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

PREPARED BY O.E. Michaelsen

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 2.37g. (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

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.
	N.A.E. Analysis		A.V. Roe	
	N.A.E. C_{D_o}	A.V. Roe C_{D_o}	Analysis	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

s.
red

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		
.V. Roe	N.A.E. Analysis		A.V. Roe	N.A.E. Analysis		A.V. Roe
nalysis	N.A.E. C_{D_o}	A.V. Roe C_{D_o}	Analysis	N.A.E. C_{D_o}	A.V. Roe C_{D_o}	Analysis
-	4.0	4.0	4.0	663	663	660
39	3.34	3.01	3.2	4,114	3,757	3,740
.61	(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
.36	(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.

8.0 REFERENCES

1.	R.C.A.F.	Specification AIR 7-3 for "Design studies of prototype supersonic all-weather interceptor aircraft". Issue I, April 1953.
2.	R.C.A.F.	Operational Requirement OR 1/1-63 "Supersonic all-weather interceptor aircraft". First issue dated November 1952.
3.	A.V. Roe Canada Limited	Design study of supersonic all-weather interceptor aircraft. Report P/C-105/1, May 1953.
4.	A.V. Roe Canada Limited	Single engine all-weather fighter C 104/1.
5.	A.V. Roe Canada Limited	Twin engine all-weather fighter C 104/2.
6.	N.A.E.	Examination of two A.V. Roe (Canada) proposals for the C-104/1 and C-104/2 supersonic all-weather interceptor aircraft. L.R.-38, October 1952.
7.	Rolls Royce	Preliminary design study and installation data for the RB 106 engines.
8.	A.V. Roe Canada Limited	C.A.L. tests Sept. 1953. Corrected plots. C 105 Report P/Wind Tunnel/7 Sept. 1953.
9.	A.V. Roe Canada Limited	C.A.L. tests Sept. 1953. Derivatives and zero values. C 105 Report P/Wind Tunnel/8 Sept. 1953.
10.	A.V. Roe Canada Limited	C.A.L. tests Sept. 1953. Comparison of estimates with wind tunnel results. C 105 Report P/Wind Tunnel/9 Sept. 1953.

11.	Warren, S.H.E.	The estimation of the drag of an aircraft at supersonic speeds. R.A.E. T.M. Aero 132, Nov. 1950.
12.	Van Driest, E.R.	Turbulent boundary layer in compressible fluids. J.Ae.Sc. Vol. 18, No. 3, March 1951.
13.	Chapman, D.R. Kester, R.H.	Measurements of turbulent skin friction on cylinders in axial flow at subsonic and supersonic velocities. J.Ae.Sc. Vol. 20, No. 7, July 1953.
14.	Lawrence, T. Kell, C.	Zero lift drag measurements on swept wings at transonic and supersonic speeds using the ground-launched rocket-boosted model technique. R.A.E. T.N. Aero 2161, May 1952.
15.	Hopko, R.N. Sandahl, C.A.	Free-flight investigation of the zero-lift drag of several wings at supersonic Mach numbers extending to 2.6. N.A.C.A. R.M. L52D29, Dec. 1952.
16.	Nelson, R.L.	Large scale flight measurements of zero-lift drag at Mach numbers from 0.86 to 1.5 of a wing-body combination having a 60° triangular wing with N.A.C.A. 65A003 sections. N.A.C.A. R.M. L50D26, 1950.
17.	Schult, E.D.	Comparison of large-scale flight measurements of zero-lift drag at Mach numbers from 0.9 to 1.7 of two wing-body combinations having similar 60° triangular wings with N.A.C.A. 65A003 sections. N.A.C.A. RM L50I22, Oct. 1950.

18. Chapman, R.
Morrow, J.D.
Longitudinal stability and drag characteristics at Mach numbers from 0.70 to 1.37 of rocket-propelled models having a modified triangular wing.
N.A.C.A. RM L52A31, May 1952.

19. Fraenkel, L.E.
The theoretical wave drag of some bodies of revolution.
R.A.E. Report Aero 2420, May 1951.

20. Hart, R.G.
Katz, E.R.
Flight investigation at high-subsonic, transonic, and supersonic speeds to determine zero-lift drag of fin-stabilized bodies of revolution having fineness ratios of 12.5, 8.91 and 6.04 and varying position of maximum diameter.
N.A.C.A. RM L9I30, Nov. 1949.

21. Welsh, C.J.
Morrow, J.D.
Flight investigation at Mach numbers from 0.8 to 1.5 of the drag of a canopy located at two positions on a parabolic body of revolution.
N.A.C.A. RM L51A29, March 1951.

22. Alexander, S.R.
Effect of windshield shape of a pilot's canopy on the drag of an N.A.C.A. RM-2 drag research model in flight.
N.A.C.A. RM L8E04, 1948.

23. Purser, P.E.
Effect of a pilot's canopy on the drag of an N.A.C.A. RM-2 drag research model in flight at transonic and supersonic speeds.
N.A.C.A. RM L7L22, 1948.

24. Nichols, M.R.
Pendley, R.E.
Performance of air inlets at transonic and low supersonic speeds.
N.A.C.A. RM L52A07, Feb. 1952.

25. Pierpont, P.K.
Braden, J.A.
Investigation at transonic speeds of a forward-located underslung air inlet on a body of revolution.
N.A.C.A. RM L52K17, Jan. 1953.

26. Heitmeyer, J.C.
Smith, W.G.
Lift, drag and pitching moment of low aspect ratio wings at subsonic and supersonic speeds - Plane triangular wing of aspect ratio 2 with N.A.C.A. 0003-63 section.
N.A.C.A. RM A50K242, Feb. 1951.

27. Smith, D.W.
Heitmeyer, J.C.
Lift, drag and pitching moment of low aspect ratio wings at subsonic and supersonic speeds - Plane triangular wing of aspect ratio 2 with NACA 0005-63 section.
N.A.C.A. RM A50K21, Feb. 1951.

28. Smith, D.W.
Heitmeyer, J.C.
Lift, drag and pitching moment of low aspect ratio wings at subsonic and supersonic speeds - Plane triangular wing of aspect ratio 2 with N.A.C.A. 0008-63 section.
N.A.C.A. RM A50K20, Feb. 1951.

29. Heitmeyer, J.C.
Lift drag and pitching moment of low aspect ratio wings at subsonic and supersonic speeds - Plane triangular wing of aspect ratio 3 with N.A.C.A. 0003-63 section.
N.A.C.A. RM A51H02, Sept. 1951.

30. Lukasiewicz, J.
Laberge, J.G.
Stewart, J.D.

Comparison of characteristics of two small aspect ratio delta wings as determined experimentally and predicted theoretically and empirically.
N.A.E. Lab. Memo. AE-45b, Aug. 1953.

31. Boyd, J.W.

Aerodynamic characteristics of two 25-percent-area trailing-edge flaps on an aspect ratio 2 triangular wing at subsonic and supersonic speeds.
N.A.C.A. RM A52D01c, July 1952.

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 3½ 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:

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

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:

$$C_D = C_{D_o}' - K_1 C_L + K_2 C_L^2 \qquad (4)$$

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_0}", 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_0}", 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 tactical 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, must 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.

Loiter

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

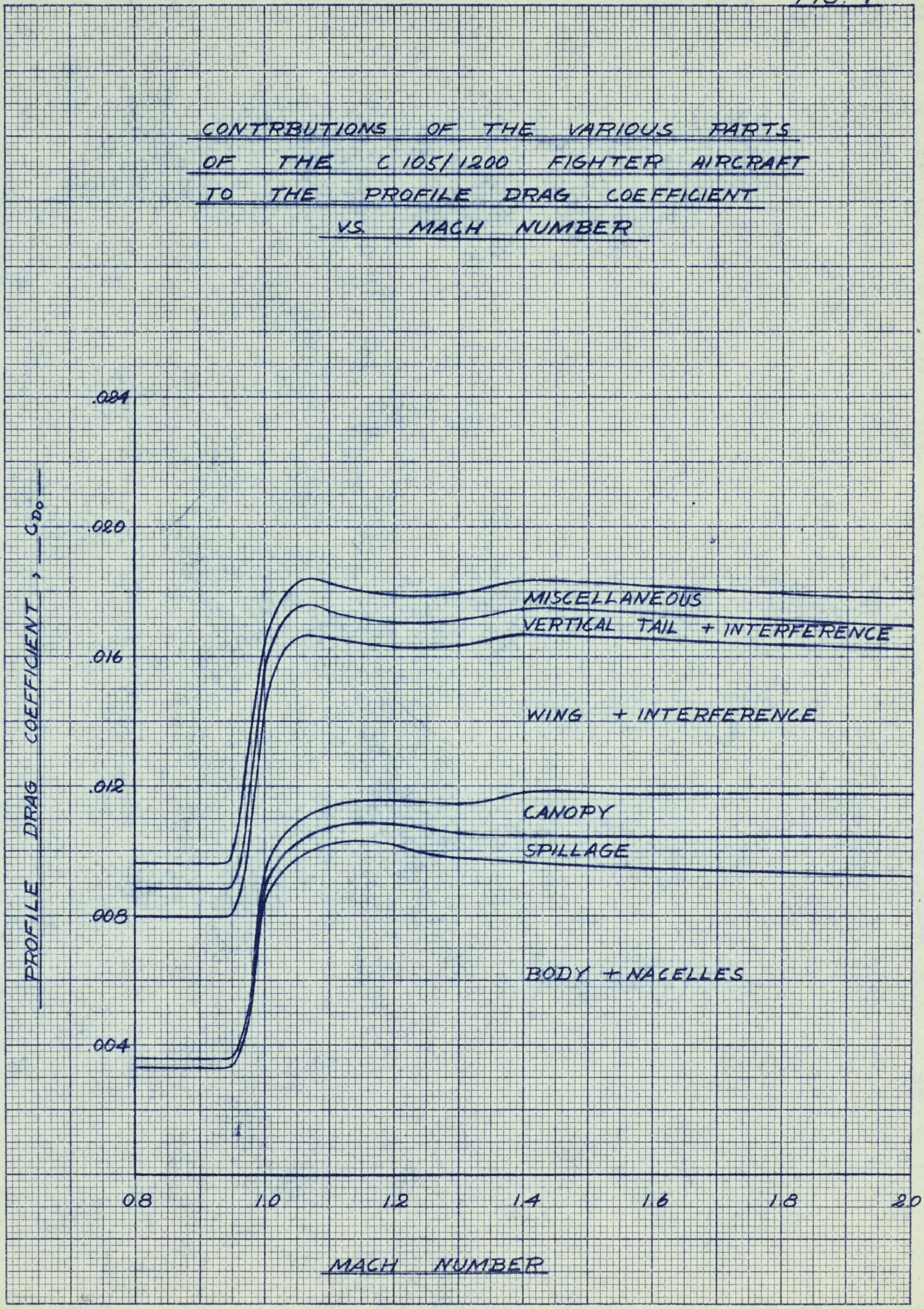
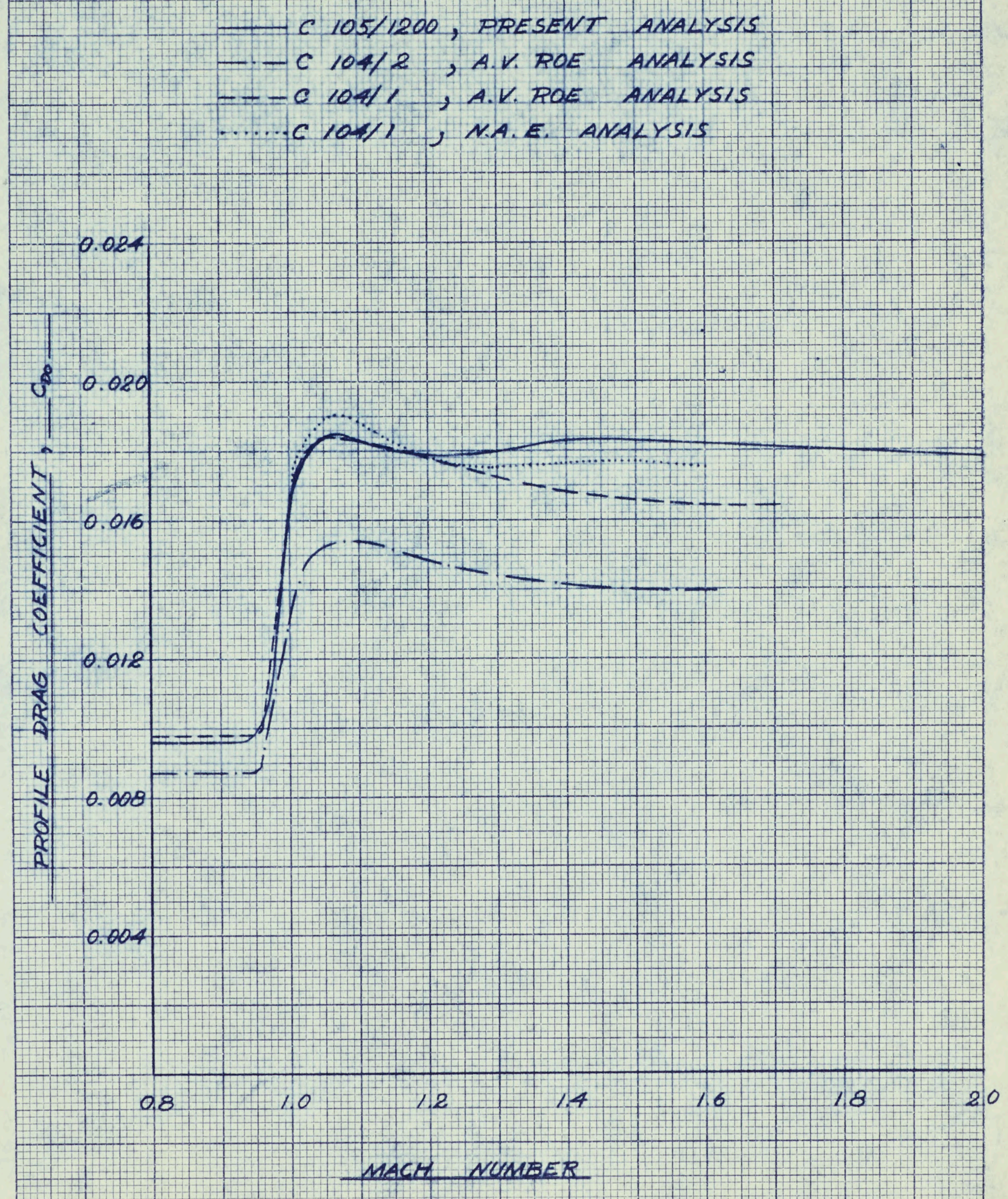

CONTRIBUTIONS OF THE VARIOUS PARTS
OF THE C 105/1200 FIGHTER AIRCRAFT
TO THE PROFILE DRAG COEFFICIENT
VS. MACH NUMBER
.024
.020
.016
.012
.008
.004
PROFILE DRAG COEFFICIENT, C_{D_0}
MISCELLANEOUS
VERTICAL TAIL + INTERFERENCE
WING + INTERFERENCE
CANOPY
SPILLAGE
BODY + NACELLES
0.8
1.0
1.2
1.4
1.6
1.8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

FIG. 2

COMPARISON OF PROFILE DRAG COEFFICIENTS.

