subsonic and transonic Mach numbers is extremely gratifying. It is, however, admitted that an agreement to this extent certainly is partly accidental. The supersonic estimates were based on a correlation of experimental data for more than 100 wings at present being prepared in the Aerodynamics Laboratory. The A.V. Roe estimate is considerably more conservative than the present estimate at transonic and supersonic speeds.

Trim Drag

A detailed trim drag analysis for the C-105/1200 aircraft was not completed by the Aerodynamics Laboratory before the Cornell wind tunnel test results were available. The variations with flight Mach numbers of the various aerodynamic coefficients needed for trim drag calculations in the present analysis are therefore based on the Cornell test results in the subsonic and transonic Mach number range and extrapolated to supersonic Mach numbers using the A.V. Roe estimates and/or available experimental evidence as a guide.

The variations of the zero lift pitching moment coefficient with Mach numbers are shown in Figure 5 for both the cambered and the uncambered wing configurations. The negative camber of the Cornell model was 3/4 percent. Although it is believed that the A.V. Roe estimate applies to the pitching moment variation due to ½ percent negative wing camber alone, the estimated curve compares reasonably well with the Cornell results for the cambered wing configuration and the curve assumed in the present analysis is thus extrapolated to supersonic Mach numbers in accordance with the estimate. The curve for the uncambered wing configuration is extrapolated partly on the assumption that the similarity to the cambered curve, obtained from the Cornell tests at subsonic and transonic Mach numbers, extends into the supersonic range and partly on the theoretical consideration of zero pitching moment coefficient at zero lift for symmetrical aerofoils past the transonic Mach number range.

Figure 6 shows the variation of the static margin with Mach number for zero elevator angle. The centre of gravity of the aircraft is assumed to be at 28 percent from the leading edge of the mean aerodynamic chord. Reference 31 indicates that the static margin has closely the same value throughout the supersonic Mach number range. This is also evidenced by recent tests on 60 degree delta wings in the high-speed tunnel of the

National Aeronautical Establishment (Reference 30), and the curve assumed in the present analysis is hence extrapolated to supersonic Mach numbers on this basis. The effect of camber on the static margin is negligible so that the curve applies to both configurations.

The assumed variation of the elevator pitching moment effectiveness for constant angle of attack with Mach number is shown in Figure 7. Reference 10 states that the experimental curve can be smoothly extrapolated to agree with the A.V. Roe estimates "above about $M = 1.5$". However, it will be seen from Figure 7 that the experimental curve can only be extrapolated smoothly to agree with the estimates at a Mach number of about 2. Corrected experimental data from Reference 31 fair in well with the Cornell test results, but give considerably more conservative values of the elevator pitching moment effectiveness in the higher supersonic Mach number range. Although the latter curve appears more probable than the former, being backed up by experimental evidence, the more favourable one has been assumed throughout the present analysis. The extreme possible limits in extrapolating the Cornell data to $M = 1.5$ are also shown to emphasize the importance of the value of this coefficient on combat performance as discussed in Appendix B.

Figure 8 shows the variation of the elevator lift effectiveness for constant angle of attack with Mach number. The curve is extrapolated to supersonic Mach numbers partly on the basis of the corrected data from Reference 31 and partly on the basis of the A.V. Roe estimates. The assumed curve is slightly more favourable than the A.V. Roe estimates above a Mach number of 1.4. (It will be shown later that it is desirable to have as high a value of elevator pitching moment effectiveness as possible associated with the lowest possible value of elevator lift effectiveness.)

The variation of the elevator drag factor for zero lift with Mach number is shown in Figure 9. The curve is extrapolated to supersonic Mach numbers on the basis of the A.V. Roe estimates and the data of Reference 31.

The effect of the elevator deflection on the induced drag is shown to be small in References 8, 9 and 10 and it has therefore been neglected in the present analysis.

It can be shown that the elevator angle required to trim the aircraft at any given flight condition is given by:

$$\delta = \frac{(C_{M_o})_{\delta=0} + \left(\frac{dC_M}{dC_L}\right)_{\delta=0} C_L}{(-C_{M_\delta})_{\alpha=\text{const.}} + \left(\frac{dC_M}{dC_L}\right)_{\delta=0} (C_{L_\delta})_{\alpha=\text{const.}}} \qquad (1)$$

where C_L is the value of the lift coefficient in trimmed flight.

Since Figures 5 to 8 give the values of the various coefficients involved in this expression for various Mach numbers, we can obtain the trim drag, given by

$$C_{D_t} = \left(\frac{dC_D}{d\delta^2}\right)_{C_L=0} \delta^2, \qquad (2)$$

for any given value of the trimmed lift coefficient, C_L.

(It should be noted that the notation for the elevator drag factor, $dC_D/d\delta^2$, (Figure 9) used herein differs from that used by A.V. Roe, $\Delta C_D/\delta^2$. The main reason for the change is the analogy between the elevator drag factor and the drag due to lift factor, dC_D/dC_L^2.)

An examination of expressions (1) and (2) in connection with Figures 5 to 9 will reveal the following interesting facts:

1. Since C_{M_o} is always positive for negative wing camber and dC_M/dC_L always is negative, there will be some positive values of the trimmed lift coefficient for which the elevator angle required to trim, and hence the trim drag, is zero. Furthermore, since the static margin is almost independent of camber (Reference 10), the value of the trimmed lift coefficient for zero elevator drag at a given Mach number can be adjusted to any desired value by changing the wing camber. For example, for the values assumed in the present analysis, the trimmed lift coefficient for the cambered wing at M = 1.5 is:

$$C_{L_t} = \left(\frac{C_{M_o}}{-\frac{dC_M}{dC_L}} \right)_{\substack{\delta=0 \\ M=1.5}} = 0.0486$$

For level flight conditions at combat weight and speed this corresponds to an altitude of about 37,000 feet.

2. Both C_{M_δ} and dC_M/dC_L are always negative whereas C_{L_δ} is always positive. The two terms in the denominator of expression (1) are therefore always of opposite sign and the net elevator effectiveness (the denominator) is lower than the elevator pitching moment effectiveness C_{M_δ}, particularly at supersonic speeds where the value of the static margin is large. Since the elevator drag is inversely proportional to the square of the net effectiveness, the importance of the value of C_{M_δ} becomes evident. For example, for the two extreme values of C_{M_δ} shown in Figure 7 at $M = 1.5$, the elevator drag assuming the lower value is 5 times the elevator drag assuming the higher value.

APPENDIX B

COMBAT PERFORMANCE

Paragraph 3.03 of Reference 1 reads:
"3.03.01. The minimum combat performance with internal armament installed shall be a combat speed of Mach 1.5 at a combat load factor of 2 and at a combat altitude of 50,000 feet."

The maximum load factor (as limited by thrust) in a sustained, level turn is given by:

$$n = \frac{qSK_1}{2WK_2}\left[1 + \sqrt{1 + \frac{4K_2}{(K_1)^2}\left(\frac{T}{qS} - C_{D_o}{}'\right)}\right], \qquad (3)$$

provided the total drag coefficient can be written in the form:

$$C_D = C_{D_o}{}' - K_1C_L + K_2C_L{}^2 \qquad (4)$$

where C_L is the trimmed lift coefficient.

It can be shown that the coefficients of Equation (4) are:

$$C_{D_o}{}' = C_{D_o} + \frac{dC_D}{d\delta^2}\left[\frac{C_{M_o}}{-C_{M_\delta} + \frac{dC_M}{dC_L}C_{L_\delta}}\right]^2 \qquad (5a)$$

$$K_1 = -\frac{2\,C_{M_o}\frac{dC_D}{d\delta^2}\frac{dC_M}{dC_L}}{\left(-C_{M_\delta} + \frac{dC_M}{dC_L}C_{L_\delta}\right)^2} \qquad (5b)$$

and $$K_2 = \frac{dC_D}{dC_L{}^2} + \frac{dC_D}{d\delta^2}\left[\frac{\frac{dC_M}{dC_L}}{-C_{M_\delta} + \frac{dC_M}{dC_L}C_{L_\delta}}\right]^2 \qquad (5c)$$

The subscripts $\delta = 0$, $\alpha = $ const. and $C_L = 0$ used in expressions (1) and (2) of Appendix A are excluded here for clarity.

The corresponding minimum radius of sustained level turn is given by:

$$R = 4.815 \frac{M^2}{\sqrt{n^2 - 1}} \text{ nautical miles} \qquad (6)$$

and the minimum time to complete a 360 degree level turn is:

$$T_t = 3.152 \frac{M}{\sqrt{n^2 - 1}} \text{ minutes} \qquad (7)$$

It should be noted that Equations (6) and (7), as they stand, are only valid above the tropopause.

The variation of the maximum load factor in a sustained, level turn at 50,000 feet with flight Mach number is shown in Figure 10. Four curves are presented to indicate the effects resulting from variations in the values of some of the important aerodynamic coefficients.

The solid curve, designated "N.A.E. C_{D_0} values", is based on the "present analysis" values of the various coefficients shown in Figures 1 to 9. It will be noted that the C-105/1200 fighter aircraft fails to meet the minimum combat performance as specified by the R.C.A.F. according to the present analysis. If we define the relative combat effectiveness of a fighter, with respect to this specification, as 100 percent for a combat load factor of 2 and zero percent for a combat load factor of unity of combat height and speed, the relative combat effectiveness of the C-105/1200 is found to be 85 percent in the present analysis. It is interesting to note that the optimum load factor occurs at a Mach number of about 1.9 and that its value is slightly greater than 2.

The broken curve, designated "A.V. Roe C_{D_0} values", is based on the present analysis values, shown in Figures 4 to 9, and the A.V. Roe C-104/2 profile drag estimate, shown in Figure 2. The increase in load factor resulting from the decreased profile drag is significant throughout the supersonic range and the relative combat effectiveness is thus found to be 103 percent.

The dotted curve, designated "No camber, N.A.E. C_{D_o}", is based on the present analysis values, but the zero lift pitching moment coefficient for the cambered configuration is replaced by that for the uncambered configuration as shown in the lower curve of Figure 5. It will be noted that the gain due to the present value of wing camber is significant throughout the whole supersonic range. The optimum load factor without wing camber occurs at the drag rise Mach number and the relative combat effectiveness is only 60 percent.

The dashed curve, designated "Zero trim drag, N.A.E. C_{D_o}", excludes the drag due to trim entirely. This curve is mainly of academic interest although it may appear possible to approach the load factor value indicated at a given Mach number if the wing camber is changed appropriately. The required value of the zero lift pitching moment coefficient to give zero trim drag in a 2.37 g turn, at combat height and speed, is 0.0475. The resulting camber is found to be minus 5 percent, assuming that the zero lift pitching moment coefficient varies linearly with camber. This degree of camber is, of course, unacceptable for several reasons on a thin delta wing, and would change the basic drag characteristics of the aircraft drastically so that the benefit of zero trim drag would, at least, be partly cancelled.

It is pointed out in Appendix A that the trim drag is extremely sensitive to the value of the elevator pitching moment effectiveness, and that this value cannot be predicted with certainty to within ±25 percent at combat speed at the present time. Since the combat load factor appears to be relatively sensitive to the value of trim drag, it was thought advisable to investigate directly the sensitivity of the combat load factor to the value of the elevator pitching moment effectiveness. The curve shown in Figure 11 is based on the present analysis values with the elevator pitching moment effectiveness as a variable. It will be observed that the relative combat effectiveness varies between 41 percent and 94 percent within the range of elevator pitching moment effectiveness values considered possible at the present time. The value of the pitching moment effectiveness obtained on the basis of the experimental data of Reference 31 yields a relative combat effectiveness of 68 percent.

It is interesting to note that if the difference in combat load factor, resulting from the difference between the N.A.E. and the A.V. Roe estimate of elevator pitching moment effectiveness (Figure 11), is added to the combat load factor obtained using the A.V. Roe profile drag estimate (Figure 10), the resulting combat load factor is 2.12. This agrees with the value of 2.14 presented in Reference 3.

Since the R.C.A.F. combat performance specification indirectly calls for a given minimum radius of turn and time to turn at combat speed, (refer to Equations 6 and 7,) it is found advisable to investigate whether or not the increase in load factor with Mach numbers above M = 1.5 will result in more favourable values of these quantities at higher Mach numbers than the (indirectly) specified values at M = 1.5. It is believed that the actual radius of turn and/or time to turn are more important quantities, from a taçtical point of view, than the load factor itself.

The variations with flight Mach number of the minimum radius of sustained level turn and the minimum time to complete a 360 degree level turn, corresponding to the load factor variation given by the solid curve in Figure 11, are given in Figure 12. In spite of the increasing value of load factor with Mach number above 1.1, both the radius of turn and time to turn increase steadily with Mach number above the drag rise value. The C-105/1200 aircraft thus fails to meet the R.C.A.F. specification in this respect as well, and the specified values of radius of turn and time to turn can only be obtained by decreasing the combat speed considerably. It is of some interest to note that the optimum values of both radius of turn and time to turn occur at the drag rise Mach number.