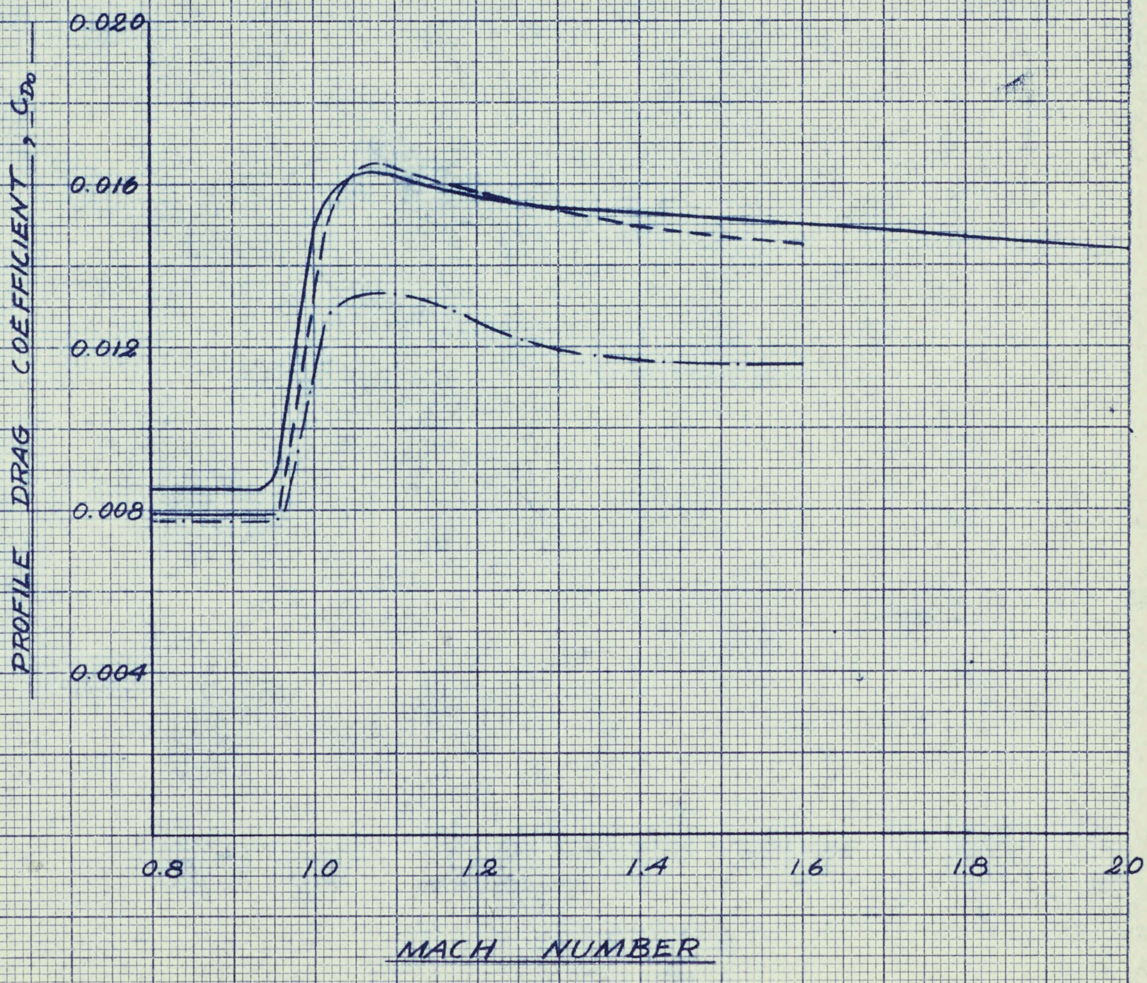


COMPARISON OF PROFILE DRAG COEFFICIENTS OF WING - BODY - VERTICAL TAIL.

——— 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.



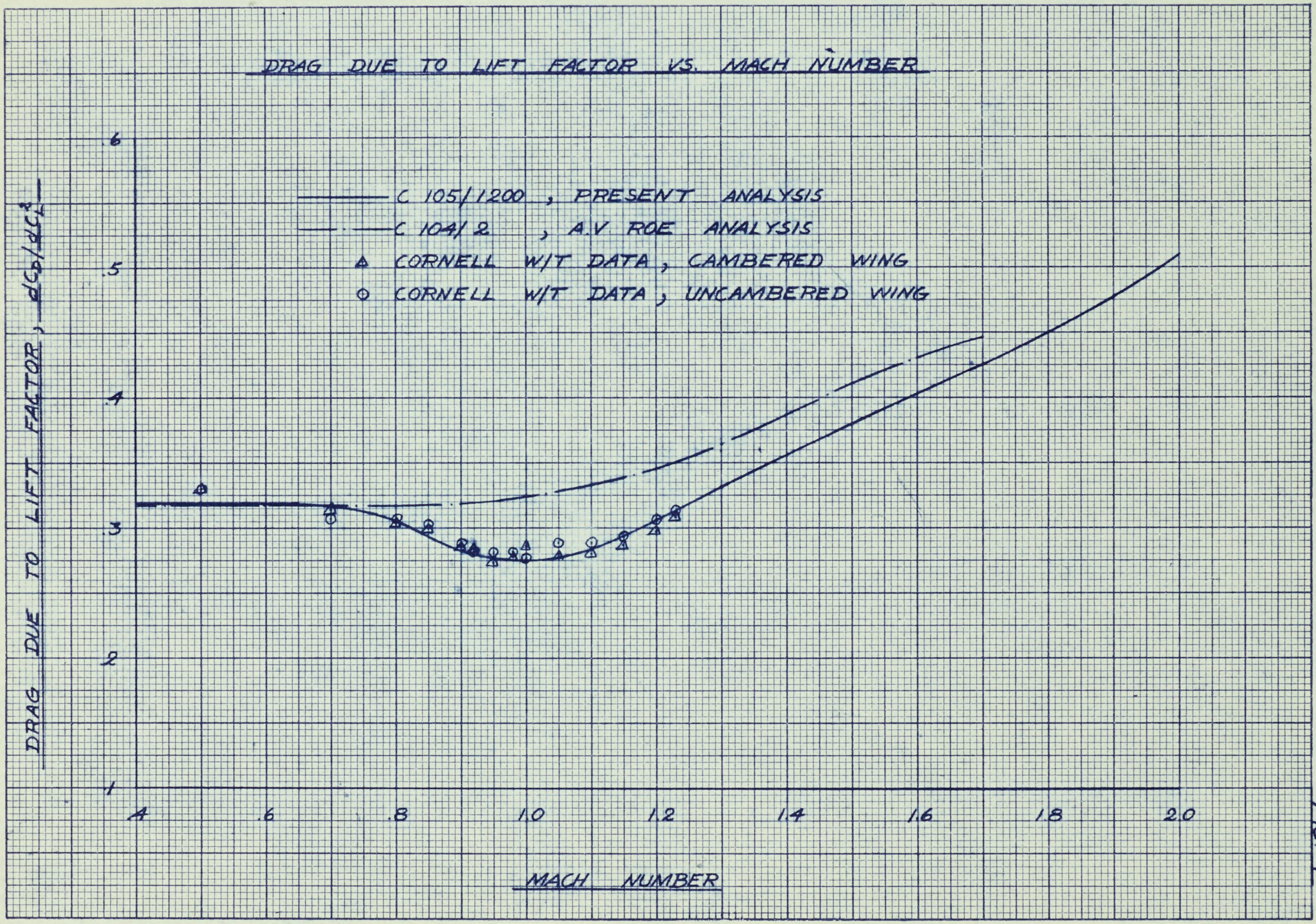

DRAG DUE TO LIFT FACTOR VS. MACH NUMBER
C 105/1200 , PRESENT ANALYSIS
C 104/2 , A.V ROE ANALYSIS
CORNELL W/T DATA , CAMBERED WING
CORNELL W/T DATA , UNCAMBERED WING
DRAG DUE TO LIFT FACTOR , dC_D/dC_L^2
6
5
4
3
2
1
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

FIG. 4

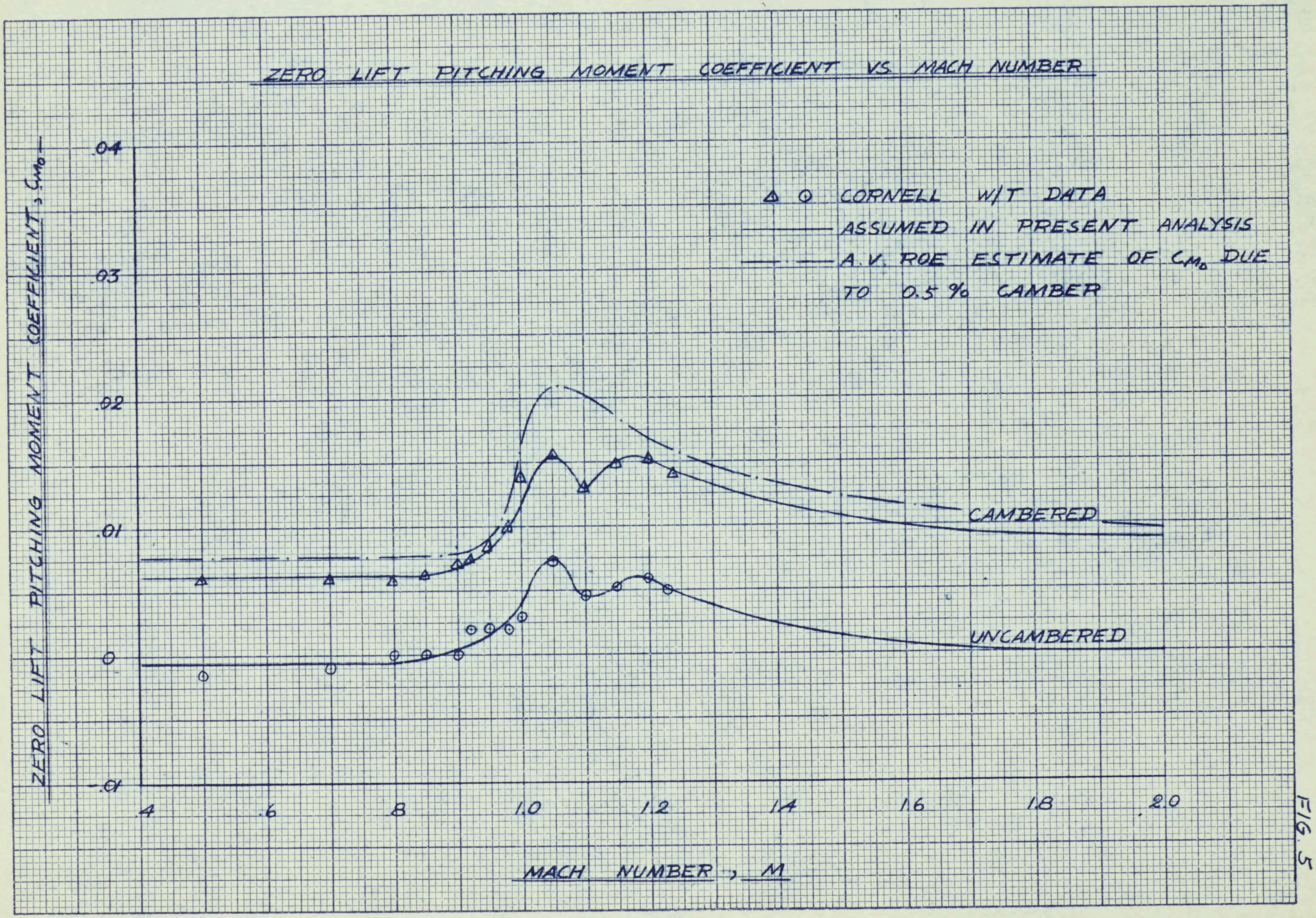

ZERO LIFT PITCHING MOMENT COEFFICIENT VS. MACH NUMBER
△ ○ CORNELL W/T DATA
—— ASSUMED IN PRESENT ANALYSIS
—·— A.V. ROE ESTIMATE OF C_{M_0} DUE TO 0.5 % CAMBER
CAMBERED
UNCAMBERED
ZERO LIFT PITCHING MOMENT COEFFICIENT, C_{M_0}
.04
.03
.02
.01
0
-.01
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER, M