APPENDIX C

COMBAT CLIMB, ACCELERATION TIME AND CEILING

Paragraphs 3.05 and 3.08 of Reference 1 read:
"3.05 Combat Climb and Acceleration Time.
3.05.01 The aircraft shall reach combat speed of Mach 1.5 in straight and level flight at 50,000 feet from a position of rest at sea level in not more than 6 minutes.

3.08 Ceiling
3.08.01 The combat ceiling shall not be less than 60,000 feet."

Rate of Climb

The rate of climb for a constant Mach number is given by:

$$R/C = \frac{30V(1-0.1330M^2)}{d_1'}\left[1 - \sqrt{1 - \frac{4d_1'}{(1-0.1330M^2)}\left(\frac{T}{W} - d_o' - d_1' + K_1\sqrt{1-\sin^2\theta}\right)}\right] \quad (8)$$

where $d_o' = \frac{qS}{W} C_{D_o}'$ and $d_1' = \frac{K_2 W}{qS}$

(C_{D_o}', K_1 and K_2 are as defined in Appendix B)

Equation (8) includes the appropriate trim drag and the reduction in induced drag due to the angle of climb, but neglects the inclination of the thrust axis to the flight path and the normal acceleration due to the flight path. Furthermore, an approximate angle of climb must be assumed initially to take account of the trim drag relief factor K_1. An initial assumption within ± 10 degrees yields, in general, an answer within 1 percent. It should be noted that the term $(1-0.1330M^2)$ drops out of Equation (8) above the tropopause. This term results from the favourable tangential acceleration due to the speed variation with altitude in a constant Mach number climb within the troposphere.

The variations of constant Mach number rates of climb with altitude are shown in Figure 13. The calculations are based on the "present analysis" values of the various coefficients shown in Figures 1 to 9. In addition, the rates of climb for M = 0.95 within the troposphere, and for M = 1.50 above the tropopause, are worked out assuming the estimated A.V. Roe profile drag values.

The following interesting points will be noted:

(i) The C-105/1200 aircraft is just capable of vertical climb at M = 0.95 at sea level.

(ii) The rates of climb at combat speed are considerably better than at the drag rise speed above 12,000 feet altitude.

(iii) The favourable effect of the tangential acceleration in a constant supersonic Mach number climb is large. The calculated rate of climb, including this effect, is 44 percent above the calculated rate of climb when the acceleration effect is neglected for a Mach number of 1.5 at the tropopause.

(iv) The difference between the calculated rates of climb for the two profile drag estimates is negligible for M = 0.95 within the troposphere. However, the rates of climb pertaining to the A.V. Roe profile drag estimate are considerably better than the values obtained with the N.A.E. estimate at M = 1.5 above the tropopause.

Time, Horizontal Distance and Fuel to Combat Height and Speed

It is clear that the time to height is less if the acceleration from the drag rise Mach number to the combat Mach number is undertaken at an altitude of, say, 20,000 feet rather than at the tropopause as assumed in previous calculations (References 4, 5 and 6) since the rates of climb at combat speed are considerably better than at the drag rise speed above 12,000 feet. The difference is appreciable for the C-105/1200 aircraft since it can also be shown that the required acceleration time is less at the lower altitude.

No attempt has been made in the present analysis to find the minimum time to combat height and speed. It is shown later that, in any event, the aircraft is well within the R.C.A.F. specification in this respect. It appears, therefore, that the climb should be made in such a way that a minimum amount of fuel is consumed in reaching combat height and speed, provided the resulting time to height for this flight plan is within the specification.

Since part of the climb fuel is used to cover horizontal distance as well as to gain height, and since this distance effectively is part of the radius of action, we can define:

Effective Climb Fuel = Total fuel to height - Fuel required to cover the same horizontal distance at combat height and speed.

In other words, this means that although the aircraft may use more fuel to combat height and speed in a flat climb at high speed than in a steep climb at low speed, it also covers more horizontal distance. The amount of fuel used at combat height and speed to cover this difference in distance must therefore be subtracted from the total fuel used during the first climb procedure in order to obtain a true comparison with the fuel used during the second climb procedure.

Similarly, we can define:

Net Acceleration Fuel = Total acceleration fuel - Fuel required to cover the same horizontal distance at combat height and speed.

The variations of the effective climb fuel per 1000 feet and the net acceleration fuel with altitude are shown in Figure 14. It is noted that the effective climb fuel at $M = 0.95$ is less than at $M = 1.5$ below 33,000 feet and between 36,000 and 37,000 feet altitude. (The double intercept results from the effect of the tangential acceleration on the constant Mach number rates of climb below the tropopause.) The minimum net acceleration fuel required to accelerate from the drag rise Mach number to the combat speed occurs just above 36,000 feet. It can be shown that a constant Mach number climb at Mach numbers less than $M = 0.95$ yields higher values of effective climb fuel even at quite low altitudes. There are indications, however, that a slight improvement in fuel consumption may result by an accelerated climb from, say M = 0.95 at 30,000 feet to M = 1.50 at 40,000 feet. It is believed that this improvement, if any, is extremely small.

These calculations indicate, then, that a drag rise Mach number climb from sea level to 36,000 feet altitude and acceleration from M = 0.95 to M = 1.50 at the tropopause followed by a constant combat Mach number climb to combat height yields closely the minimum effective fuel consumption to combat height and speed. It should be remembered, however, that only maximum reheated thrust is considered in the above considerations. It is felt that any gains that may be had by employing part reheated thrust are small within the limitation imposed by the specified time to combat height and speed.

Figure 15 shows the time to height versus altitude assuming the above mentioned flight plan. In addition, the time to height for acceleration from M = 0.95 to M = 1.5 at 20,000 feet altitude is calculated to show that the above flight plan is not the optimum one with respect to time to height. It will, however, be noted that both flight plans yield values of time to combat height and speed considerably below the 6 minutes specified by the R.C.A.F. The differences in time to height between the calculations pertaining to the two profile drag estimates are similar to the differences in the rates of climb.

The variations of horizontal distance and fuel consumptions with altitude for the above mentioned flight plan are shown in Figures 16 and 17 respectively.

By comparing the end values of Figures 15, 16 and 17 with the corresponding values given in Table 5 of Reference 3, the following differences will be noted:

(i) The time to combat height and speed presented in Reference 3 is 4.2 percent less than that obtained in the present analysis with the N.A.E. profile drag estimate, but 6.3 percent greater than that obtained with the A.V. Roe profile drag estimate. Since, however, the time quoted in Reference 3 is 11.5 percent above the present value for the alternative flight plan, it is believed that Reference 3 employed a flight plan similar to the basic one assumed in this analysis.

(ii) The horizontal distance to combat height and speed presented in Reference 3 is 22 percent greater than the value obtained in the present analysis with the N.A.E. profile drag estimate and 40 percent greater than that obtained with the A.V. Roe estimate. It can be shown that the horizontal distance is less for the alternative flight plan than for the basic one. No explanation of these large differences between the present estimates and the A.V. Roe estimate has been found to date.

(iii) The fuel to combat height and speed presented in Reference 3 is 7.9 percent less than that obtained in the present analysis with the N.A.E. profile drag estimate and only ½ percent less than the value obtained with the A.V. Roe estimate.

Combat Ceiling

Reference 1 defines the combat ceiling as the altitude where the sustained rate of climb has fallen to 500 feet per minute.

The variation of combat ceiling with Mach number is shown in Figure 18. The optimum value occurs at a Mach number of 1.9 and is about 65,000 feet. The values obtained at supersonic speeds are slightly below those quoted in Reference 3, but well above the value specified by the R.C.A.F. The combat ceiling is just over 60,000 feet at the drag rise Mach number according to the present analysis. This is slightly more favourable than the value obtained in Reference 3, probably as a result of the higher drag efficiency obtained in the present analysis at this Mach number.

APPENDIX D

COMBAT RADIUS OF ACTION

Paragraph 3.06.01.01 of Reference 1 reads:
"3.06.01.01 Combat Radius of Action. The combat radius of action shall be 200 nautical miles. It shall be based on the following mission:

3.06.01.01.01 Taxi and warm-up: 4 minutes.

3.06.01.01.02 Combat climb to 50,000 feet and acceleration to combat speed of Mach 1.5:6 minutes.

3.06.01.01.03 Cruise out at combat speed of Mach 1.5 at 50,000 feet altitude to a radius of action of 200 nautical miles from base.

3.06.01.01.04 Combat under combat performance conditions for 5 minutes.

3.06.01.01.05 Return to base at economical cruising speed.

3.06.01.01.06 Loiter above 30,000 feet for 15 minutes.

3.06.01.01.07 Descend to sea level.

3.06.01.01.08 Land with 5 minutes sea level loiter reserve."

Taxi and Warm-up

The four minutes taxi and warm-up fuel is assumed to be equivalent to the fuel consumed by one engine at maximum continuous thrust, reheat off, at standstill at sea level. The resulting fuel consumption is 663 pounds which agrees well with the value presented in Table 5 of Reference 3.

Combat Climb and Acceleration

This part of the combat radius of action is already discussed in detail in Appendix C. The fuel required (excluding taxi and warm-up) is 4,114 pounds with the N.A.E. profile drag estimate and 3,757 pounds with the A.V. Roe estimate for the flight plan assumed in the present analysis. The corresponding values of horizontal distance covered are 32 nautical miles and 27.9 miles.

Cruise Out at Combat Speed

Since only part reheat is required during the cruise, the degree of reheat and the corresponding specific fuel consumption must be determined. Figure 7 of Reference 7 gives, fortunately, the estimated performance of the R.B. 106 for varying degrees of reheat at the tropopause for the combat Mach number. Corresponding values can be obtained at 50,000 feet altitude if the following simplified reasoning is used:

The ratio of the maximum unreheated thrust at $M = 1.5$ and 36,000 feet to the maximum unreheated thrust at $M = 1.5$ and 50,000 feet is the same as the ratio of the maximum reheated thrusts at $M = 1.5$ at the two altitudes. (Refer to Figures 5 and 6 of Reference 7). It can thus be assumed that the same degree of reheat is required at both altitudes to produce a given percentage of the respective maximum reheated thrusts. The specific fuel consumption without reheat is the same at both altitudes for any given Mach number. However, the specific fuel consumption at 50,000 feet is higher than at 36,000 feet with maximum reheat, and the difference depends on the Mach number. The difference is only about 6 percent at $M = 1.5$ and it is therefore assumed, in the present analysis, that this difference varies linearly with reheat temperature. The specific fuel consumption for the combat cruise can thus be determined by:

(i) Calculating the percentage of the maximum reheated thrust required.

(ii) Finding the reheat temperature required and the corresponding specific fuel consumption at 36,000 feet from Figure 7 of Reference 7.

(iii) Multiplying the difference in the maximum reheat specific fuel consumptions at the two altitudes by the required reheat temperature ratio and adding this product to the specific fuel consumption required at 36,000 feet.

The variation of the thrust required, resulting from variations in the aircraft weight during the combat cruise, is taken into account in the calculations by the use of several steps.

The total fuel required for the combat cruise is 3,180 pounds with the N.A.E. profile drag estimate and 2,425 pounds with the A.V. Roe profile drag value. It is probably accidental that the ratio of these values is closely the same as the ratio of the profile drag values. The value presented in Reference 3 (2,230 pounds) is less than the values obtained in the present analysis. However, the average fuel consumption per nautical mile is 13.85 pounds from Reference 3 and 14.1 pounds for the A.V. Roe profile drag value in the present analysis.

Combat

The fuel required for combat is 3,265 pounds, namely the fuel consumed by two engines in 5 minutes at maximum reheated thrust at combat height and speed. It is apparent that the A.V. Roe Company does not consider it necessary to employ maximum thrust during combat since the value quoted in Table 5 of Reference 3 is only 3,050 pounds.

Return to Base

The optimum cruise condition occurs when the fuel flow per nautical mile is a minimum.

Figure 19c shows the variation with Mach number of the minimum cruise fuel flow and Figure 19a the corresponding altitude for minimum fuel flow. (The minimum fuel flow for any given Mach number occurs at a specific altitude, that is to say, the altitude varies along the two fuel flow curves shown in Figures 19b and c in accordance with the curve plotted in Figure 19a). It is assumed in these calculations that the specific fuel consumption is independent of altitude above 36,000 feet and that the percentage variation of specific fuel consumption with the ratio of actual thrust to maximum thrust, reheat off, is the same as that obtained from Figure 2 of Reference 7. The specific fuel consumption at altitudes below 36,000 feet is obtained by plotting specific fuel consumption versus altitude for constant Mach numbers from Figure 6 of Reference 7 and using Figure 2 of Reference 7 as mentioned above. Figure 19 applies to the "present analysis" values shown in Figures 1 to 9.