FIG. 5

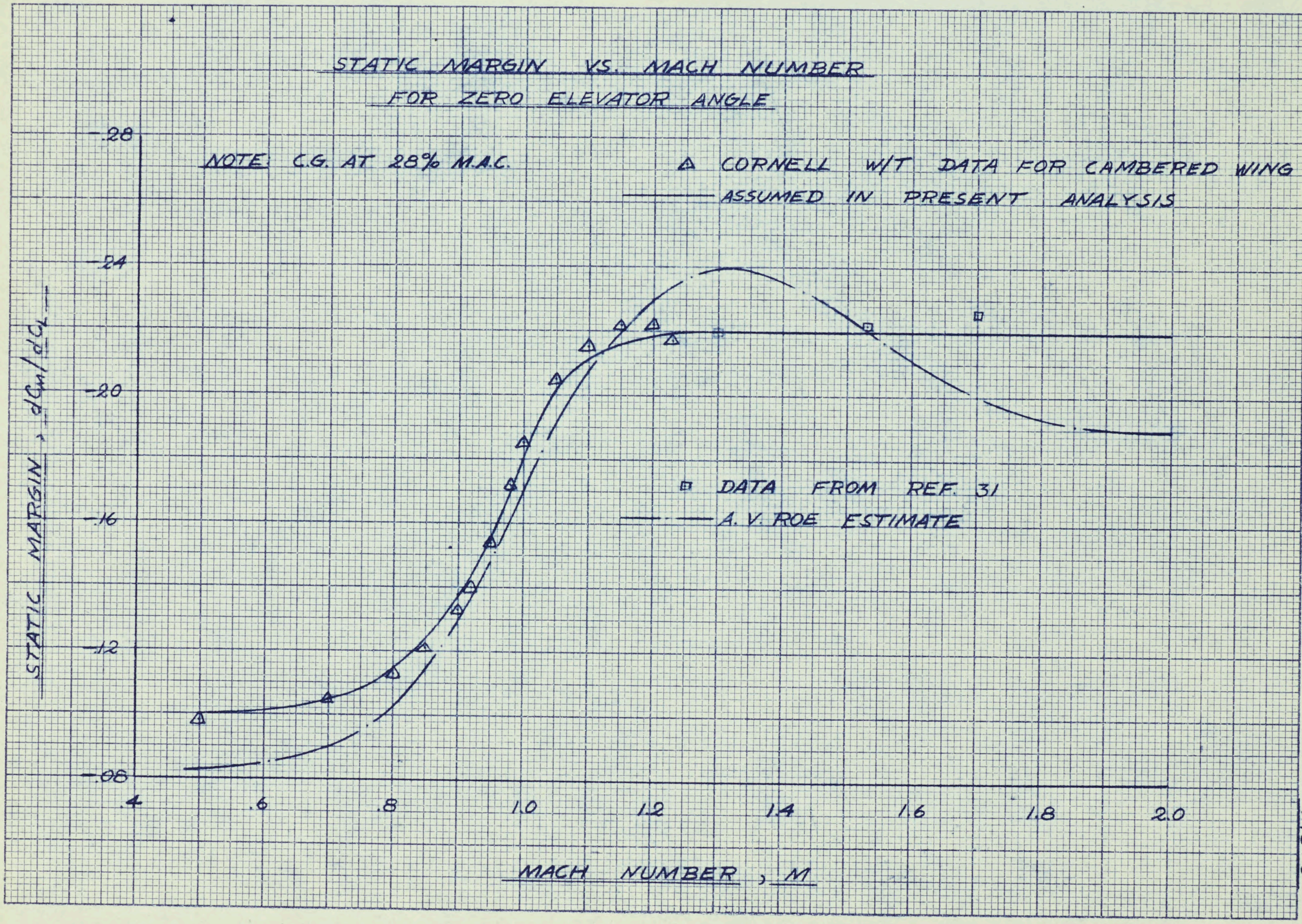

STATIC MARGIN VS. MACH NUMBER
FOR ZERO ELEVATOR ANGLE
NOTE: C.G. AT 28% M.A.C.
△ CORNELL W/T DATA FOR CAMBERED WING
—— ASSUMED IN PRESENT ANALYSIS
⊡ DATA FROM REF. 31
—·— A.V. ROE ESTIMATE
.28
.24
.20
.16
.12
.08
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
STATIC MARGIN, dCm/dCL
MACH NUMBER, M

FIG. 6

ELEVATOR EFFECTIVENESS FOR CONSTANT ANGLE OF ATTACK VS. MACH NUMBER.

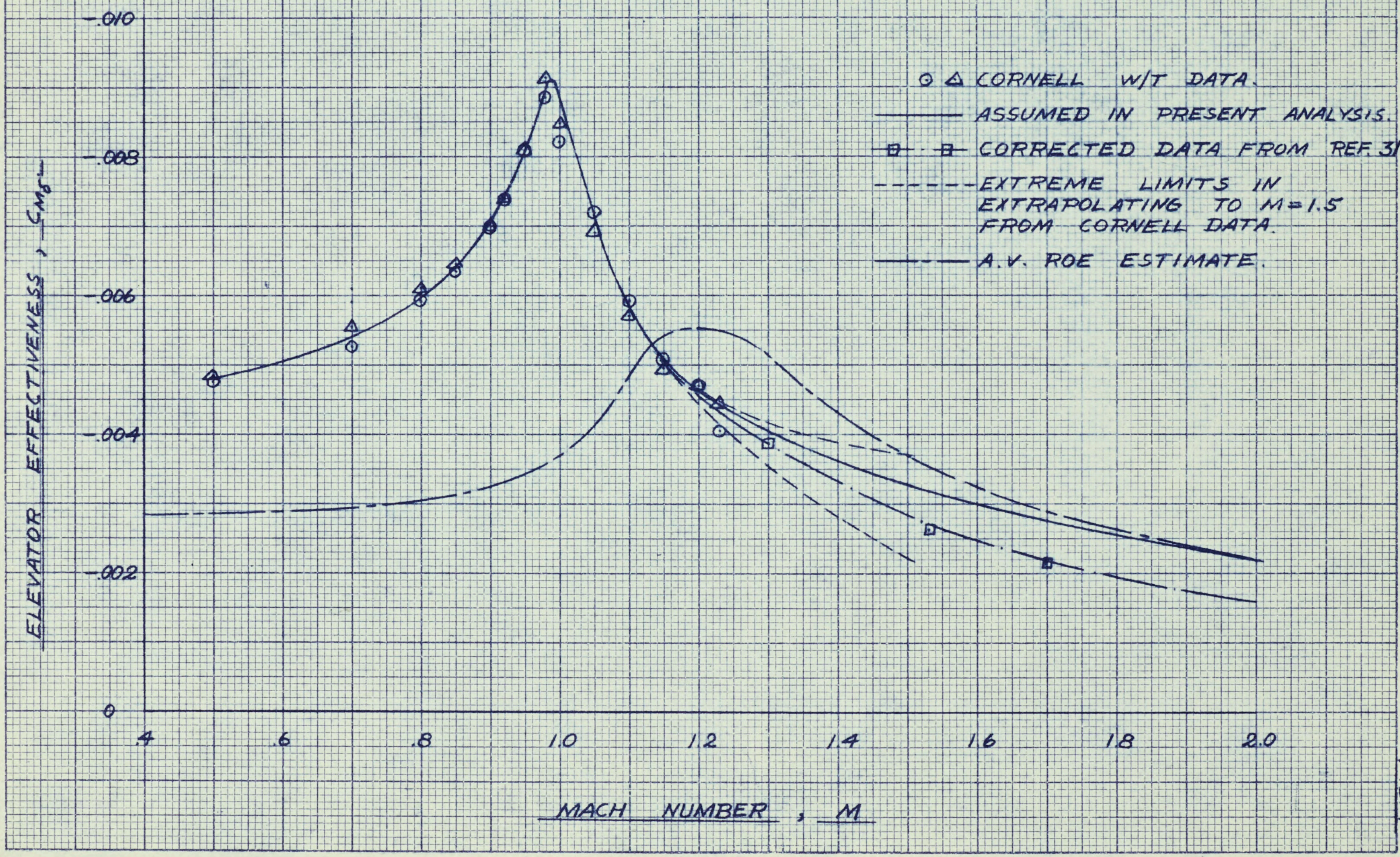
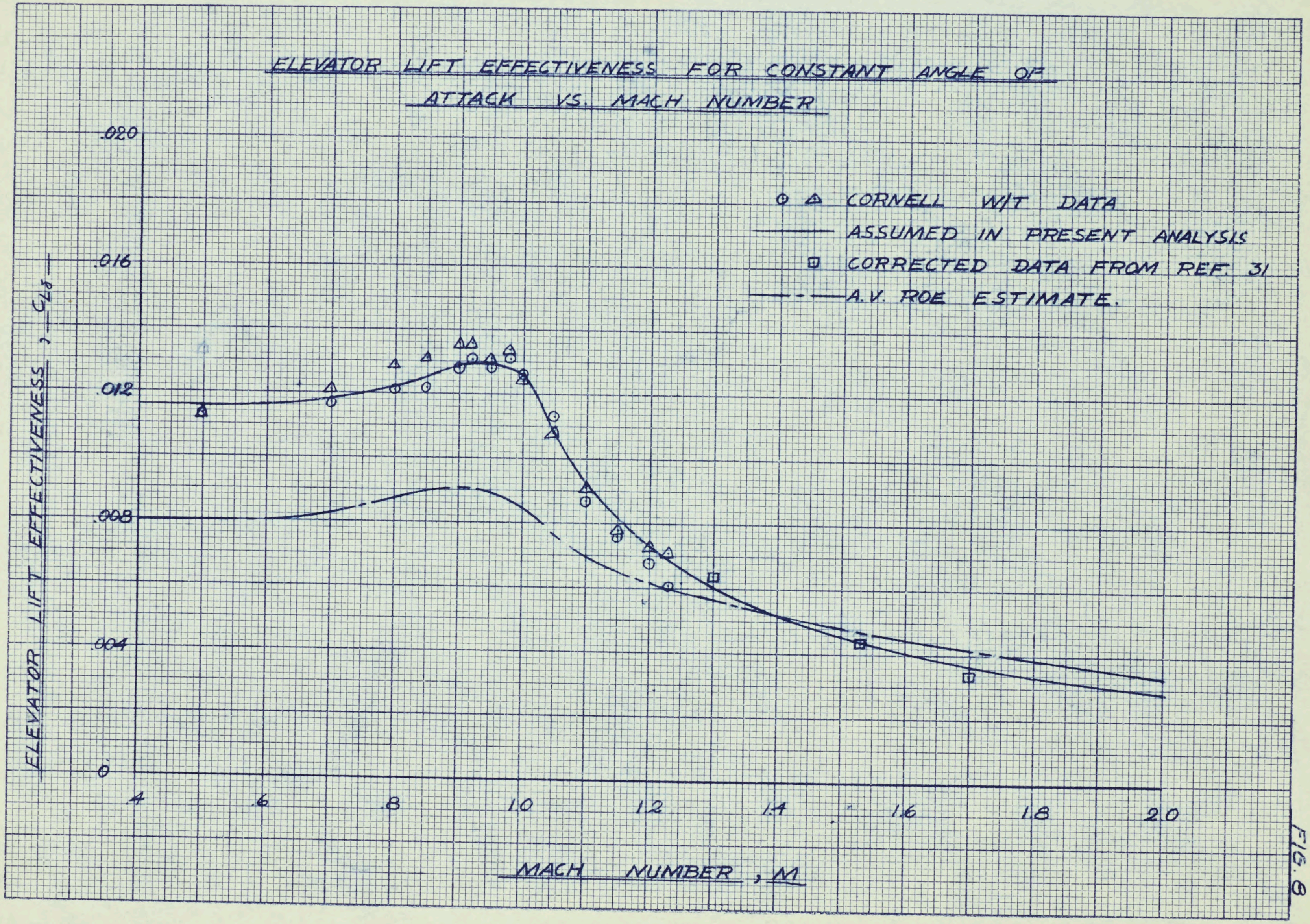


FIG. 7



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

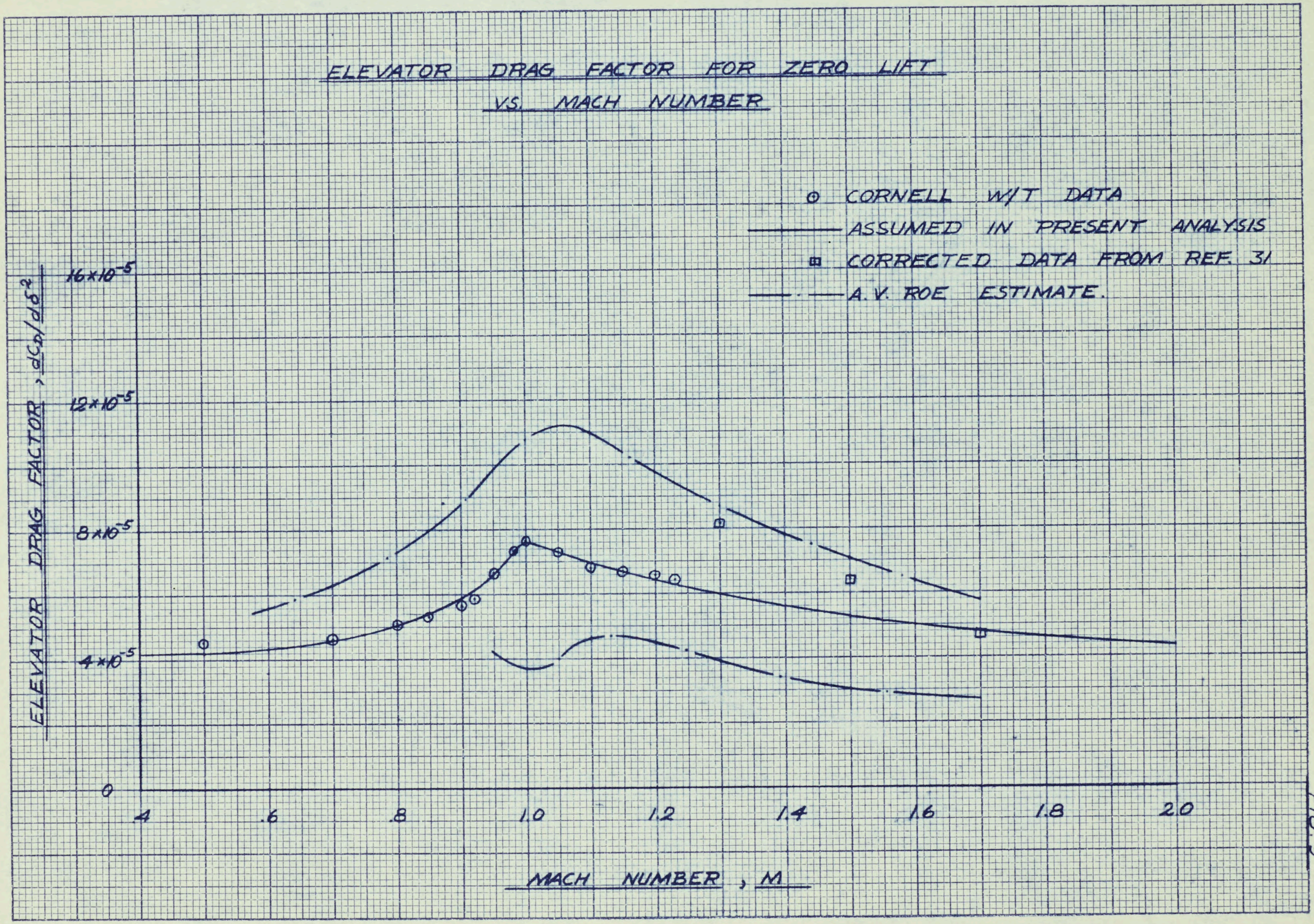

ELEVATOR DRAG FACTOR FOR ZERO LIFT
VS. MACH NUMBER
CORNELL W/T DATA
ASSUMED IN PRESENT ANALYSIS
CORRECTED DATA FROM REF. 31
A.V. ROE ESTIMATE.
ELEVATOR DRAG FACTOR, $dC_D/d\delta^2$
16×10^{-5}
12×10^{-5}
8×10^{-5}
4×10^{-5}
0
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER, M