It will be seen from Figure 19 that the optimum cruise condition occurs at the drag rise Mach number at an altitude of 46,900 feet. The fuel flow is 7.85 pounds per nautical mile. The corresponding values with the A.V. Roe profile drag estimate (not shown in Figure 19) are a minimum fuel flow at 7.49 pounds per nautical mile at M = 0.95 and at an altitude of 45,900 feet. Both these fuel flow values are considerably below the value of 8.73 pounds per nautical mile (935 pounds in 107 miles) presented in Reference 3 where the economical cruise is assumed to be at 35,000 feet altitude. However, Reference 3 assumes that the loitering can be performed 64 miles away from the base and that this remaining distance can be covered during the descent to sea level. It is felt that paragraph 3.06.01.01.05 of Reference 1: "Return to base at economical cruising speed", can only be interpreted to mean that the full return radius of action, that is 200 nautical miles, <u>must</u> be covered under this heading. The fuel required for return to base is thus 1,570 pounds with the N.A.E. profile drag estimate and 1,498 pounds with the A.V. Roe profile drag value. The effects on the fuel consumption of the deceleration from M = 1.5 to M = 0.95 and the descent from 50,000 feet to the optimum cruising altitude are probably small and are neglected in the present analysis.

<u>Loiter</u>

The optimum loiter condition occurs when the fuel flow per second is a minimum.

Figure 19b shows the variation of the minimum loiter fuel flow with Mach number for the corresponding altitude variation given in Figure 19a. The specific fuel consumption is determined as described in the previous paragraph. The optimum loiter condition occurs at a Mach number of 0.92 at 45,000 feet altitude where the fuel flow is 1.18 pounds per second for the N.A.E. profile drag estimate.

The specified minimum loiter altitude is, however, 30,000 feet and it appears reasonable to descend to this altitude during loiter. It will be observed from Figure 19 that the fuel flow is reasonably close to the optimum down to 30,000 feet provided the proper Mach number is maintained. It is assumed in the present analysis that the aircraft descends steadily from the cruising altitude to 30,000 feet during the loiter. This steady rate of descent decreases the required thrust by 900 pounds, and the resulting average fuel flow between these two altitudes is thus only 0.984 pounds per second for the N.A.E. profile drag estimate and 0.945 pounds per second for the A.V. Roe estimate. It is not understood how Reference 3 obtains a fuel flow of only 0.971 pounds per second in a level flight loiter at 35,000 feet.

The total fuel required for 15 minutes loiter is 885 pounds with the N.A.E. profile drag estimate and 850 pounds with the A.V. Roe estimate.

Descent to Sea Level

This part of the radius of action is rather indeterminate. However, it is supposed in the present analysis that the aircraft descends at a mean rate of sink of 5,000 feet per minute, that is, the total descent from 30,000 feet to sea level requires 6 minutes. The engines are effectively idling since no thrust is required. By plotting fuel flow versus thrust from Figure 2 of Reference 7 and extrapolating the curve to zero thrust, it will be found that the minimum fuel flow at sea level is about 3,200 pounds per hour for two engines. It is feld that it would be difficult to keep the engines running at altitude at lower fuel flow values, and this value is therefore assumed to prevail during the descent. The resulting descent fuel consumption is hence 320 pounds. This is less than one half the value quoted in Reference 3 where considerable distance is covered during the descent.

Sea Level Loiter Reserve

It can be shown that the Mach number for the most economical loiter condition at sea level is M = 0.344 for the N.A.E. profile drag estimate. The minimum fuel flow for this condition is found, by means of Figures 2 and 6 of Reference 7, to be 1.46 pounds per second and the required loiter reserve is hence 437 pounds. The corresponding value for the A.V. Roe profile drag estimate is 422 pounds. Both these values are considerably below the 535 pounds quoted in Reference 3.

FIG. 1

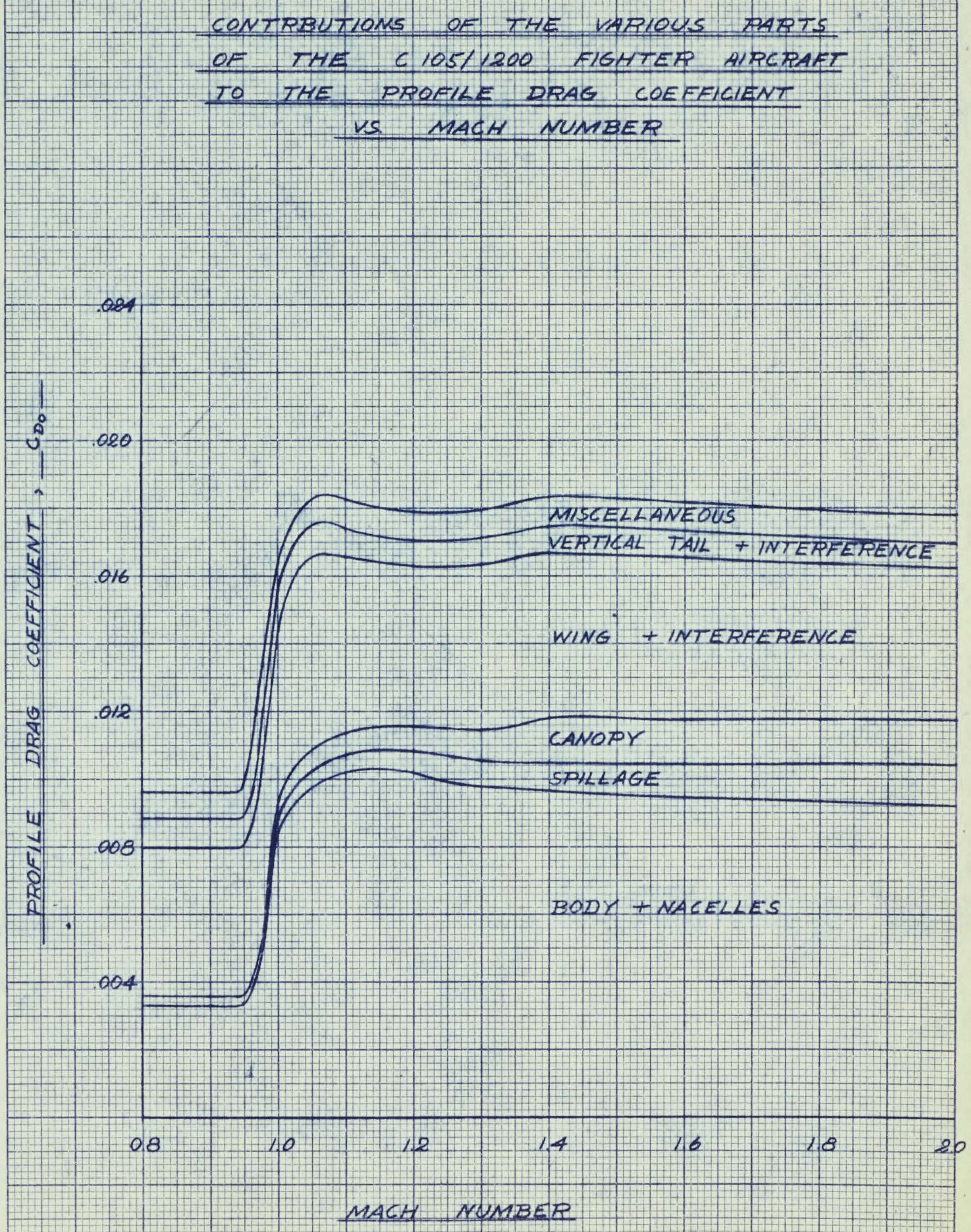

CONTRBUTIONS OF THE VARIOUS PARTS
OF THE C 105/1200 FIGHTER AIRCRAFT
TO THE PROFILE DRAG COEFFICIENT
VS. MACH NUMBER
PROFILE DRAG COEFFICIENT , C_{D_0}
.024
.020
.016
.012
.008
.004
MISCELLANEOUS
VERTICAL TAIL + INTERFERENCE
WING + INTERFERENCE
CANOPY
SPILLAGE
BODY + NACELLES
0.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

FIG. 2

COMPARISON OF PROFILE DRAG COEFFICIENTS.