FIG. 9

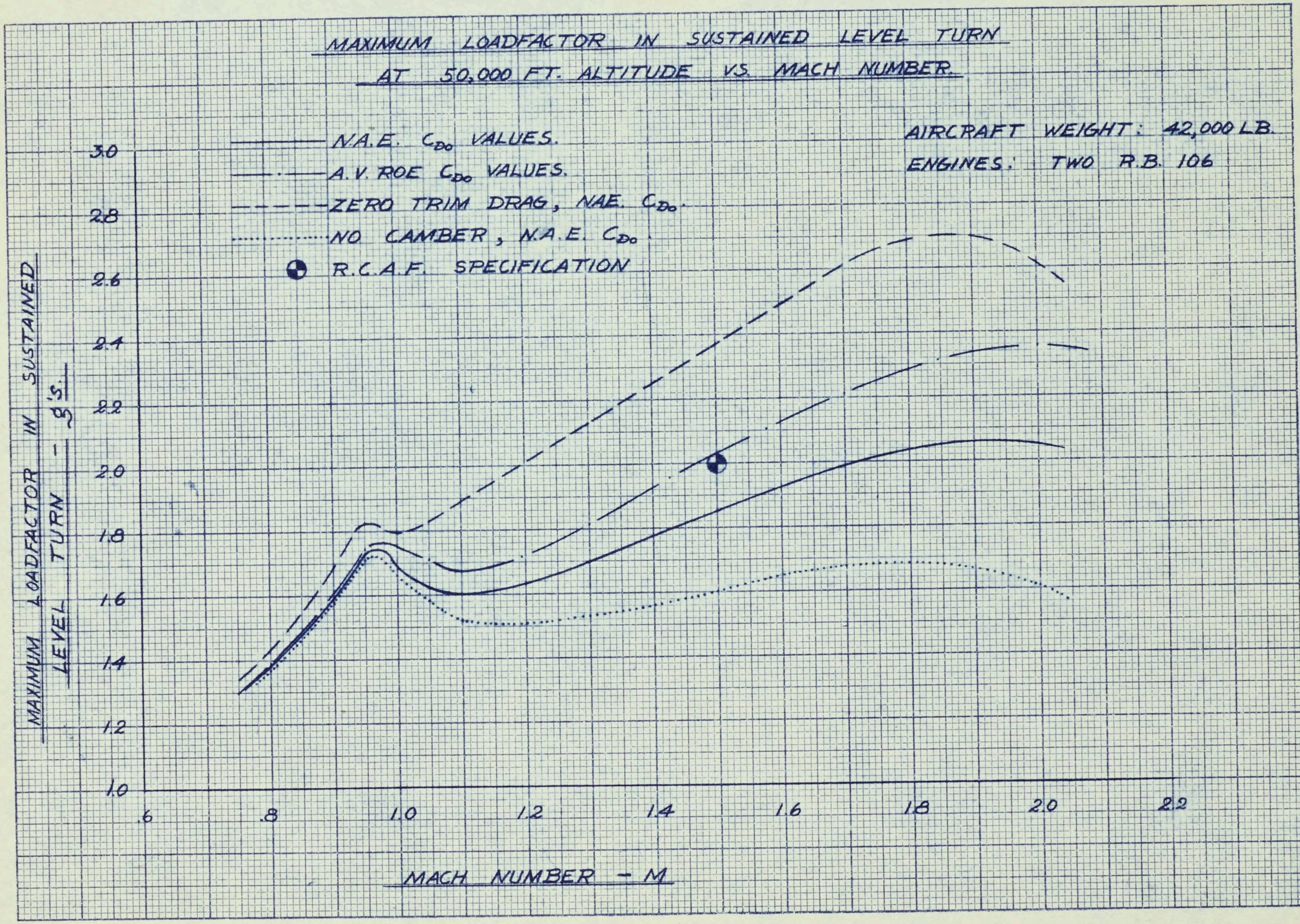
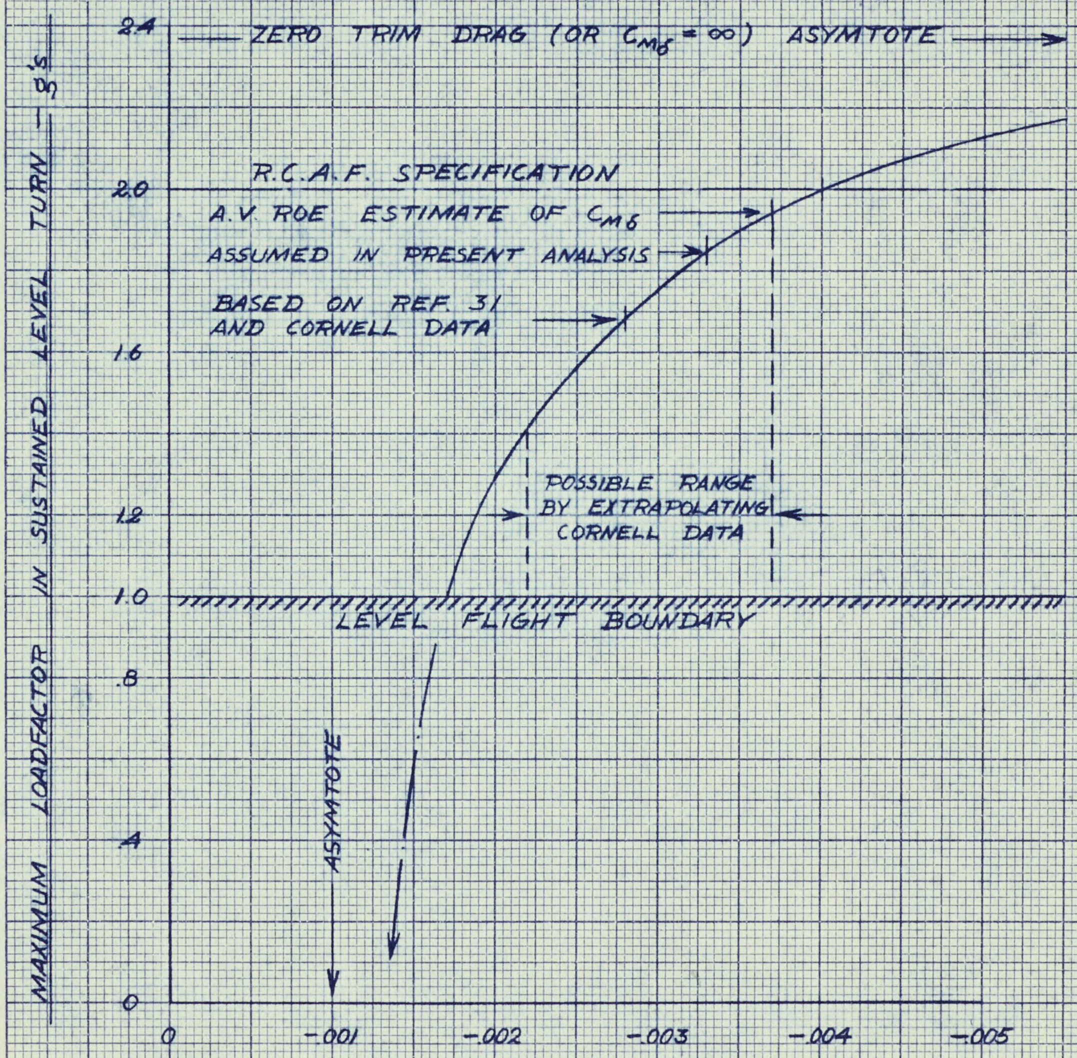

MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN
AT 50,000 FT. ALTITUDE VS. MACH NUMBER.
N.A.E. C_{Do} VALUES.
A.V. ROE C_{Do} VALUES.
ZERO TRIM DRAG, NAE. C_{Do}.
NO CAMBER, N.A.E. C_{Do}
R.C.A.F. SPECIFICATION
AIRCRAFT WEIGHT: 42,000 LB.
ENGINES: TWO R.B. 106
MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN - 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

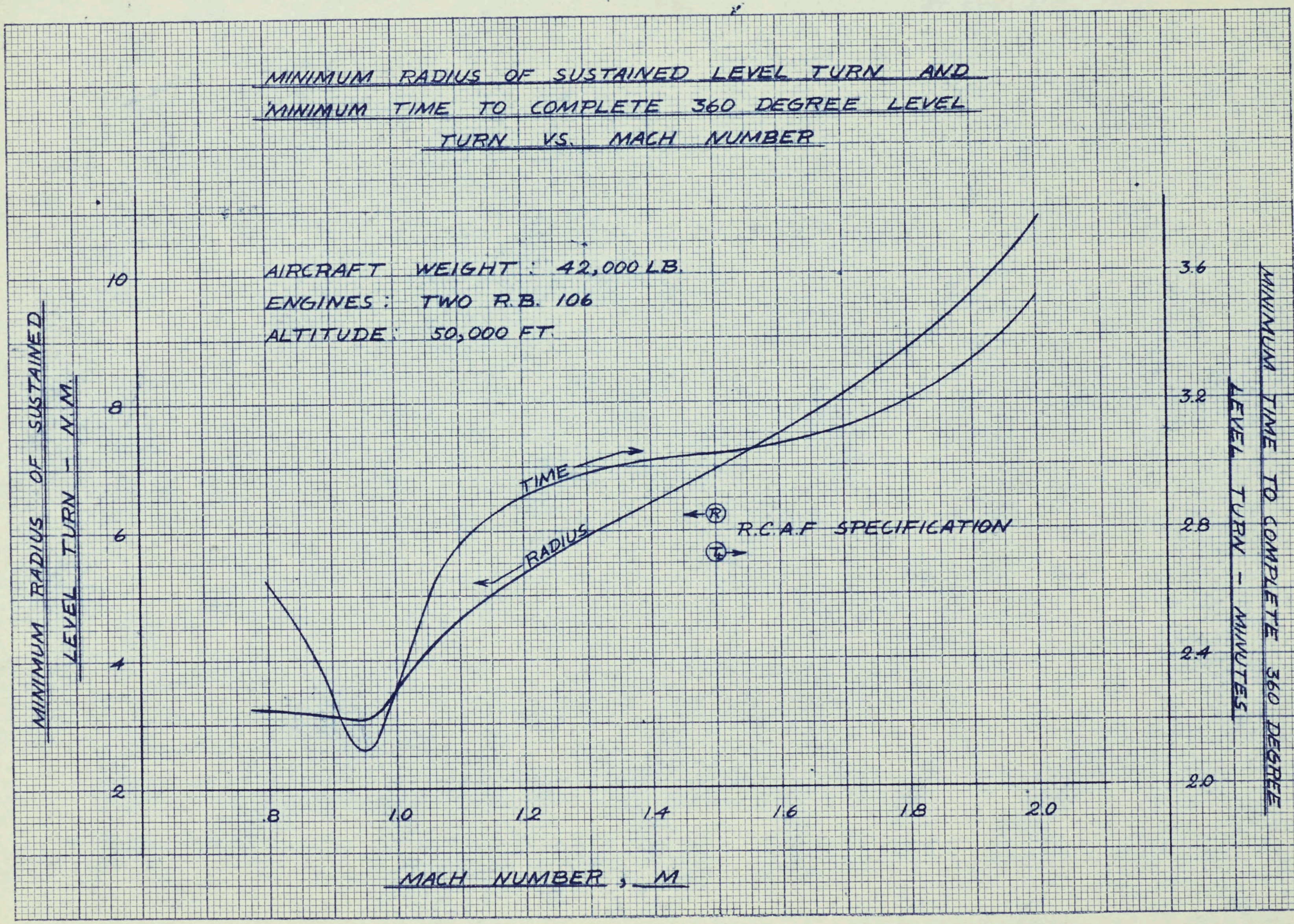
SENSITIVITY OF MAXIMUM LOADFACTOR IN SUSTAINED LEVEL TURN AT M = 1.5 AND 50,000 FT. ALTITUDE TO VALUE OF ELEVATOR EFFECTIVENESS $C_{M\delta}$

AIRCRAFT WEIGHT : 42,000LB

ENGINES : TWO RB. 106



ELEVATOR EFFECTIVENESS, $C_{M\delta}$



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.
TIME
RADIUS
R.C.A.F SPECIFICATION
MINIMUM RADIUS OF SUSTAINED LEVEL TURN - N.M.
MINIMUM TIME TO COMPLETE 360 DEGREE LEVEL TURN - MINUTES
MACH NUMBER , M
.8
1.0
1.2
1.4
1.6
1.8
2.0
2
4
6
8
10
2.0
2.4
2.8
3.2
3.6

FIG. 12

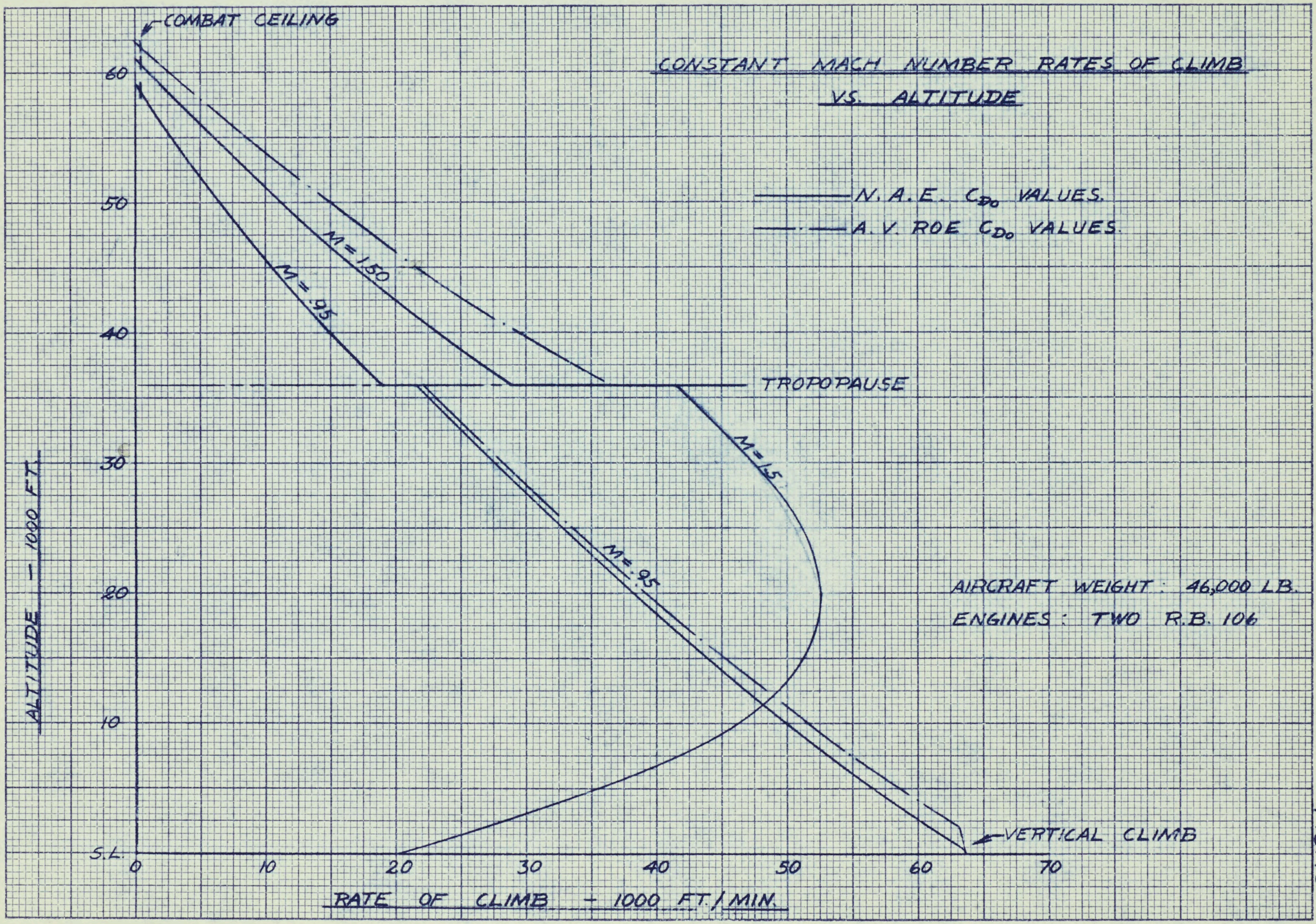
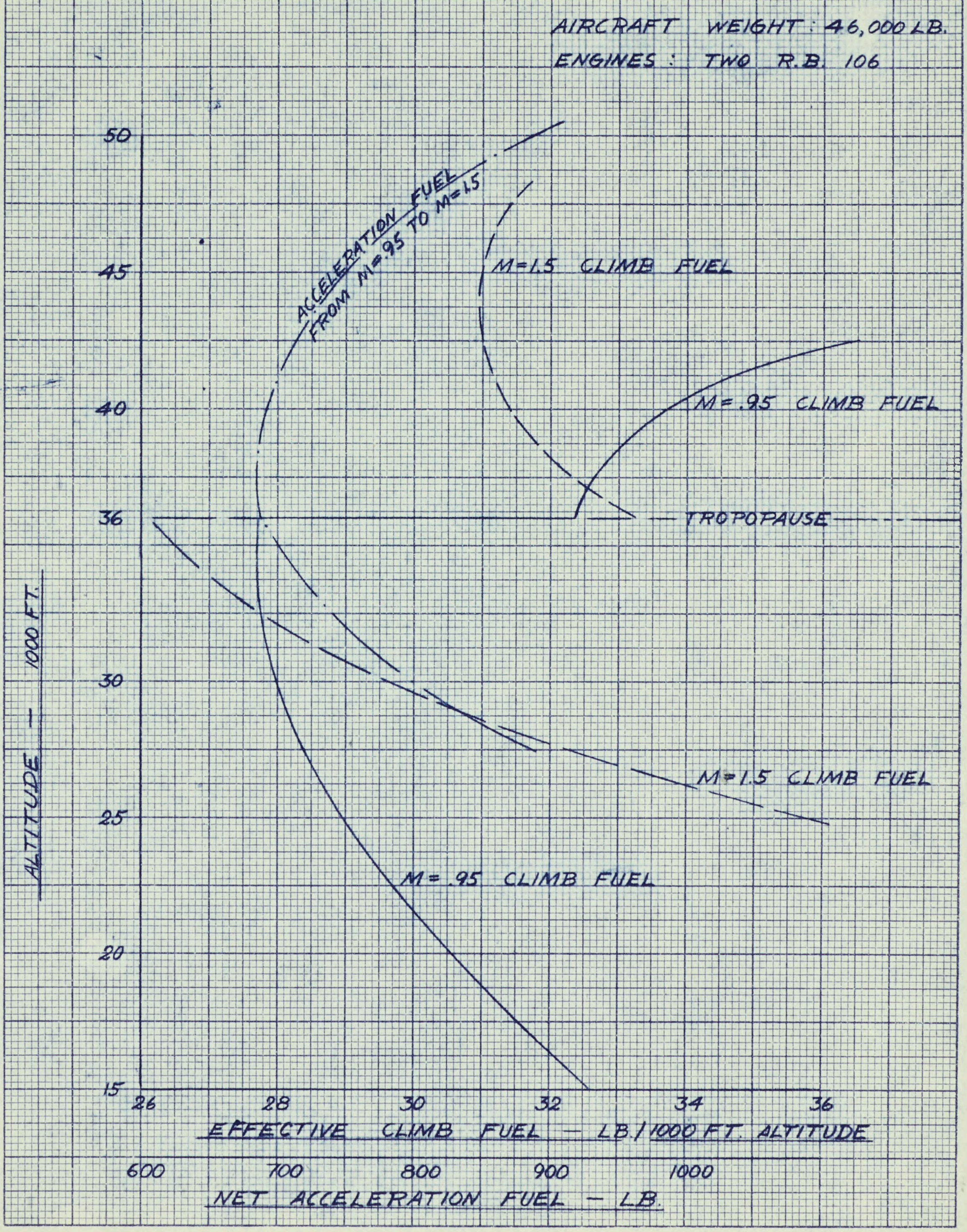
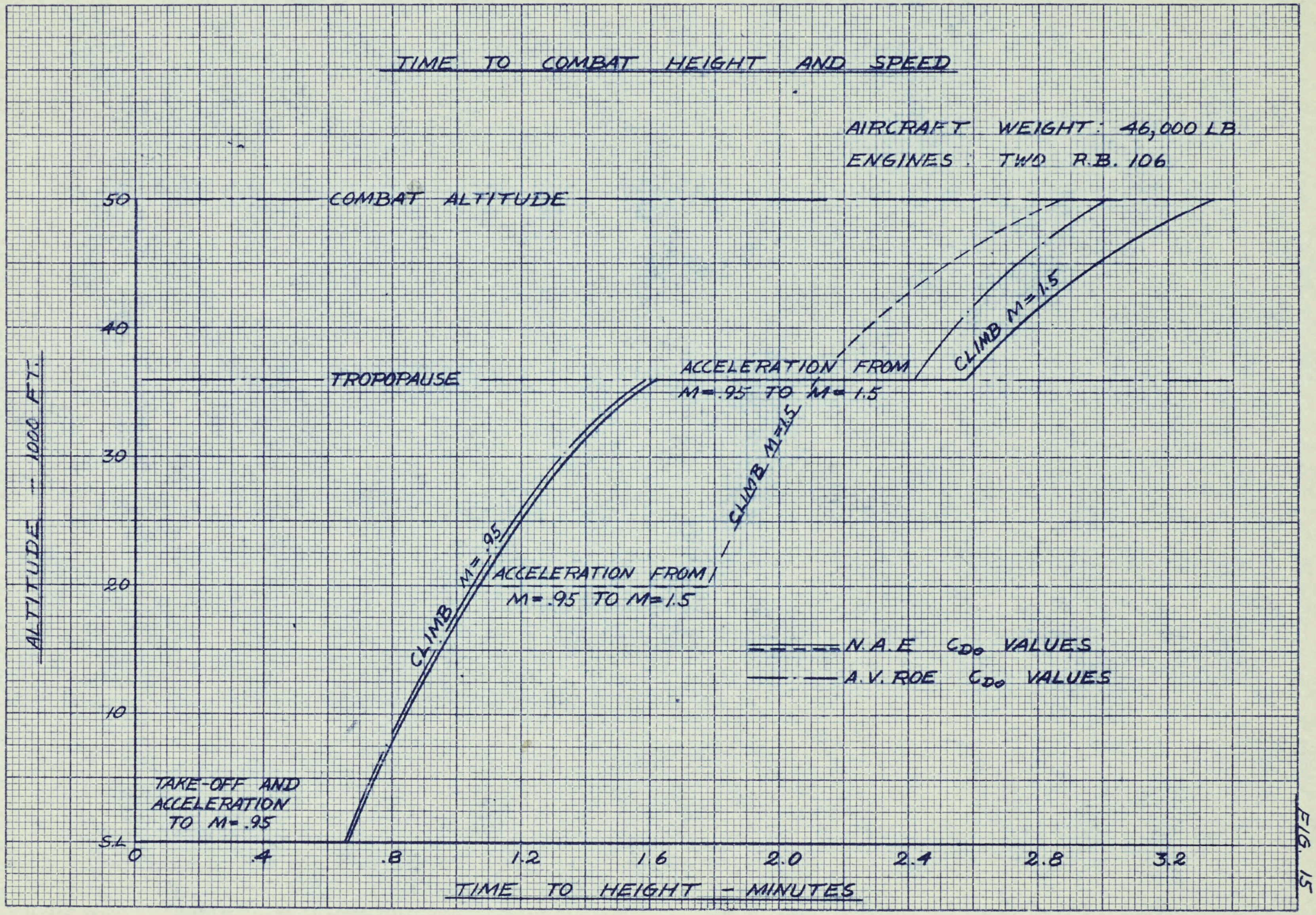

CONSTANT MACH NUMBER RATES OF CLIMB
VS. ALTITUDE
N. A. E. C_{D_0} VALUES.
A. V. ROE C_{D_0} VALUES.
COMBAT CEILING
TROPOPAUSE
M = 1.50
M = .95
M = 1.5
M = .95
VERTICAL CLIMB
AIRCRAFT WEIGHT : 46,000 LB.
ENGINES : TWO R.B. 106
ALTITUDE — 1000 FT.
RATE OF CLIMB — 1000 FT./MIN.
S.L.
10
20
30
40
50
60
0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

FIG. 13

FIG. 14

EFFECTIVE CLIMB FUEL AND NET ACCELERATION FUEL VS ALTITUDE





TIME TO COMBAT HEIGHT AND SPEED
AIRCRAFT WEIGHT: 46,000 LB.
ENGINES: TWO R.B. 106
COMBAT ALTITUDE
TROPOPAUSE
ACCELERATION FROM M=.95 TO M=1.5
CLIMB M=1.5
CLIMB M=1.5
CLIMB M=.95
ACCELERATION FROM M=.95 TO M=1.5
TAKE-OFF AND ACCELERATION TO M=.95
N.A.E C_{Do} VALUES
A.V. ROE C_{Do} VALUES
ALTITUDE - 1000 FT.
50
40
30
20
10
S.L.
0
.4
.8
1.2
1.6
2.0
2.4
2.8
3.2
TIME TO HEIGHT - MINUTES