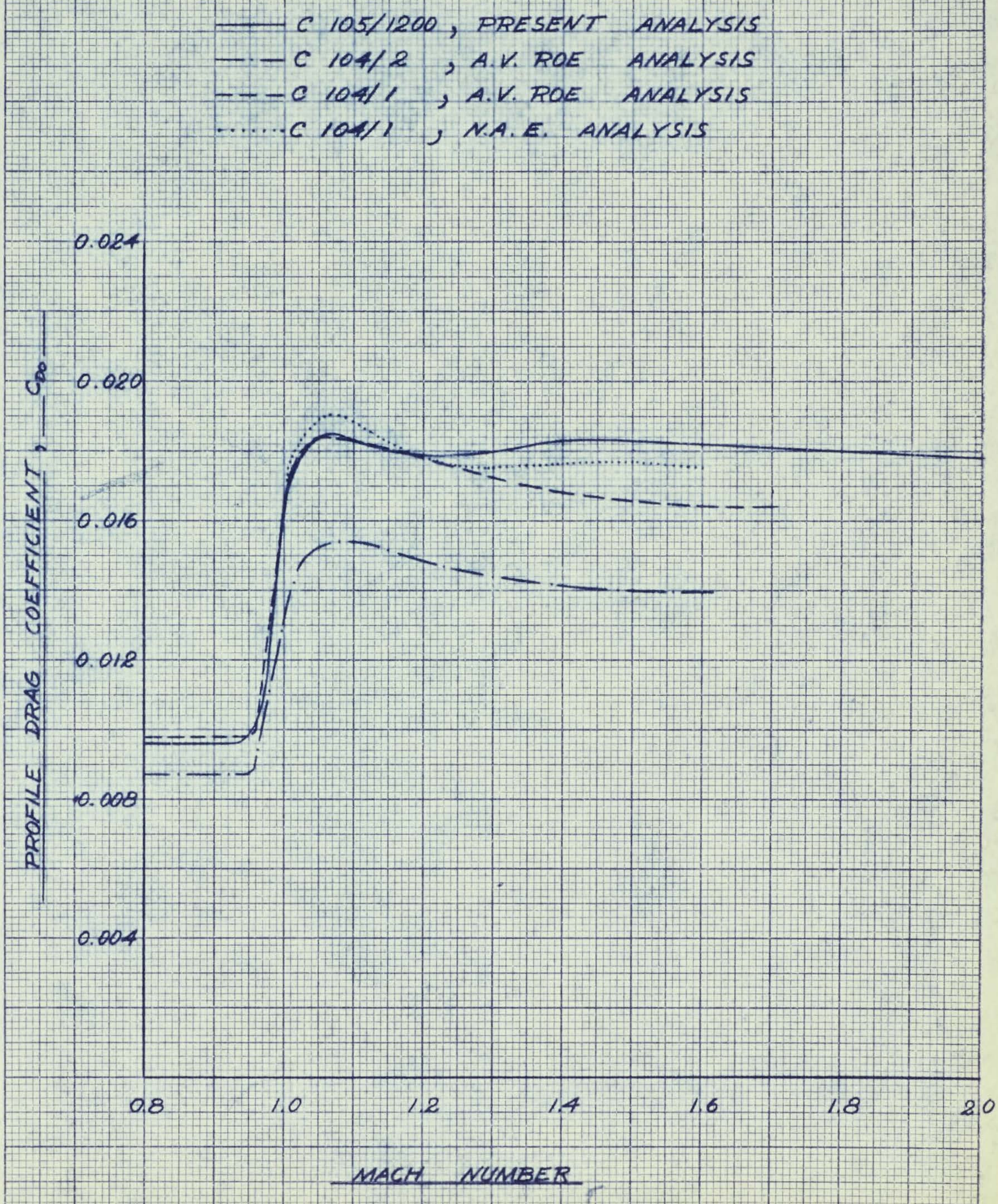


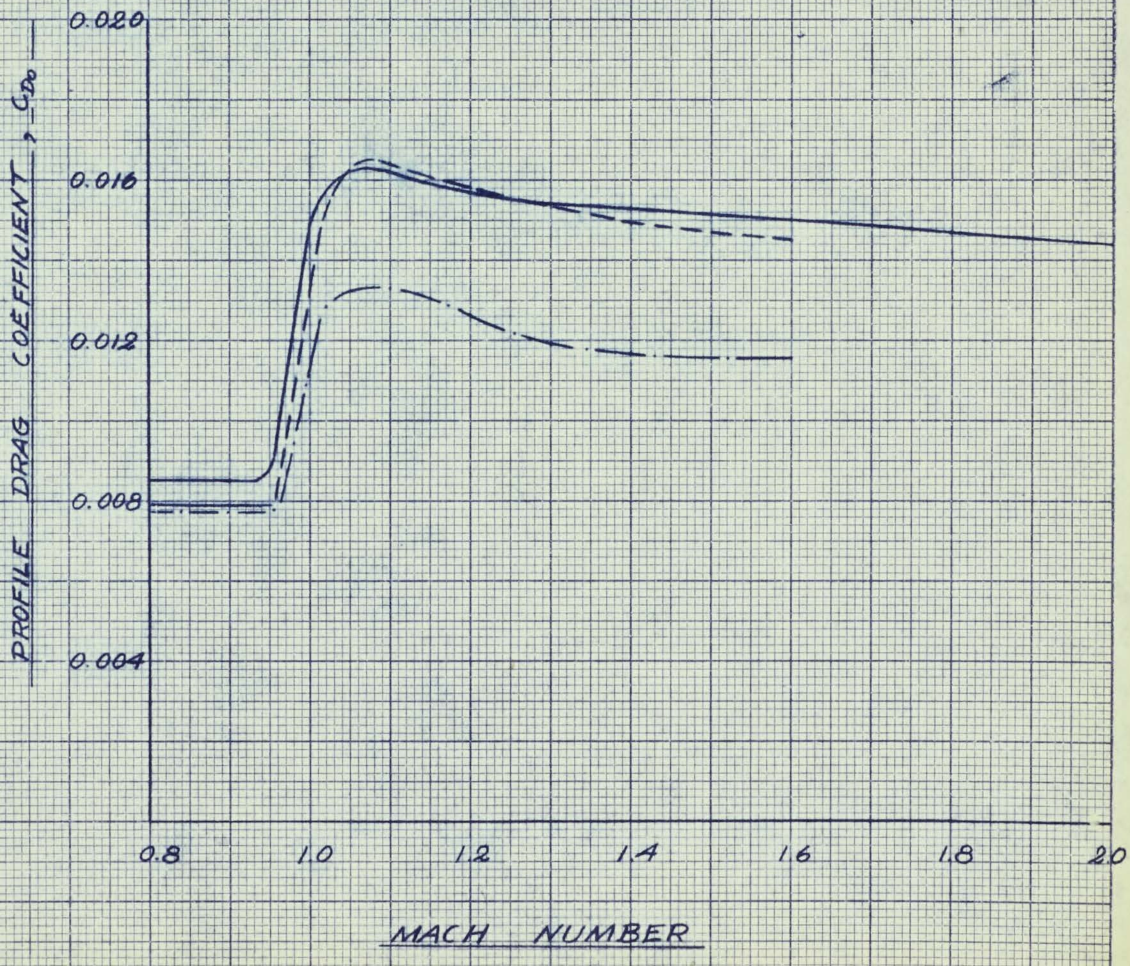
FIG. 3

COMPARISON OF PROFILE DRAG COEFFICIENTS OF WING - BODY - VERTICAL TAIL.

——— C 105/1200 , PRESENT ANALYSIS

—·— C 104/2 , A.V. ROE ANALYSIS

- - - - MEAN DRAG COEFFICIENT FOR THE TWO CONFIGURATIONS REPORTED IN REF. 17 , CORRECTED TO FUSELAGE FINENESS RATIO OF 8 AND C 105/1200 REYNOLDS NUMBER AT 40,000 FT. ALT.



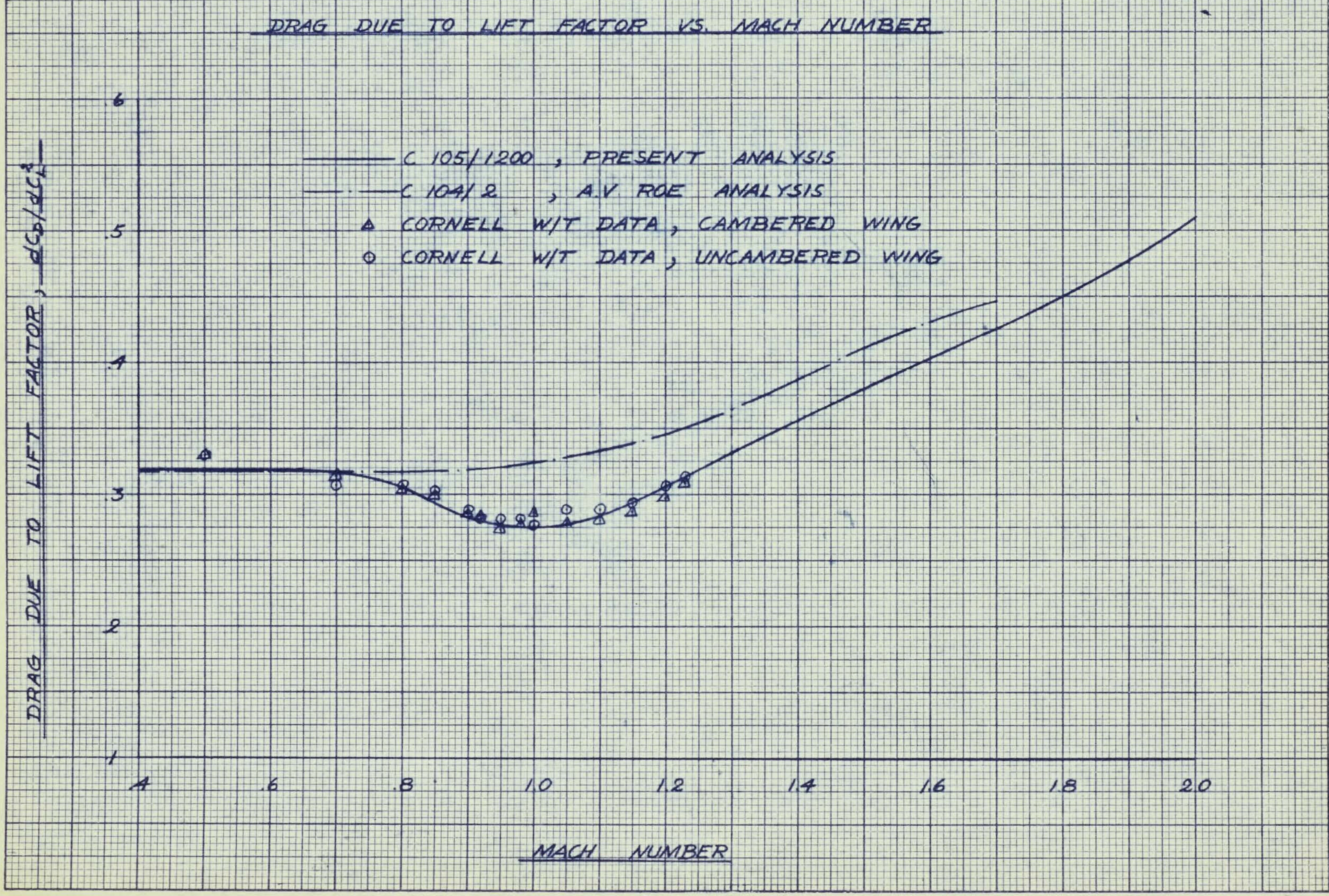

DRAG DUE TO LIFT FACTOR VS. MACH NUMBER
C 105/1200 , PRESENT ANALYSIS
C 104/2 , A.V ROE ANALYSIS
CORNELL W/T DATA , CAMBERED WING
CORNELL W/T DATA , UNCAMBERED WING
DRAG DUE TO LIFT FACTOR , dC_D/dC_L^2
.6
.5
.4
.3
.2
.1
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

FIG. 4

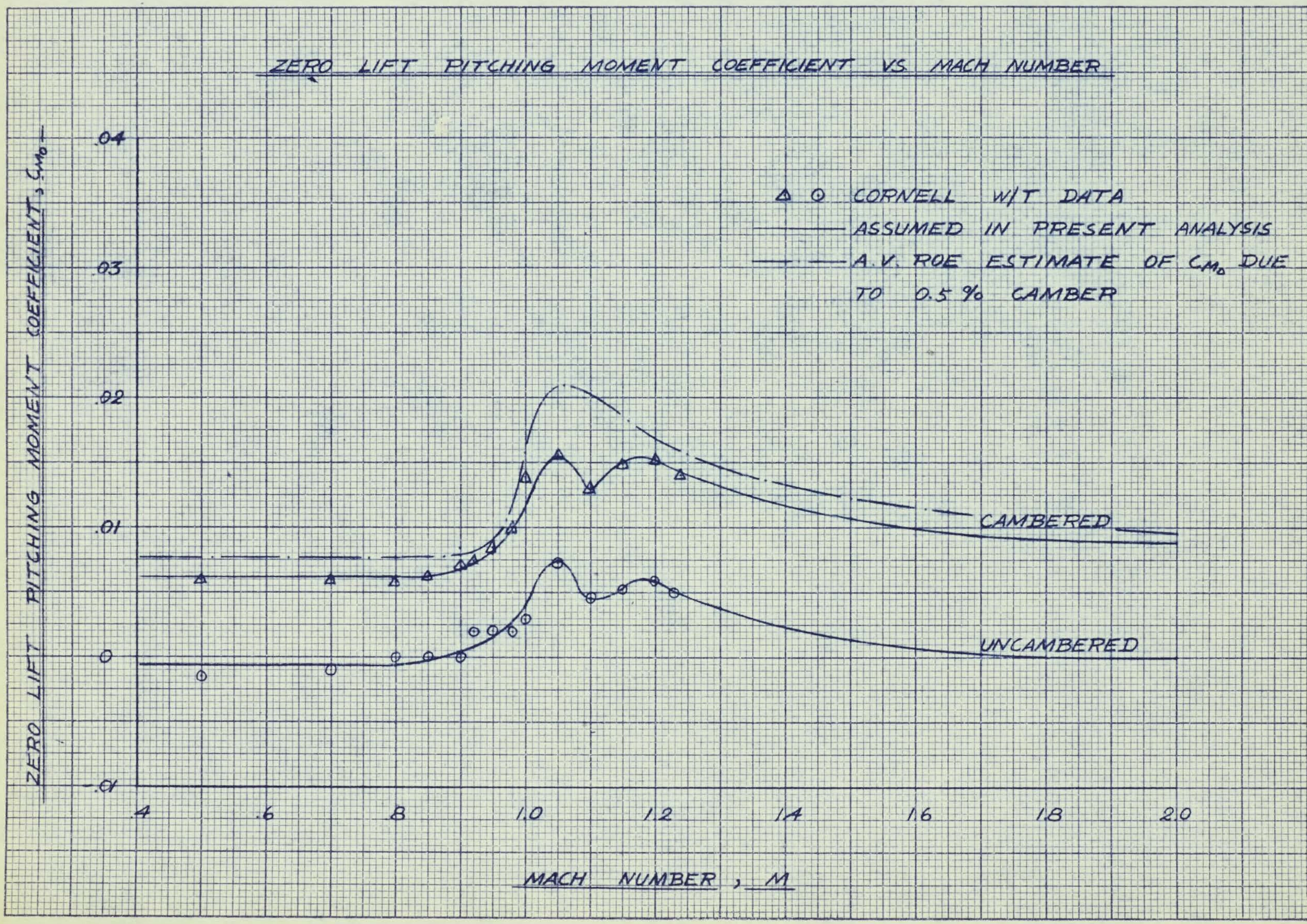
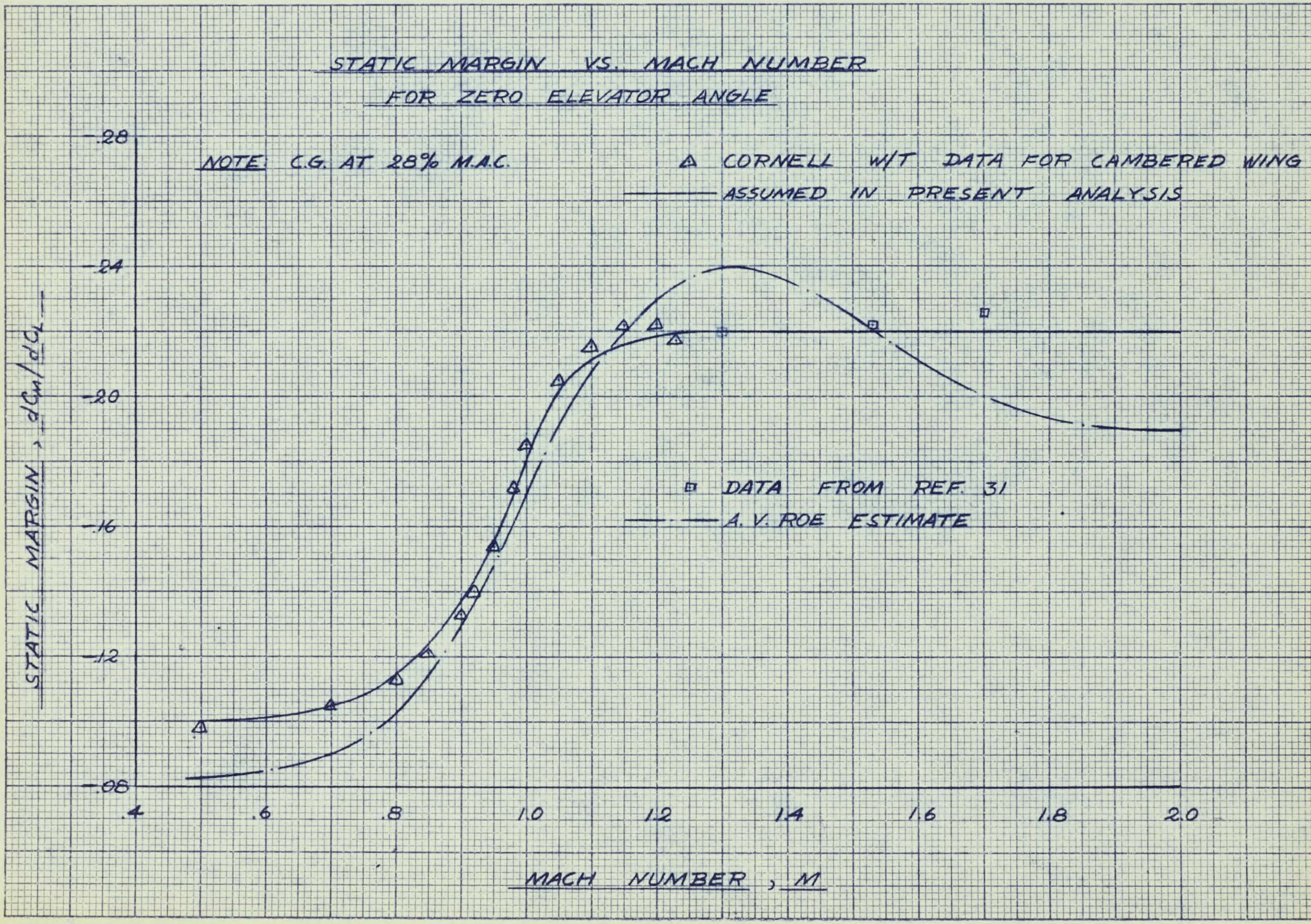
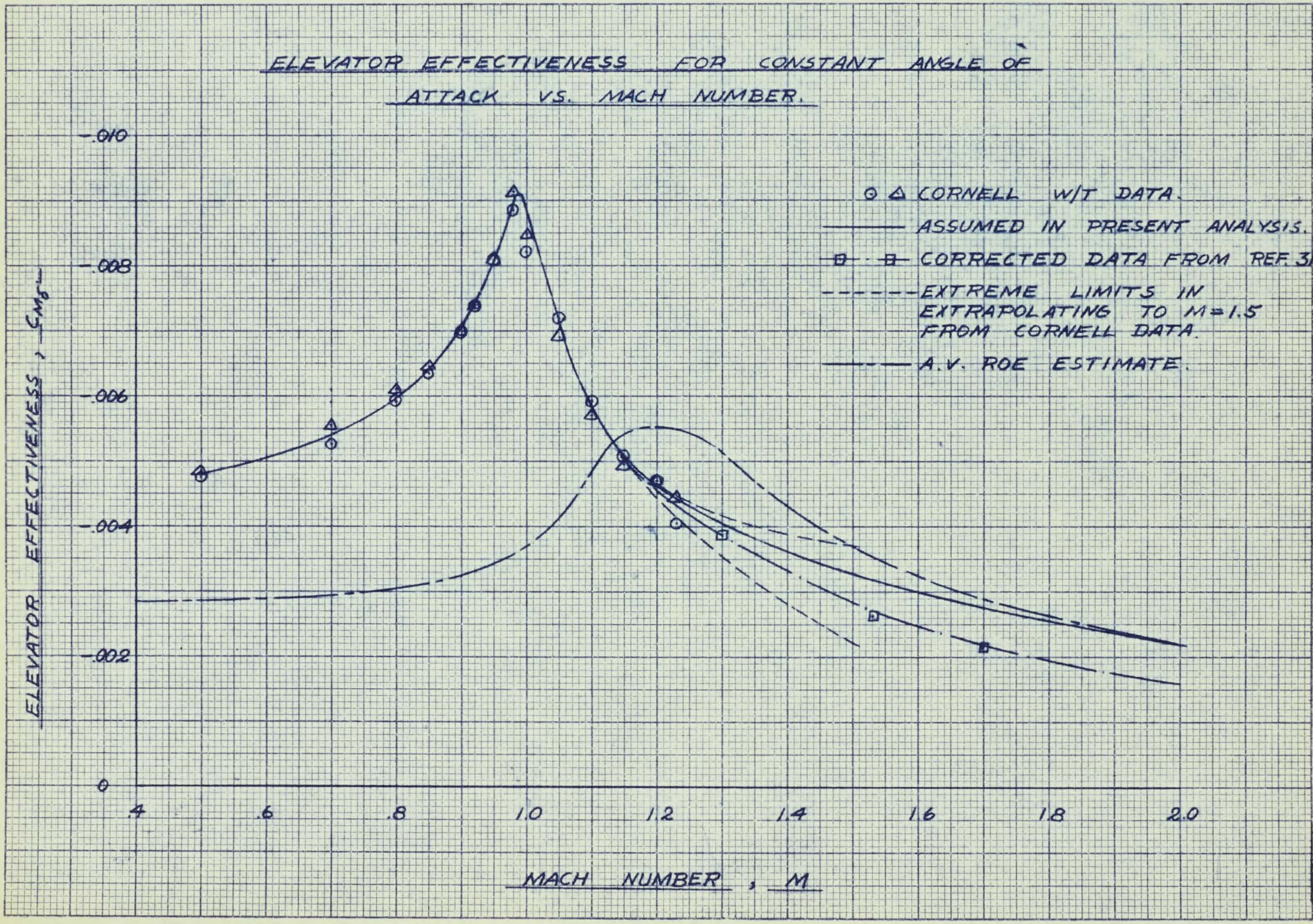

ZERO LIFT PITCHING MOMENT COEFFICIENT VS. MACH NUMBER
△ ⊙ CORNELL W/T DATA
ASSUMED IN PRESENT ANALYSIS
A.V. ROE ESTIMATE OF C_{M_0} DUE TO 0.5 % CAMBER
CAMBERED
UNCAMBERED
ZERO LIFT PITCHING MOMENT COEFFICIENT, C_{M_0}
.04
.03
.02
.01
0
-.01
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER, M

FIG. 5



STATIC MARGIN VS. MACH NUMBER
FOR ZERO ELEVATOR ANGLE
NOTE: C.G. AT 28% M.A.C.
CORNELL W/T DATA FOR CAMBERED WING
ASSUMED IN PRESENT ANALYSIS
DATA FROM REF. 31
A. V. ROE ESTIMATE
STATIC MARGIN , dC_M/dC_L
MACH NUMBER , M
.28
.24
.20
.16
.12
.08
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0

FIG. 6



ELEVATOR EFFECTIVENESS FOR CONSTANT ANGLE OF ATTACK VS. MACH NUMBER.
CORNELL W/T DATA.
ASSUMED IN PRESENT ANALYSIS.
CORRECTED DATA FROM REF. 31
EXTREME LIMITS IN EXTRAPOLATING TO M=1.5 FROM CORNELL DATA.
A.V. ROE ESTIMATE.
-.010
-.008
-.006
-.004
-.002
0
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
ELEVATOR EFFECTIVENESS, C_{M_δ}
MACH NUMBER, M

FIG. 7

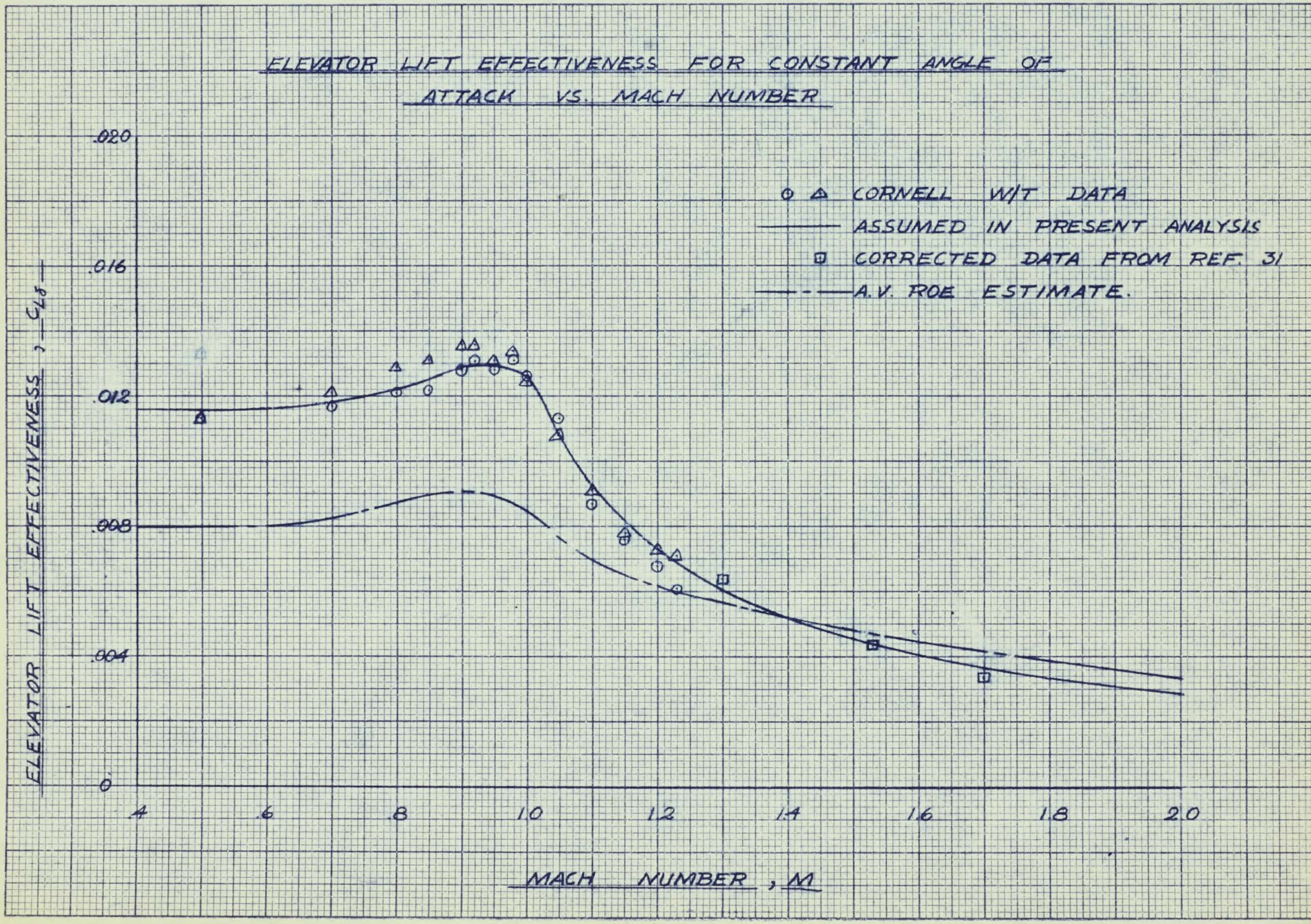

ELEVATOR LIFT EFFECTIVENESS FOR CONSTANT ANGLE OF ATTACK VS. MACH NUMBER
CORNELL W/T DATA
ASSUMED IN PRESENT ANALYSIS
CORRECTED DATA FROM REF. 31
A.V. ROE ESTIMATE.
ELEVATOR LIFT EFFECTIVENESS, $C_{L\delta}$
.020
.016
.012
.008
.004
0
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER, M