FIG. 15

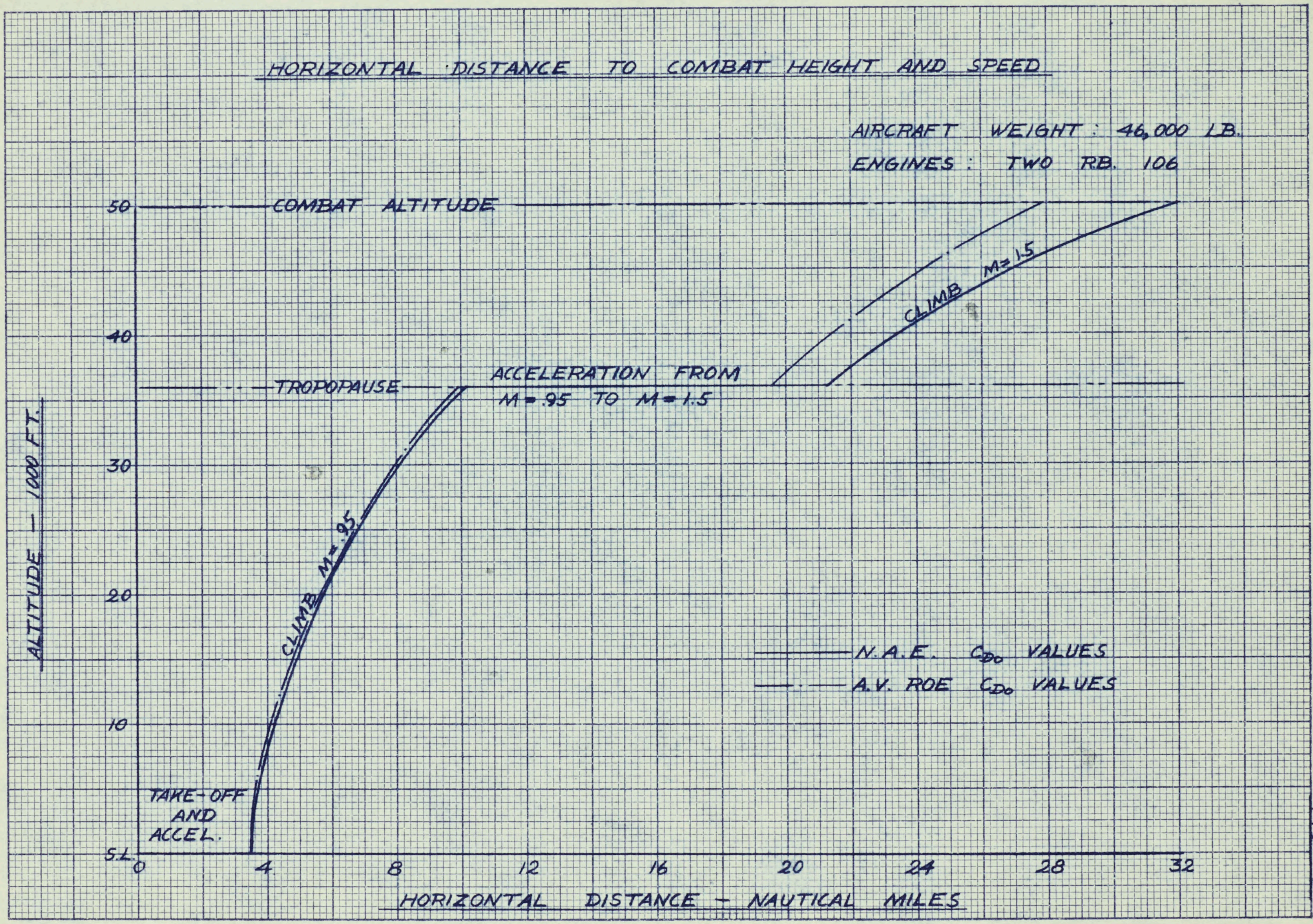
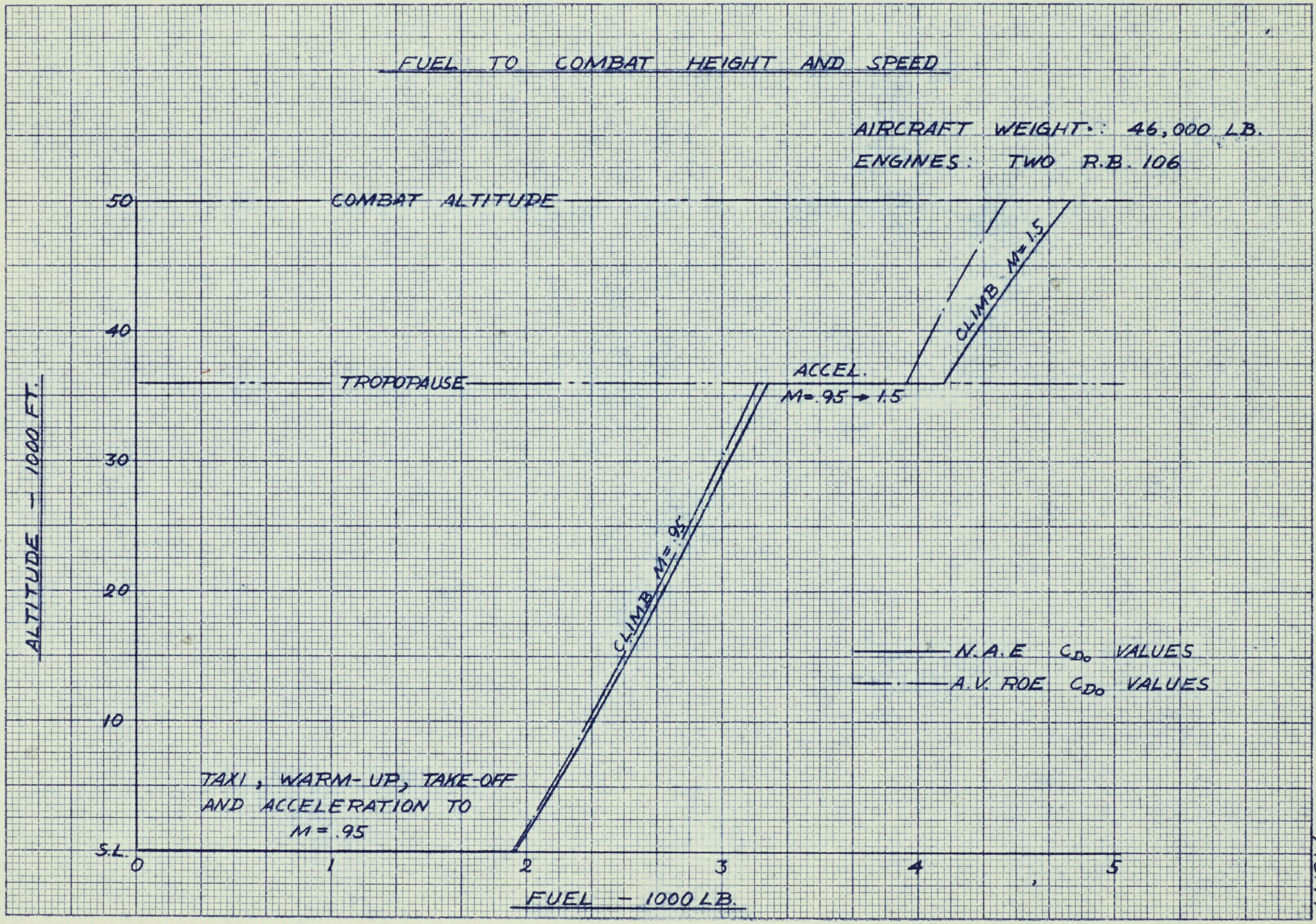

HORIZONTAL DISTANCE TO COMBAT HEIGHT AND SPEED
AIRCRAFT WEIGHT : 46,000 LB.
ENGINES : TWO RB. 106
COMBAT ALTITUDE
TROPOPAUSE
ACCELERATION FROM M = .95 TO M = 1.5
CLIMB M = 1.5
CLIMB M = .95
TAKE-OFF AND ACCEL.
N.A.E. C_{Do} VALUES
A.V. ROE C_{Do} VALUES
ALTITUDE — 1000 FT.
HORIZONTAL DISTANCE — NAUTICAL MILES
S.L.
10
20
30
40
50
0
4
8
12
16
20
24
28
32

FIG. 16



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
N.A.E C_{D_0} VALUES
A.V. ROE C_{D_0} VALUES
TAXI, WARM-UP, TAKE-OFF
AND ACCELERATION TO
M=.95
50
40
30
20
10
S.L.
0
1
2
3
4
5
ALTITUDE — 1000 FT.
FUEL — 1000 LB.

FIG. 17

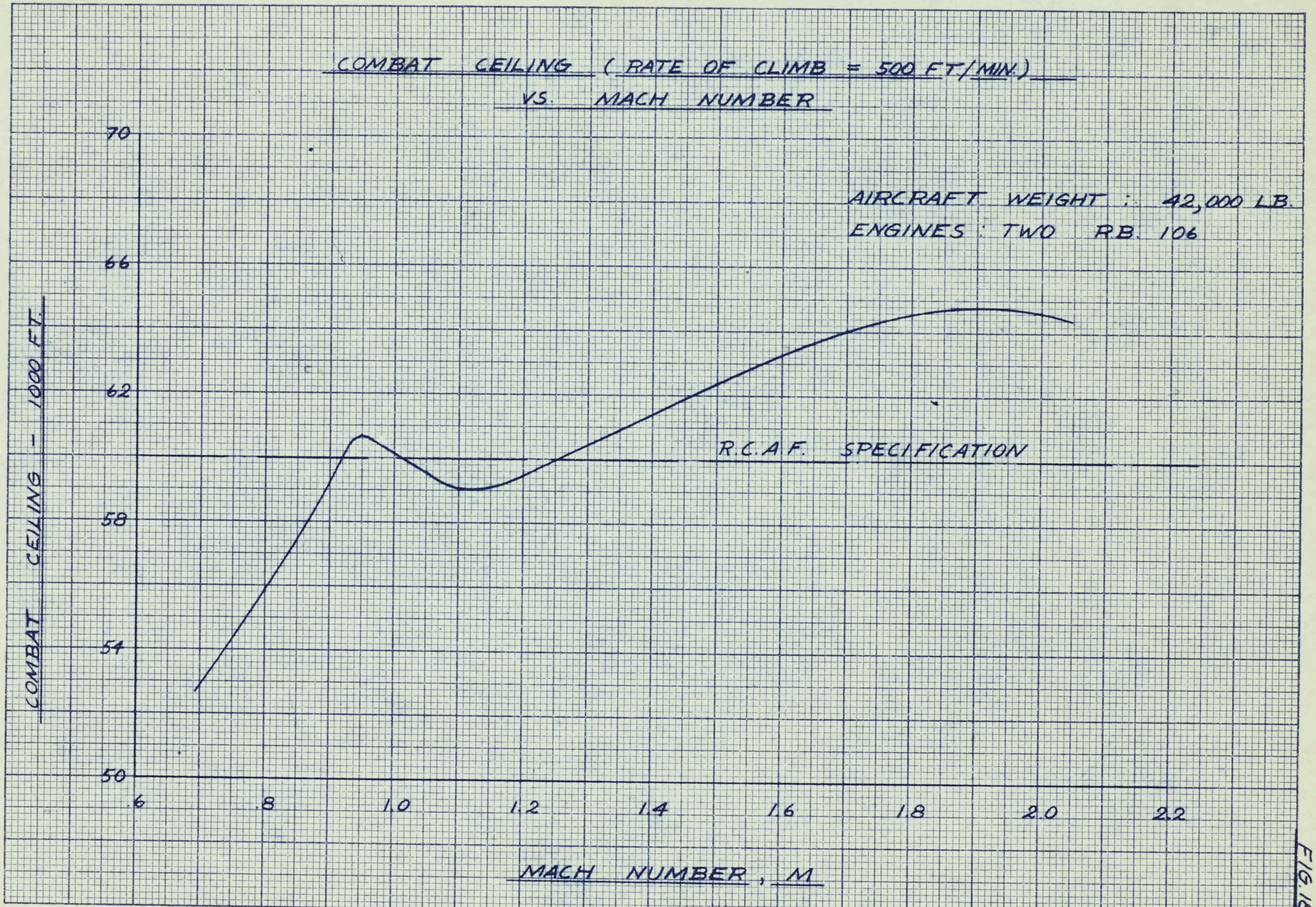
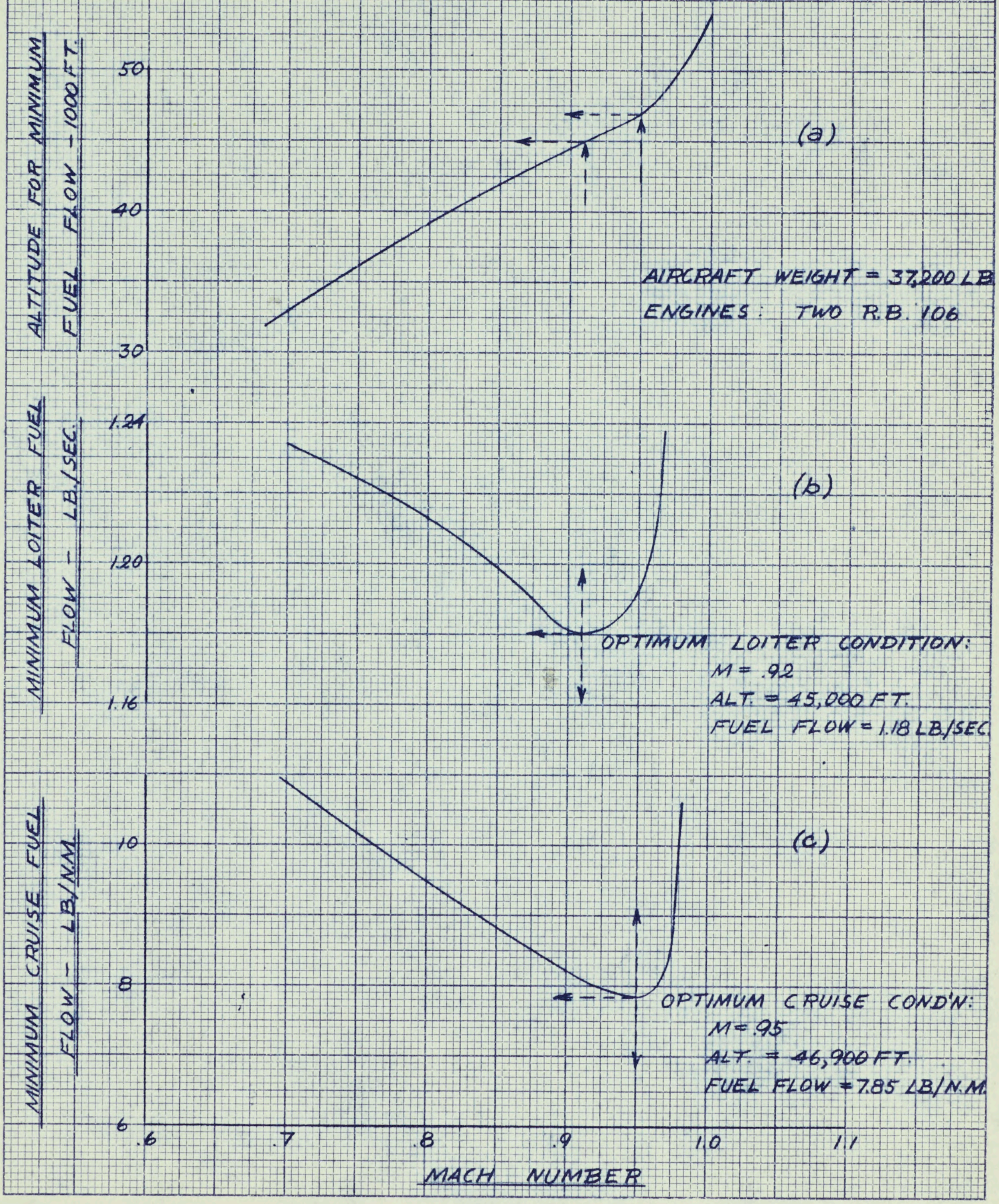

COMBAT CEILING (RATE OF CLIMB = 500 FT./MIN.)
VS. MACH NUMBER
AIRCRAFT WEIGHT : 42,000 LB.
ENGINES : TWO RB. 106
R.C.A.F. SPECIFICATION
COMBAT CEILING - 1000 FT.
70
66
62
58
54
50
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
MACH NUMBER, M

FIG. 18

FIG. 19

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



Mr. J. Lukasiewicz

L.O. NAE-125

FILE BM49-7-12

PREPARED BY RJT

CHECKED BY

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OTTAWA, CANADA

LABORATORY MEMORANDUM

SECTION Aerodynamics

NO. AE-62

COPY NO. 9

DATE 1 Feb., 1955

SECURITY CLASSIFICATION Secret

SUBJECT NOTE ON SUPERSONIC BOMBERS POWERED BY TURBO-JETS

PREPARED BY R.J. Templin

ISSUED TO Internal

SUMMARY

This note examines the possibility of achieving long range with turbo-jet bombers designed to cruise at supersonic speeds. It is concluded that still air ranges up to 5000 miles from the top of the climb are possible at low supersonic speeds in view of recent aerodynamic advances. At Mach numbers between 1.5 and 2.0, however, maximum range appears to decrease to about 3000 miles. In all cases little increase in range is achieved by increasing aircraft gross weight above 300,000 to 400,000 pounds. Altitudes over the target are over 50,000 ft., and in some cases 70,000 ft.