FIG. 8

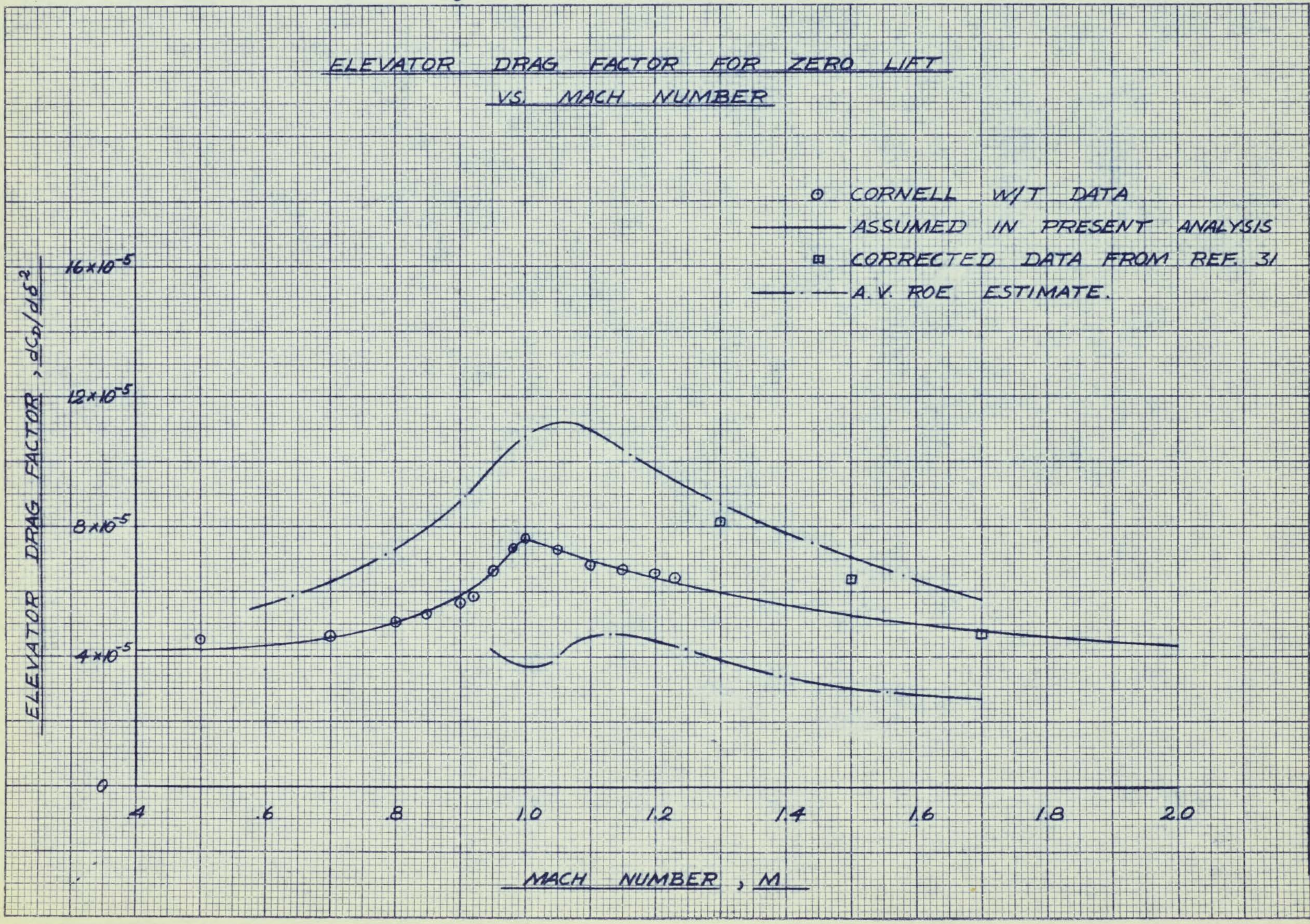

ELEVATOR DRAG FACTOR FOR ZERO LIFT
VS. MACH NUMBER
CORNELL W/T DATA
ASSUMED IN PRESENT ANALYSIS
CORRECTED DATA FROM REF. 31
A.V. ROE ESTIMATE.
ELEVATOR DRAG FACTOR , $dC_D/d\delta^2$
16×10^{-5}
12×10^{-5}
8×10^{-5}
4×10^{-5}
0
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER , M

FIG. 9

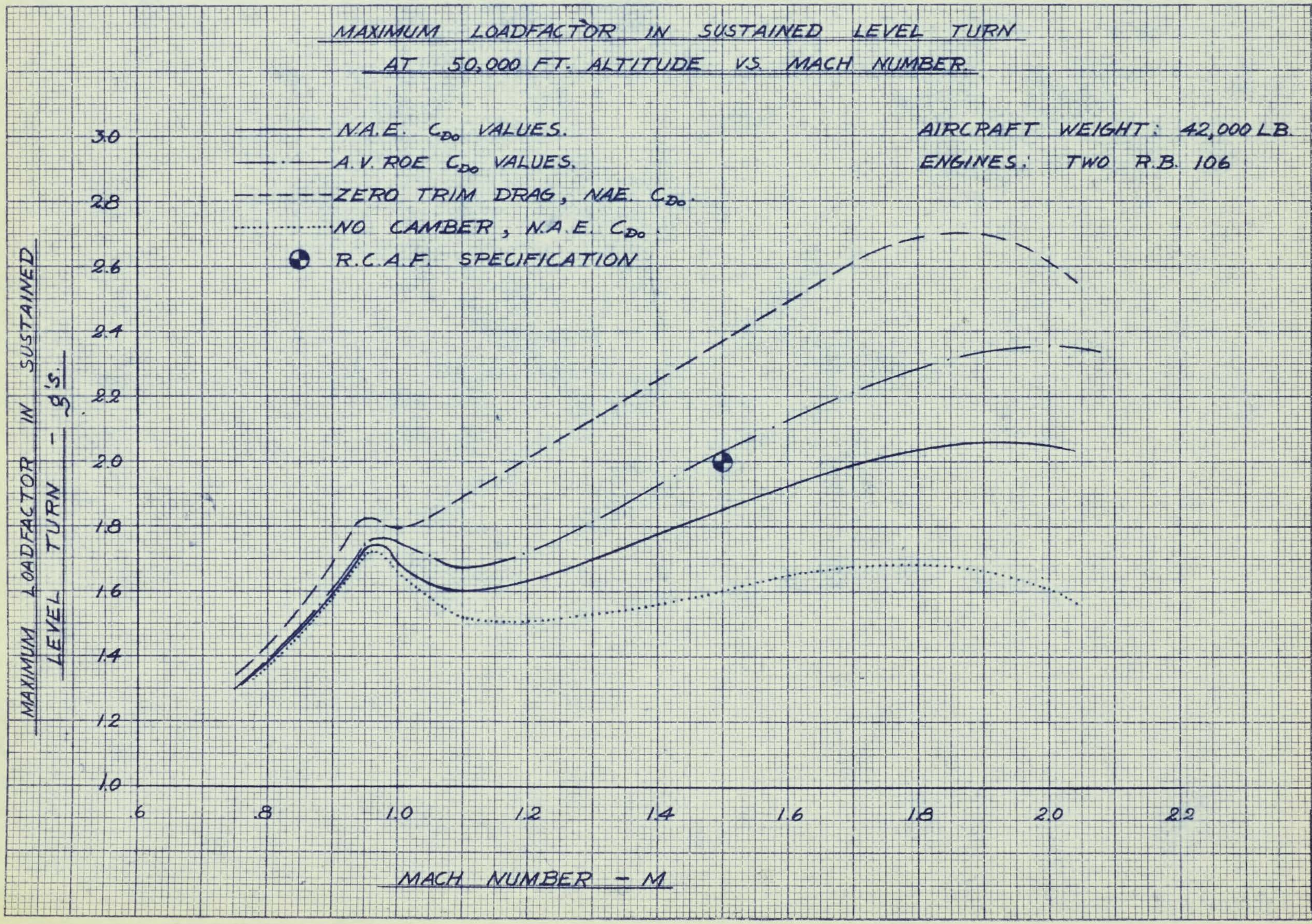
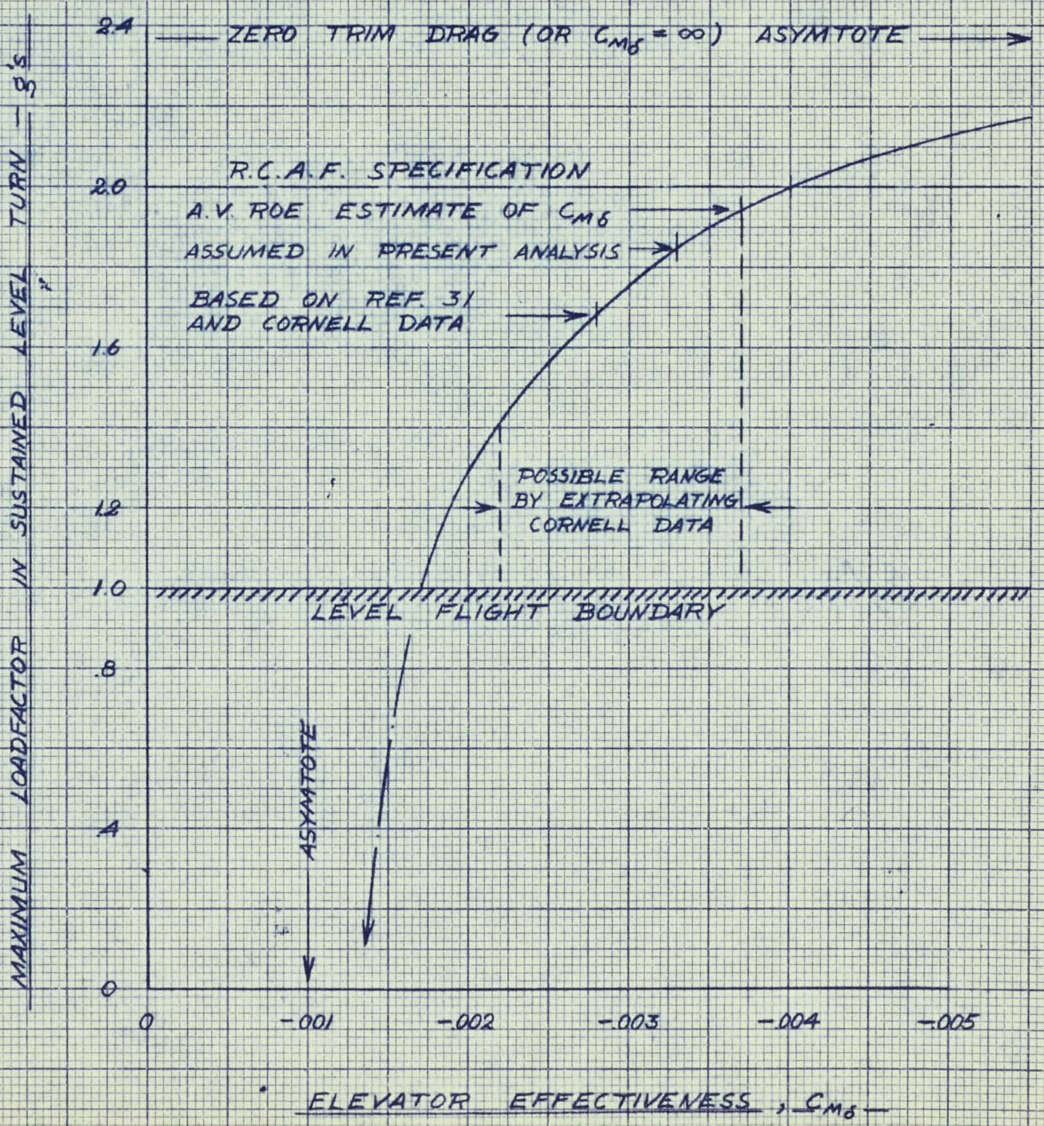

MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN
AT 50,000 FT. ALTITUDE VS. MACH NUMBER.
N.A.E. C_{Do} VALUES.
A.V. ROE C_{Do} VALUES.
ZERO TRIM DRAG, NAE. C_{Do}.
NO CAMBER, N.A.E. C_{Do}.
R.C.A.F. SPECIFICATION
AIRCRAFT WEIGHT: 42,000 LB.
ENGINES: TWO R.B. 106
MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN - g's.
3.0
2.8
2.6
2.4
2.2
2.0
1.8
1.6
1.4
1.2
1.0
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
MACH NUMBER - M
FIG. 10