TABLE OF CONTENTS

LIST OF SYMBOLS

a	speed of sound (ft./sec.)
A	wing aspect ratio
b	wing span (ft.)
c	engine specific fuel consumption (lb./lb.-hr.)
C_{D_b}	fuselage drag coefficient (based on fuselage frontal area)
C_{D_f}	wing skin friction drag coefficient
C_{D_t}	wing thickness drag coefficient
$C_{D_{o_w}} = C_{D_f} + C_{D_t}$	= wing zero lift drag coefficient
D_c	cruising drag (lb.)
e	wing span efficiency
K_e	$\frac{\text{engine weight} \times o}{\text{thrust}}$ (assumed to be a function of Mach number only in the stratosphere)
M	design cruising Mach number
n	ultimate load factor (assumed to be 6.0)
R	still air range from top of climb (miles)
S	gross wing area (sq.ft.)
S_b	fuselage fontal area (sq.ft.)
W	aircraft weight (lb.)
W_o	aircraft gross weight at take-off (lb.)
W_f	weight of fuel remaining at top of climb (lb.)
W_{f_o}	weight of fuel at take-off (lb.)
W_1	aircraft weight at top of climb (lb.)
W_w	wing structure weight (lb.)
$\frac{dC_D}{dC_L^2}$	drag due to lift factor
β	$\sqrt{M^2-1}$
Λ_o	wing leading edge sweepback

Λ wing structural sweep

ρ_o standard sea level relative density = 0.00238 slug/cu.ft.

σ relative density at cruising altitude (varies during flight)

σ_o relative density at effective initial cruising altitude

τ wing thickness-chord ratio.

LIST OF ILLUSTRATIONS

1.0 INTRODUCTION

Some months ago it was decided to initiate within the Division a study of supersonic bomber capabilities, in order to provide a guide toward future work relating to interception devices. Mr. A.D. Wood has completed one part of this study, dealing with long range ballistic rockets (Reference 1).

The present memorandum describes the results of calculations of range for conventional bomber configurations powered by turbo-jet engines. It is understood that a third memorandum will be published by the Gas Dynamics Laboratory, which discusses the possibility of increasing range by the use of "lifting" engines of a type similar to those which have been proposed by Messrs. Rolls-Royce.

The cruising speed range dealt with in the present analysis extends from a Mach number of 0.9 to 2.0.

In addition to the primary reason for this study, several other aims were borne in mind in carrying out this part of the work. In the first place, it provided an opportunity to assess the potential benefits from the area rule and from the use of wing camber at low supersonic speeds. Secondly, the collection and correlation of supersonic wing data, which was carried out by the Aerodynamics Laboratory during the past two years, had never been put to use in a systematic analysis of aircraft configurations. For the supersonic bombers considered here, the wing configurations have been chosen from the results of these empirical correlations.

2.0 OUTLINE OF METHOD OF ANALYSIS

This section (paragraphs 2.01 to 2.16 inclusive) may be omitted by the reader who is interested only in the results of the analysis. The following paragraphs describe the method of calculating range and the assumptions made regarding items of weight and the estimation of drag.

2.01 Payload

Since in general the weight of some parts of the structure of an aircraft can not be assumed to be a constant fraction of design gross weight, independently of gross weight itself, it appeared necessary to assume at the outset an absolute value for payload. This was chosen to be 10,000 lb.

2.02 Fuselage

For a manned bomber carrying this order of payload, the fuselage diameter tends to be fixed at about 10 feet. Since, also, we are discussing supersonic bombers, it is possible to specify a desirable fuselage fineness ratio of about 10 in order to keep total fuselage drag at a minimum. Hence a fuselage 100 ft. long and 10 ft. in diameter was chosen. The fuselage is thus a body of fixed size (and structure weight), whose drag can be calculated immediately for any given Mach number and altitude. At supersonic speeds its drag coefficient was taken to be 0.20 (based on frontal area) and at subsonic speeds, 0.008. The supersonic value is conservative when compared with measured drag coefficients of good smooth bodies. A slightly conservative value was chosen because it was decided in the interests of simplicity not to assume any drag associated with the propulsion system installation.

According to the statistical data of Reference 2, this fuselage weighs 10,000 lb.

2.03 Engine Weight

In order to be consistent with the analysis carried out by the Gas Dynamics Section, the engine specific weight in the present case was taken to be the same as that assumed by Mr. Tyler. This is outlined below:

Mach number:	1.0	1.25	1.50	1.75	2.0
Engine weight per lb. of thrust at 50,000 ft:	1.385	1.16	1.01	0.93	0.97

These data were extrapolated to a value of 1.5 at a Mach number of 0.9. It is understood from Mr. Tyler that these figures are slightly optimistic as compared with practical values at the present time.

In order to calculate engine weight as a fraction of aircraft weight it was assumed that the engines are just large enough to produce a thrust equal to cruising drag at maximum continuous rating.

2.04 Fixed Equipment

The weight of fixed equipment, including electronic bombing aids, instruments, etc., was taken to be 5000 lb.

2.05 Undercarriage

The weight of undercarriage was assumed to be 0.06 W_o where W_o is the aircraft gross weight.

2.06 Climb Fuel

It was assumed that the weight of fuel required for take-off and climb to cruising altitude is 0.05 W_o. This value was taken from an analysis of turbo-jet transports carried out in the laboratory a number of years ago. It appears likely that the supersonic bombers considered here would climb initially at subsonic speeds.

2.07 Tail Weight

The weight of the tail is usually a small fraction of gross weight and need not be estimated with great accuracy. According to Driggs (Fig. 37) the tail may be expected to weigh about 5 lb. per square foot of tail area. If the tail area is about one-third of the wing area and if the aircraft wing loading is, say, 100 lb./sq.ft., the tail weight is of the order of 0.02 W_o. This value has been assumed.

In cases where the aircraft might conceivably be of tailless design this value is still retained. Usually there is very little to choose between a tailed and a tailless configuration (where both are possible) because the structure weight saving in a tailless design is likely to be largely offset by increased trimming drag.

2.08 Wing Weight

The wing structure weight has been estimated using Drigg's wing weight equation (Reference 2). The actual formula given by Driggs was simplified for the present case by assuming that all of the likely wings would be highly tapered. The revised wing weight equation is:

$$\frac{W_w}{W_o} = \frac{0.809\ n\ b}{45000\ \cos\Lambda}\left[4.95 + \frac{0.065\ A}{\tau\ \cos\Lambda}\right]$$

where n = ultimate load factor (assumed to be 6.0)
b = wing span (feet)
Λ = wing structure sweep (assumed to be the sweep of the mid-chord line)
A = wing aspect ratio
τ = wing thickness-chord ratio

2.09 Fuel Weight

When all of the above items are added and subtracted from gross weight, the remainder is the available weight of fuel for cruising, W_f.

2.10 Range Equation

The aircraft is assumed to cruise at constant Mach number M, in such a way that W/σ remains constant, where W is aircraft weight and σ is the relative density at cruising altitude. Thus the altitude increases as fuel is consumed. This cruise procedure results in constant lift coefficient and hence constant ratio of drag to weight. Thus cruising drag decreases as altitude increases and at the same rate as the reduction of engine thrust with altitude. Hence maximum continuous power is required throughout the cruise. Under these circumstances it can be shown that:

$$R = \frac{Ma}{c\left(\frac{D_c}{W}\right)} \log_e\left(1 + \frac{W_f}{W_1}\right)$$

where R = range (miles)
- a = speed of sound
- c = specific fuel consumption (lb./lb.-hr.)
- D_c/W = cruising drag-weight ratio
- W_f = fuel weight available for cruise
- W_1 = aircraft weight at top of climb

Now it is a good approximation (for W_f/W_1 up to 0.6), to approximate this by the relation

$$R \approx \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times 0.83\frac{W_f}{W_1} \quad \text{if a is in miles per hour}$$

$$\text{or } R = 0.570 \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times \frac{W_f}{W_1} \quad \text{if a is in ft./sec.}$$

$$= 0.600 \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times \frac{W_{f_o}}{W_o}$$

where W_{f_o} is the total fuel load at take-off, such that

$$\frac{W_{f_o} - W_f}{W_o} = 0.05$$

The fuel weight fraction $\frac{W_{f_o}}{W_o}$ can be computed by adding up all other items of fractional weight and subtracting the sum from unity. However, before some of these items can be calculated (wing weight for example) it is necessary to know the wing aspect ratio, sweep and thickness ratio, as well as wing area, and cruising altitude. Similarly, these quantities must be chosen before a calculation of the cruising drag-weight ratio is possible. The choice of these quantities is discussed below.

2.11 Choice of Wing Aspect Ratio

The above considerations may be summarized by writing the range equation in the following functional form:

$$R = f(W_o, M, A, \Lambda_o, \tau, S, \sigma)$$

where Λ_o = wing leading edge sweep
σ = relative density associated with the cruising altitude at a particular point on the cruise.

All the other quantities are as previously defined.

The method of analysis used here was to fix the cruising Mach number M at a particular value and compute range R for several values of W_o usually ranging from 100,000 lb. to 500,000 lb. Thus at any one value of W_o and M, the above relation reduces to

$$R = f(A, \Lambda_o, \tau, S, \sigma)$$

Ordinarily it would be desirable to choose values for all of these variables such that R is a maximum for the given values of W_o and M. However, the computations involved would be prohibitive and not worth the effort in the present case at least. If the functional relation could be written down analytically, the optimum solution could be found in theory at least by solving the 5 equations:

$$\frac{\partial f}{\partial A} = 0, \frac{\partial f}{\partial \Lambda_o} = 0, \frac{\partial f}{\partial \tau} = 0, \frac{\partial f}{\partial S} = 0, \frac{\partial f}{\partial \sigma} = 0$$

The drag correlation data of References 3 and 4 do provide all the information necessary, together with the assumptions already made regarding weight items, to permit the function f to be so expanded, but it is so complex that partial differentiation is nearly hopeless, much less a solution of the resulting equations.

On the other hand, some degree of optimization is desirable since it is clearly impossible to try and guess simultaneous values of all of the above five variables which will guarantee something like the best possible range. Furthermore it is of interest to know what the optimum values of some of these variables are, at least approximately. For example if the optimum value of cruising altitude (represented by the variable σ) should turn out to be a low one, say less than 35,000 ft., then this in itself tends to rule out such a bomber from serious consideration as a threat.

Intuitively one expects that each one of these five variables has an optimum value, for fixed values of the other four. This can be seen by considering what happens at extreme low and extreme high values of each. Consider, for example, wing thickness-chord ratio τ. If all of the other four quantities are held fixed temporarily while τ is allowed to vary from very low to very high values it is obvious that at very low values no fuel can be carried because the wing structure weight becomes too high. At very high values of τ, wing weight is low and fuel can be carried, but the wing drag eventually becomes so large that the range decreases rapidly. Hence there is an optimum value of τ. Similarly for A, Λ_o, S, and σ.

Fortunately, for two of these variables, aspect ratio A, and leading edge sweep Λ_o, the correlation data in References 3 and 4 permit a choice to be made which is clearly not far from an optimum.

The correlation of drag due to lift of swept wings given in Reference 3 showed that to a reasonable degree of approximation, this parameter can be calculated (for uncambered wings) by the Busemann relation

$$\frac{dC_D}{dC_L^2} = \frac{\beta}{4} \times \frac{\beta A}{(\beta A - \frac{1}{2})}$$

where $\beta = \sqrt{M^2-1}$

It is noteworthy that this expression as developed by Busemann was meant to apply only to finite rectangular wings. The expression shows that dC_D/dC_L2 has a minimum value equal to $\beta/4$ and that the effect of aspect ratio is negligible if A is large. Hence there is no point in choosing an aspect ratio larger than some certain value, since the only result will be an increase in wing weight. It was therefore decided to choose aspect ratio so that dC_D/dC_L^2 is just 20 percent above the theoretical minimum, $\beta/4$. The above equation can be used to show that this results in the condition that $\beta A = 3$, and hence the aspect ratio is specified at each design cruising Mach number. When this condition is evaluated, the following values of wing aspect ratio are obtained for the design cruising Mach numbers assumed:

M =	1.2	1.4	1.6	1.8	2.0
A =	4.60	3.08	2.40	2.00	1.73

It should be emphasized that the correlation of data presented in Reference 3 applies only to uncambered wings and analysis showed that such wings develop little or no leading edge suction at supersonic speeds. In the present study it was decided to apply modifications to the M = 1.2 bomber in the form of area-rule drag savings, and wing camber. Consequently one case was worked out for the unmodified and uncambered bomber, and in this case drag due to lift was calculated from the Busemann formula. Later, computations were made assuming that a reasonably large fraction of the full theoretical leading edge suction was realized at the design lift coefficient. In this case the wing plan form was left unchanged, but a value of the span efficiency e was chosen equal to 0.6, instead of the value 0.35 which the above method of calculation predicts for the uncambered wing at M = 1.2.

For the subsonic bomber an aspect ratio of 6 was chosen arbitrarily.

2.12 Choice of Wing Leading Edge Sweep

The drag correlation for swept wings at supersonic speeds, which is contained in Reference 4, permits a crude but rational choice of wing leading edge sweep to be made. The correlation showed that provided the Mach lines are swept behind the trailing edge, but ahead of the leading edge, the wing thickness drag (wave drag) is given approximately by the relation:

$$\frac{C_{D_t}}{\tau^2} \tan \Lambda_0 = 5$$

On the other hand, when the tangent of the sweep of the Mach lines is greater than about 1.5 times the tangent of leading edge sweep, the following relation holds

$$\frac{C_{D_t}}{\tau^2} = \frac{7}{\beta}$$

Both of these expressions hold only for wings with "conventional" aerofoil sections.

The second expression shows that a straight, or nearly straight wing has a comparatively high drag at low supersonic speeds, and the first of the two expressions shows that it can be greatly reduced by wing sweep, at least up to a sweepback of 50°. On the other hand, sweepback much in excess of this causes a rapid increase in wing structure weight. As a guess, therefore, a sweep of 50° was chosen for low supersonic speeds. At the other end of the scale, at M = 2, the effects of sweep on thickness drag are small until the sweep exceeds 50°. However, the low aspect ratio already chosen means that even for a leading edge sweep of 50°, the penalty on wing structure weight is negligible because the structural sweep is small. For convenience, therefore, a leading edge sweep of 50° has been chosen for all of the supersonic bombers. It should be pointed out here that the drag correlation of Reference 4 failed to confirm that swept wings have higher drag than straight or nearly straight wings when the Mach lines are swept behind the leading edge, at least for wings with conventional aerofoil sections.