FIG. 11

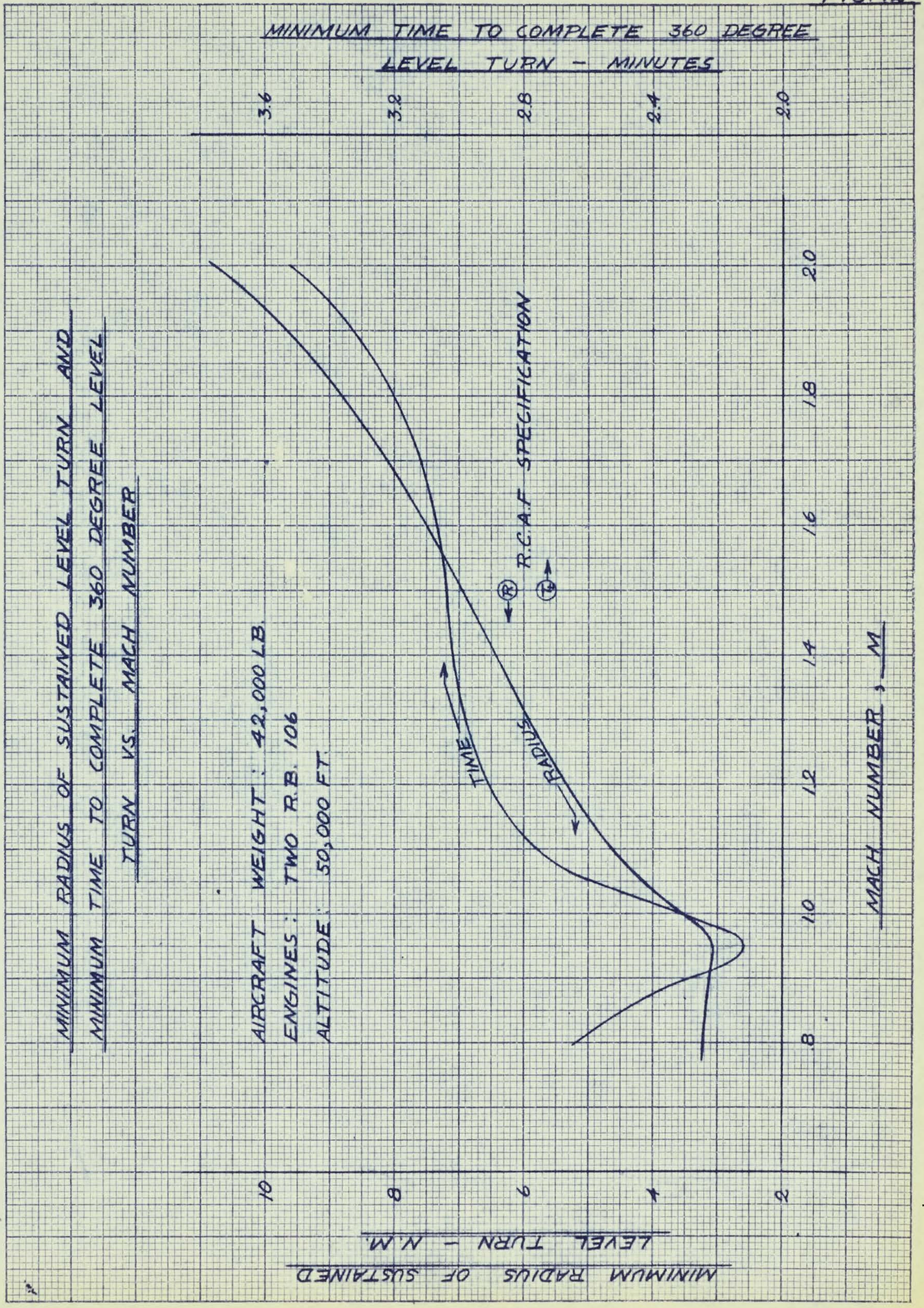
SENSITIVITY OF MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN AT M = 1.5 AND 50,000 FT. ALTITUDE TO VALUE OF ELEVATOR EFFECTIVENESS C_{M_δ}

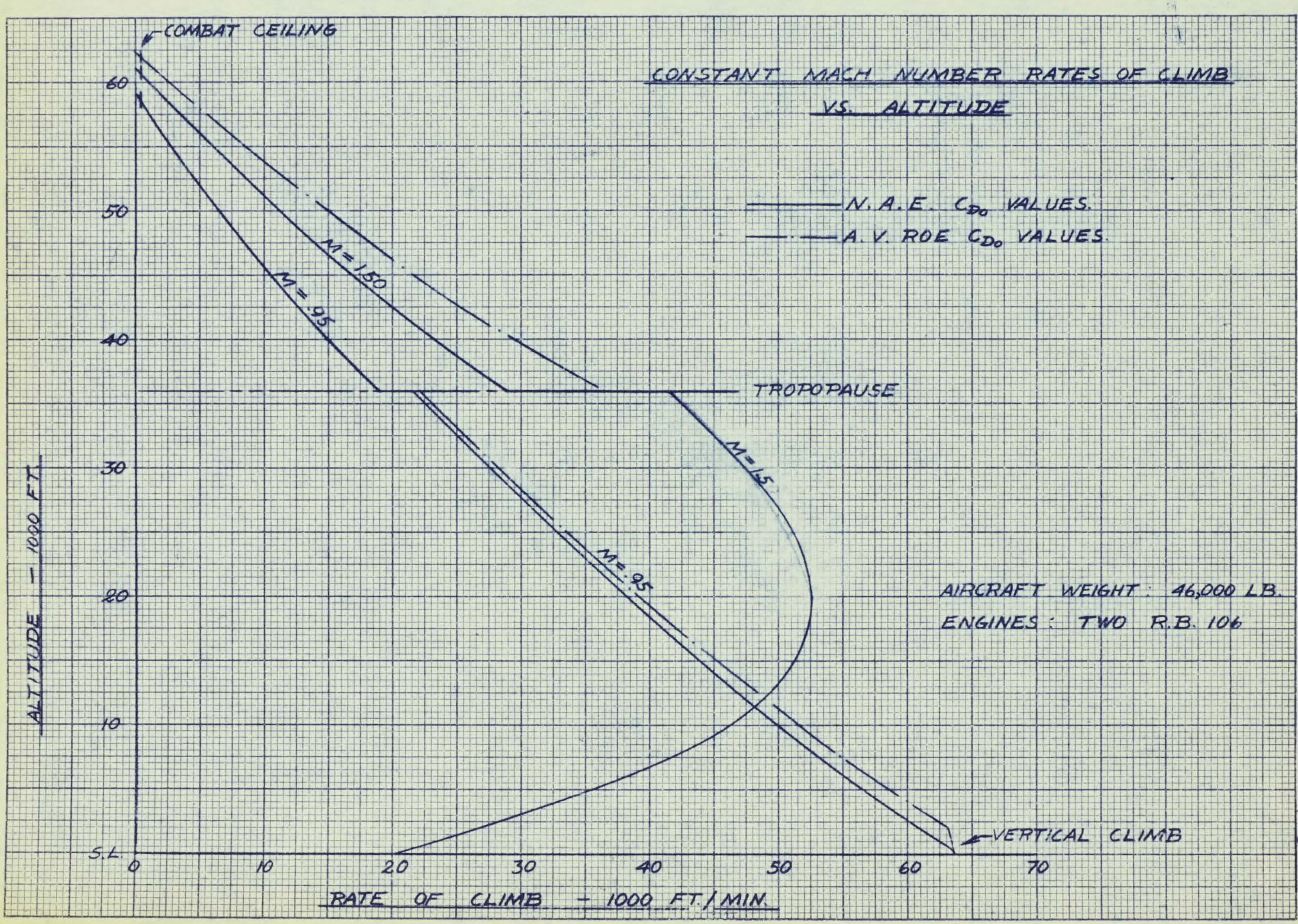
AIRCRAFT WEIGHT : 42,000LB.
ENGINES : TWO RB. 106



ELEVATOR EFFECTIVENESS , C_{M_δ}

FIG. 12


MINIMUM RADIUS OF SUSTAINED LEVEL TURN AND
MINIMUM TIME TO COMPLETE 360 DEGREE LEVEL
TURN VS. MACH NUMBER
AIRCRAFT WEIGHT: 42,000 LB.
ENGINES: TWO R.B. 106
ALTITUDE: 50,000 FT.
MINIMUM TIME TO COMPLETE 360 DEGREE
LEVEL TURN - MINUTES
3.6
3.2
2.8
2.4
2.0
MINIMUM RADIUS OF SUSTAINED
LEVEL TURN - N.M.
10
8
6
4
2
MACH NUMBER, M
.8
1.0
1.2
1.4
1.6
1.8
2.0
TIME
RADIUS
R.C.A.F SPECIFICATION
R
T



COMBAT CEILING
CONSTANT MACH NUMBER RATES OF CLIMB VS. ALTITUDE
N.A.E. C_{D_0} VALUES.
A.V. ROE C_{D_0} VALUES.
M = 1.50
M = .95
TROPOPAUSE
M = 1.5
M = .95
AIRCRAFT WEIGHT: 46,000 LB.
ENGINES: TWO R.B. 106
VERTICAL CLIMB
ALTITUDE — 1000 FT.
60
50
40
30
20
10
S.L.
0
10
20
30
40
50
60
70
RATE OF CLIMB — 1000 FT./MIN.

FIG. 13

FIG. 14

EFFECTIVE CLIMB FUEL AND NET ACCELERATION FUEL VS ALTITUDE

AIRCRAFT WEIGHT: 46,000 LB.
ENGINES: TWO R.B. 106

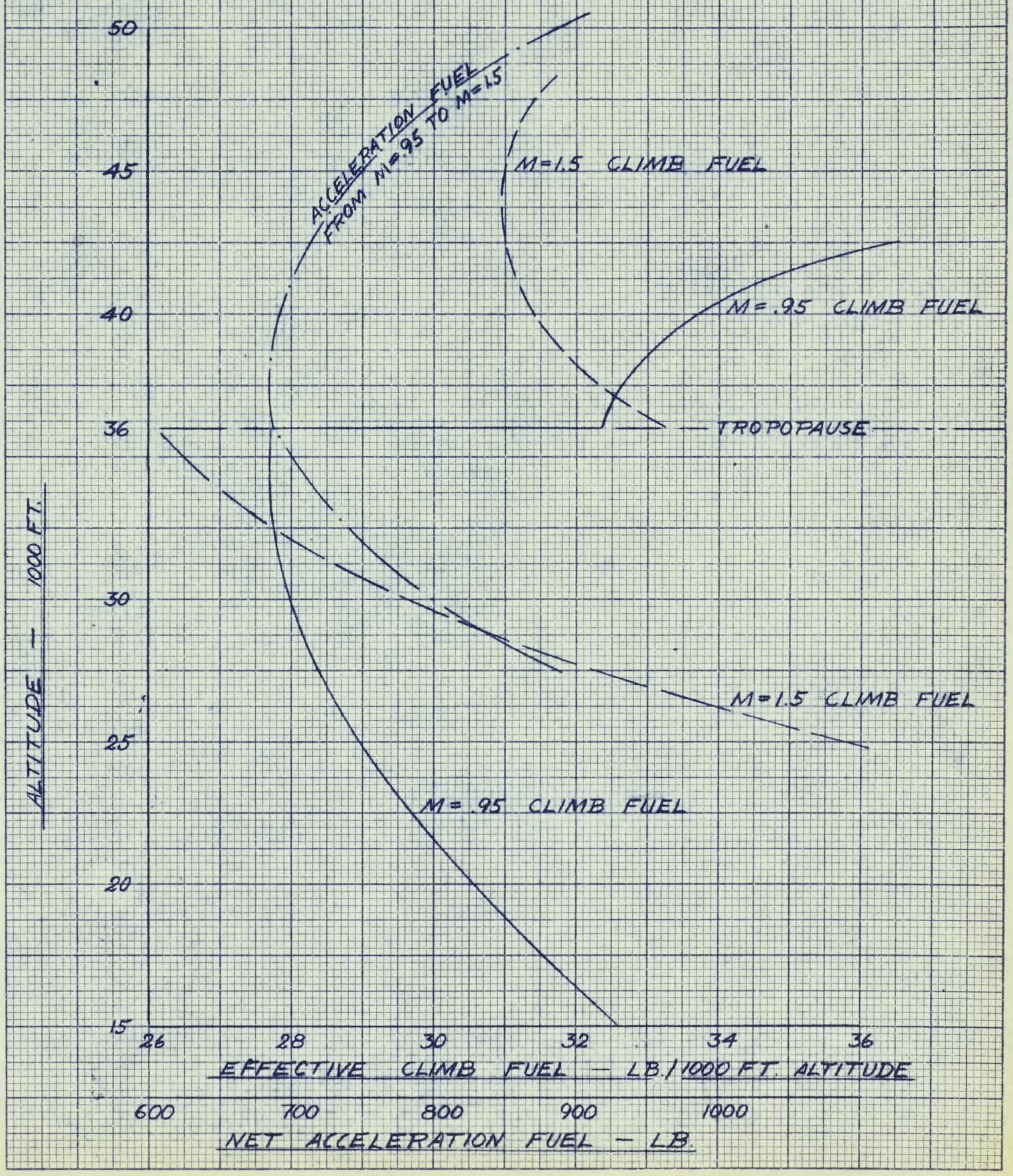
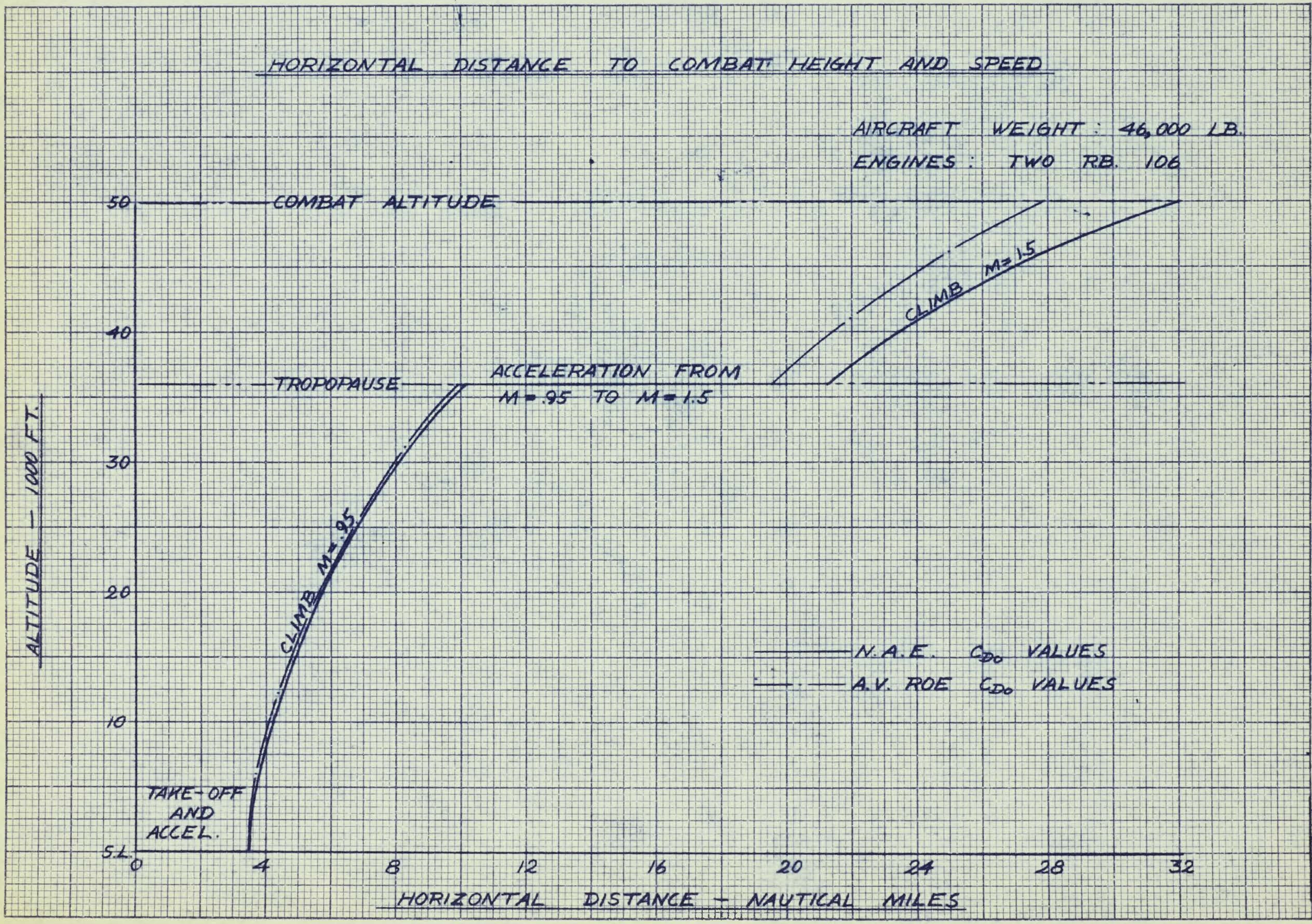


FIG. 16

HORIZONTAL DISTANCE TO COMBAT HEIGHT AND SPEED



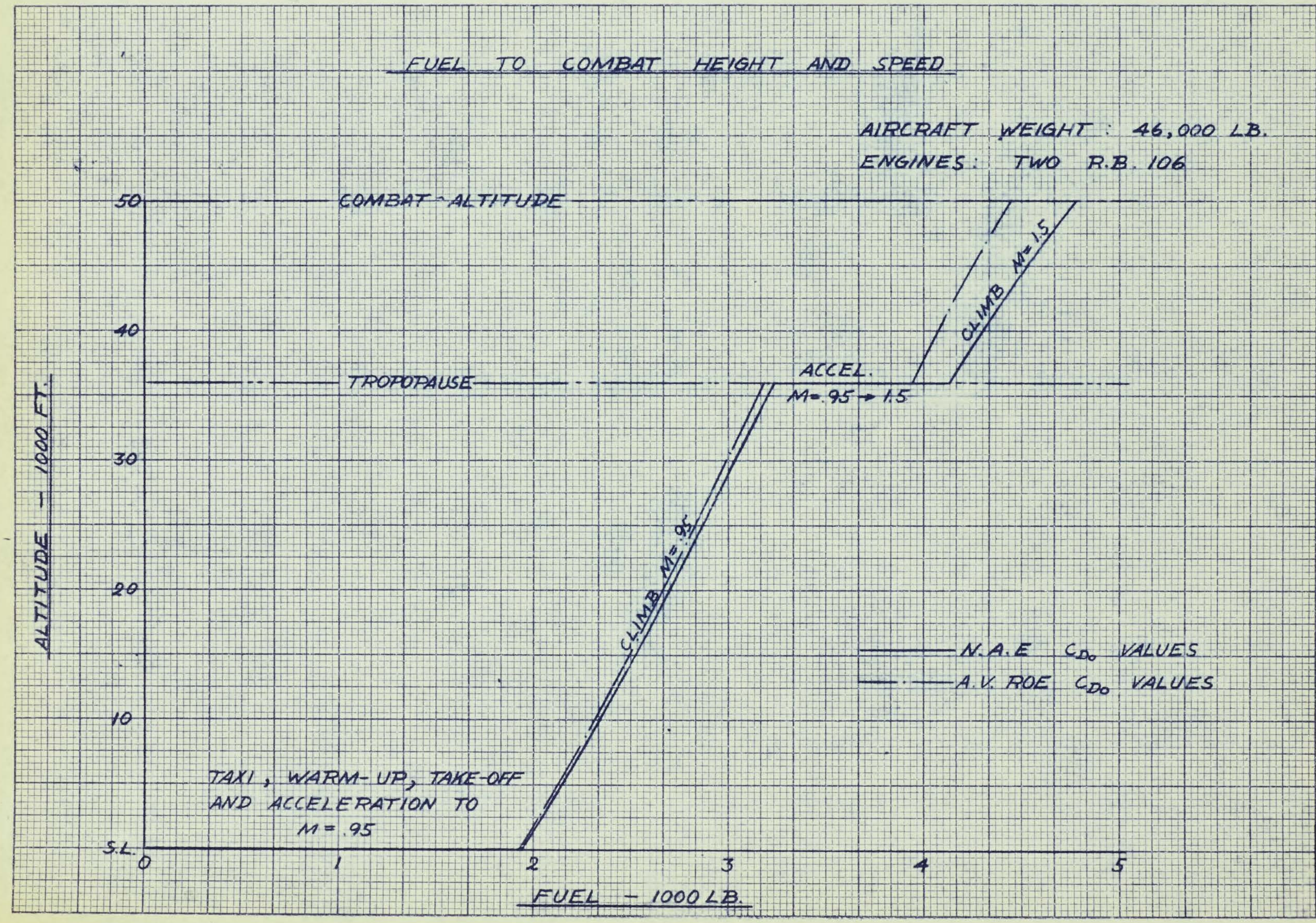
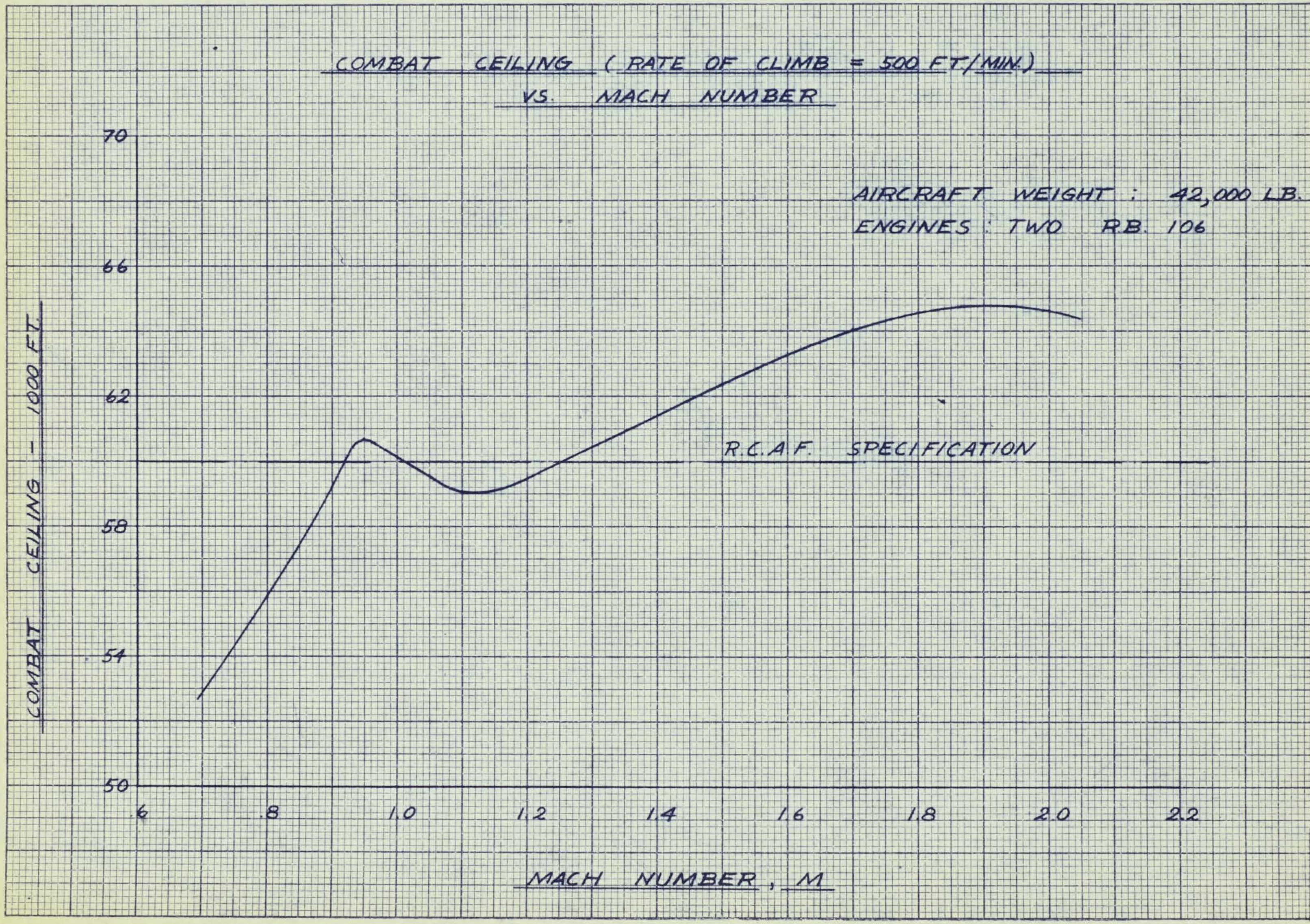

FUEL TO COMBAT HEIGHT AND SPEED
AIRCRAFT WEIGHT : 46,000 LB.
ENGINES: TWO R.B. 106
COMBAT ALTITUDE
TROPOPAUSE
ACCEL.
M = .95 → 1.5
CLIMB M = 1.5
CLIMB M = .95
TAXI, WARM-UP, TAKE-OFF
AND ACCELERATION TO
M = .95
N.A.E C_{D_0} VALUES
A.V. ROE C_{D_0} VALUES
ALTITUDE – 1000 FT.
FUEL – 1000 LB.
50
40
30
20
10
S.L.
0
1
2
3
4
5

FIG. 17

FIG. 18



COMBAT CEILING (RATE OF CLIMB = 500 FT./MIN.)
VS. MACH NUMBER
AIRCRAFT WEIGHT : 42,000 LB.
ENGINES : TWO R.B. 106
R.C.A.F. SPECIFICATION
COMBAT CEILING - 1000 FT.
70
66
62
58
54
50
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
MACH NUMBER, M

FIG. 19

MINIMUM CRUISE AND MINIMUM LOITER FUEL FLOW AND ALTITUDE FOR MINIMUM CRUISE AND LOITER FUEL FLOW VS. MACH NUMBER