2.13 Choice of Thickness-Chord Ratio

For the longest range bombers of this series the wing weight is a considerably smaller fraction of gross weight than is the weight of fuel. Hence it might be expected that it is more important to save drag and fuel consumption than to save wing weight. In other words the optimum wing thickness-chord ratio may be such that the second term in the wing weight equation (see paragraph 2.08) is somewhat greater than the first term, which does not contain thickness ratio. In order to check this the range was calculated for a representative case (M = 1.6, W_o = 200,000 lb.) leaving only thickness-chord ratio a variable. It was found that for maximum range the thickness ratio was such that the thickness term in the weight equation was about 1.5 times the other term. This ratio was held constant for all other cases, and results in the following values of thickness ratio for the supersonic bombers:

Design Cruising Mach No:	1.2	1.4	1.6	1.8	2.0
Thickness-Chord Ratio:	0.0505	0.0306	0.0223	0.0178	0.0151

The very small wing thickness at high Mach numbers is interesting.

For the area rule bomber which cruises at M = 1.2, the wing thickness ratio was arbitrarily increased to 10 percent on the assumption that the wing wave drag could be cancelled by indenting the fuselage. For the subsonic bomber the thickness ratio was again taken to be 10 percent.

2.14 Calculation of Wing and Tail Zero Lift Drag Coefficient

In order to compute the cruising drag-weight ratio, it is necessary to estimate the zero lift drag of the aircraft. The assumptions regarding fuselage drag have been discussed previously (paragraph 2.02). For the wing a value of skin friction coefficient of 0.005 was assumed at supersonic speeds and 0.006 at subsonic speeds. The wing thickness drag was calculated from the relations given in paragraph 2.12, which are based on the empirical correlation of Reference 4. In estimating tail weight (paragraph 2.07) it was assumed that the tail area is about 30 percent of wing area. Thus the zero lift drag of the wing has been increased by 30% to include the tail drag.

In the case of the area rule bomber, a different procedure was followed. The wing skin friction drag coefficient was again taken to be 0.005, and this was increased by 30% as a tail allowance. However, the effective wing and tail wave drag coefficient was assumed to be zero, since the experimental evidence to date indicates that, apart from skin friction, the drag of a wing-body combination can be reduced approximately to that of the body alone, by suitable changes of body shape.

Before the total zero lift drag coefficient of the aircraft can be calculated, the wing area must be chosen because the fuselage size is fixed.

2.15 Choice of Wing Area and Design Cruising Altitude

The method used to choose near-optimum values of these two variables requires that they be dealt with together.

Strictly speaking, design cruising altitude is not a constant for any one aircraft, but varies throughout the flight in accordance with the assumption that W/σ remains constant (see paragraph 2.10). If W/σ remains constant it is meaningful to specify an effective initial cruising altitude at which the relative density is σ_0, where

$$\sigma_0 = \frac{W_0}{W/\sigma}$$

As pointed out in paragraph 2.10, an aircraft flying at constant Mach number and constant W/σ, cruises with a constant ratio of drag to weight.

As a first approximation it might be supposed that the optimum design cruising altitude would be such that total drag is a minimum (for constant wing area, wing configuration, Mach number and weight) since this would require minimum fuel consumption and since this variable has no effect on structure weight. In this case the drag due to lift (induced drag) would be equal to total profile drag. However, further consideration will show that if the design cruising altitude is lowered somewhat from this value, the cruising drag will increase slightly, but the required engine weight will decrease appreciably. Hence more fuel can be carried and range will be greater. Sample calculations of range versus design cruising altitude were carried out for the same case as previously used in obtaining a criterion for thickness ratio. It was found that the optimum design cruising altitude was such that total aircraft profile drag was about 1.5 times the induced drag. This ratio was retained.

Hence,

$$\frac{\sigma_o \rho_o M^2 a^2}{2W_o}\left[C_{D_b} S_b - 1.3 C_{D_{o_w}} S\right] = 1.5 \times \frac{2W_o}{\sigma_o \rho_o M^2 a^2 S} \times \frac{dC_D}{dC_L{}^2}$$

where ρ_o = sea level standard density

C_{D_b} = body drag coefficient

S_b = body frontal area

$C_{D_{o_w}}$ = wing zero lift drag coefficient

S = wing area

The other quantities in the above are as defined previously.

The choice of wing area would appear at first to be more complex because it has a direct effect on both drag and structure weight. However, once again a crude guess can be made, which can be checked by sample calculations. In the present group of aircraft the body drag is fixed if cruising altitude and Mach number are determined. Hence changes in wing area have no effect on this part of the drag. It might therefore be expected that the optimum wing area would be close to the value which gives minimum total wing and tail drag, i.e., where wing plus tail profile drag is equal to wing induced drag. In this condition the fuel consumption will be a minimum and also the engine weight will be a minimum (at constant altitude, aircraft weight, Mach number, etc.). As wing area decreases, wing weight also decreases although not rapidly, and because of this it is to be expected that the optimum wing area is actually a little lower than the first guess. However, the same sort of sample calculations as were carried out for thickness-chord ratio and cruising altitude showed that maximum range was achieved for a wing area such that wing plus tail profile drag was very nearly equal to (but slightly less than) wing induced drag. Hence, in the analysis for all bombers these two drag items were kept equal. In other words,

$$\frac{\sigma_o \rho_o M^2 a^2}{2W_o}\left[1.3C_{D_{o_w}} S\right] = \frac{2W_o}{\sigma_o \rho_o M^2 a^2 S} \times \frac{dC_D}{dC_L^2}$$

This equation, together with the one given above can be solved for σ_o and S, and the result is

$$S = \frac{2C_{D_b} S_b}{1.3C_{D_{o_w}}}$$

and

$$\frac{\sigma}{W} = \frac{\sigma_o}{W_o} = \frac{\sqrt{1.3\frac{dC_D}{dC_L^2} \times C_{D_{o_w}}}}{C_{D_b} S_b \rho_o M^2 a^2}$$

It will be noted that wing area is independent of design gross weight and cruising altitude decreases as gross weight increases.

2.16 Calculation of Range

When all of the above assumptions regarding weight and drag are gathered together, the following range equation can be written down:

$$\text{Range (miles)} = \frac{\frac{0.6Ma}{c}\left\{0.87 - \frac{25000}{W_o} - \frac{K_e}{\sigma_o}\left(\frac{D}{W}\right)_c - \frac{0.809n\sqrt{AS}}{45000 \cos\Lambda}\left[4.95 + \frac{.065A}{\tau\cos\Lambda}\right]\right.}{\left(\frac{D}{W}\right)_c}$$

where W_o = design gross weight

$K_e = \dfrac{\text{engine weight} \times \sigma}{\text{thrust}}$

$\left(\frac{D}{W}\right)_c$ = cruising drag-weight ratio

$= 2.5\sqrt{1.3\frac{dC_D}{dC_L^2} \times C_{D_{o_w}}}$

$$\frac{dC_D}{dC_L^2} = 0.30\sqrt{M^2-1}$$ (except in case of subsonic and cambered supersonic bombers)

c = specific fuel consumption (lb./lb.-hr.)

$$S = \text{wing area} - \frac{2C_{D_b}S_b}{1.3C_{D_{o_w}}}$$

$$\sigma_o = \frac{W_o\sqrt{1.3\frac{dC_D}{dC_L^2} \times C_{D_{o_w}}}}{C_{D_b}S_b\rho_o M^2 a^2}$$

This equation was evaluated for a range of weights from 100,000 to 400,000 lb. and a range of Mach numbers from 0.9 to 2.0.

3.0 DISCUSSION OF RESULTS

3.01 Aircraft Configurations

According to the methods outlined above for arriving at near-optimum configurations, the wing configuration and wing area are independent of design gross weight. Furthermore the fuselage dimensions were assumed to be fixed for all aircraft. Hence it is possible to sketch the plan views of the aircraft which result from the analysis, and these sketches will be a function only of design cruising Mach number. The aircraft configurations are shown in Figure 2. Two are drawn for a Mach number of 1.2. The upper one is the bomber which makes full use of the area rule for reducing profile drag and of wing camber for reducing drag due to lift. This is the largest aircraft of the group in terms of wing area, a fact which is explained by the equation for wing area developed in paragraph 2.15.

Some of these configurations have a peculiar appearance, to say the least. In practice they would probably vary considerably from those shown, because if these shapes produce nearly maximum range for a given gross weight it follows that range should not change greatly with relatively large changes in configuration.

3.02 Range

The calculated range as a function of design cruising Mach number and gross weight is plotted in Fig. 1. These ranges are based on turbo-jet engines having the following characteristics as suggested by Mr. Tyler of the Gas Dynamics Laboratory:

Mach Number	1.2	1.4	1.6	1.8	2.0
Engine weight per pound thrust at 50,000 ft:	1.385	1.16	1.01	0.93	0.97
Specific fuel consumption:	1.145	1.17	1.22	1.28	1.35

It will be noticed at once that the supersonic bombers which make no use of the area rule or wing camber have much lower still air ranges than the subsonic bomber. In general this is due to wave drag. As design cruising Mach number is increased above 1.2 there is at first an increase in range for a given gross weight. As the Mach number approaches 2, however, range again begins to decrease. It should be noted that the analysis took no account of the effects of aerodynamic heating on structure weight. These effects would become noticeable at Mach numbers slightly above 2.0 and at the same time the reduction of thrust of a turbo-jet engine would further decrease range in this area. It therefore appears that a Mach number of 2.0 represents an upper limit to the design of turbo-jet powered bombers in the foreseeable future.

The range of the "simple" supersonic bombers is of the order of 3000 miles at the largest weights, and it increases very little as gross weight is increased above 300,000 to 400,000 lb.

The question arises as to whether the range of a bomber could be increased if it flies most of the distance at subsonic speeds. It is reasonable to suppose, because of the relatively short range of intercepting devices, that the bomber may have little to fear over most of its mission and hence it would be sufficient to provide a burst of speed only during a few hundred miles. This question has not been examined at length in the present analysis, but it is clear that the optimum design for efficient supersonic flight is usually much different from that required for economical subsonic cruising. Although the maximum lift-drag ratios of the "simple" supersonic configurations are considerably lower than that of a good subsonic aircraft, the configurations designed for Mach numbers above about 1.6 would in themselves have poor subsonic efficiency.

The area rule bomber would have a high lift-drag ratio at subsonic speeds but the calculations indicate that its supersonic range may be nearly as great as that of a subsonic bomber in any case.

It is therefore concluded that the supersonic ranges shown in Fig. 1 could not be greatly increased by flying most of the distance at subsonic speeds.

The very large benefits due to employing camber and area rule modifications are clear from Figure 1. Although the calculations must be taken as representative of an ideal case, it is felt that they are not unrealistic. Wind tunnel results are available for a bomber configuration generally similar to the one considered here (Reference 5). At a Mach number of 1.15, these tests gave a maximum lift-drag ratio of 14.5. The methods of drag estimation used here predict a maximum lift-drag ratio of 15 for the cambered area rule bomber shown in Fig. 2, at a Mach number of 1.2.

3.03 Altitude over the Target

Although the cruising altitude over the target would normally be taken as one of the design specifications of a bomber, it has been chosen here only from the point of view of maximizing still-air range. In any practical case, therefore, if the required cruising altitude varies greatly from that shown in Fig. 1 for a specified range and cruising Mach number, the design would have to be compromised in such a way that gross weight would increase.

For the supersonic bombers, the altitudes over the target are generally greater than 50,000 ft. and in some cases (short ranges and high Mach numbers), over 70,000 ft.

For a given range the area rule bomber designed for M = 1.2 is heavier than the subsonic bomber, but can cruise much higher over the target.

4.0 CONCLUSIONS

The following conclusions are drawn from the above analysis, which considers the range possibilities for supersonic turbo-jet bombers carrying a payload of 10,000 lb.

(a) Still air ranges from 3000 to 5000 miles appear to be possible from the top of the initial climb for bombers designed to cruise at Mach numbers between 1.2 and 2.0. The operational radii would be about one-half of these values.

(b) If the benefits of the area rule and of wing camber are not made use of, the maximum still-air range remains approximately constant at about 3000 miles throughout the supersonic speed range.

(c) If, on the other hand, these recent aerodynamic refinements are fully applied, the range can be increased to about 5000 miles at least for design cruising Mach numbers of about 1.2.

(d) Since the potential benefits to be expected from these refinements tend to become small at Mach numbers above about 1.6, it is doubtful if the range can be increased greatly above 3000 miles at the upper end of the speed scale (up to M = 2.0).

(e) At Mach numbers above 2.0 two factors begin to come into effect which tend to reduce range. These are the effect of aerodynamic heating on structure weight, and the increase in turbo-jet specific weight.

(f) Very little increase in range is evident in all cases for increases in gross weight above 300,000 to 400,000 lb.

(g) Altitudes over the target tend to decrease as range (or gross weight) is increased, and as design cruising Mach number is decreased.

(h) Altitudes over the target are generally of the order of 50,000 feet for the supersonic bombers, and in some cases may be as high as 70,000 ft.

(i) The advantages to be gained by the full use of wing camber and the application of the area rule appear to be so great, at least for low supersonic Mach numbers, that this would seem to be not only a possible, but a very probable future trend in the development of long range bombers. An increase in cruising speed from, say, 0.9 to 1.2 increases greatly the difficulty of interception, at least by manned interceptors.

(j) Although no detailed consideration has been given here to the possibility of increasing bomber range by flying only a few hundred miles at supersonic speed, rough considerations indicate that little is to be gained. This question, however, possibly requires some analysis.

5.0 REFERENCES

1. Wood, A.D. — Brief Performance Estimates for Long Range Ballistic Rockets. NAE Lab. Memo. FR-30, 18 Jan., 1955.

2. Driggs, I.H. — Aircraft Design Analysis. Journ. Roy. Aero. Soc. Vol. LIV, 1950, page 65.

3. Callan, M.M., Templin, R.J. — A Collection and Analysis of Experimental Data on the Aerodynamic Characteristics of Finite Wings. Part I. Lift Curve Slope and Drag due to Lift of Swept Wings at Supersonic Speeds. NAE LR-93, January 1954.

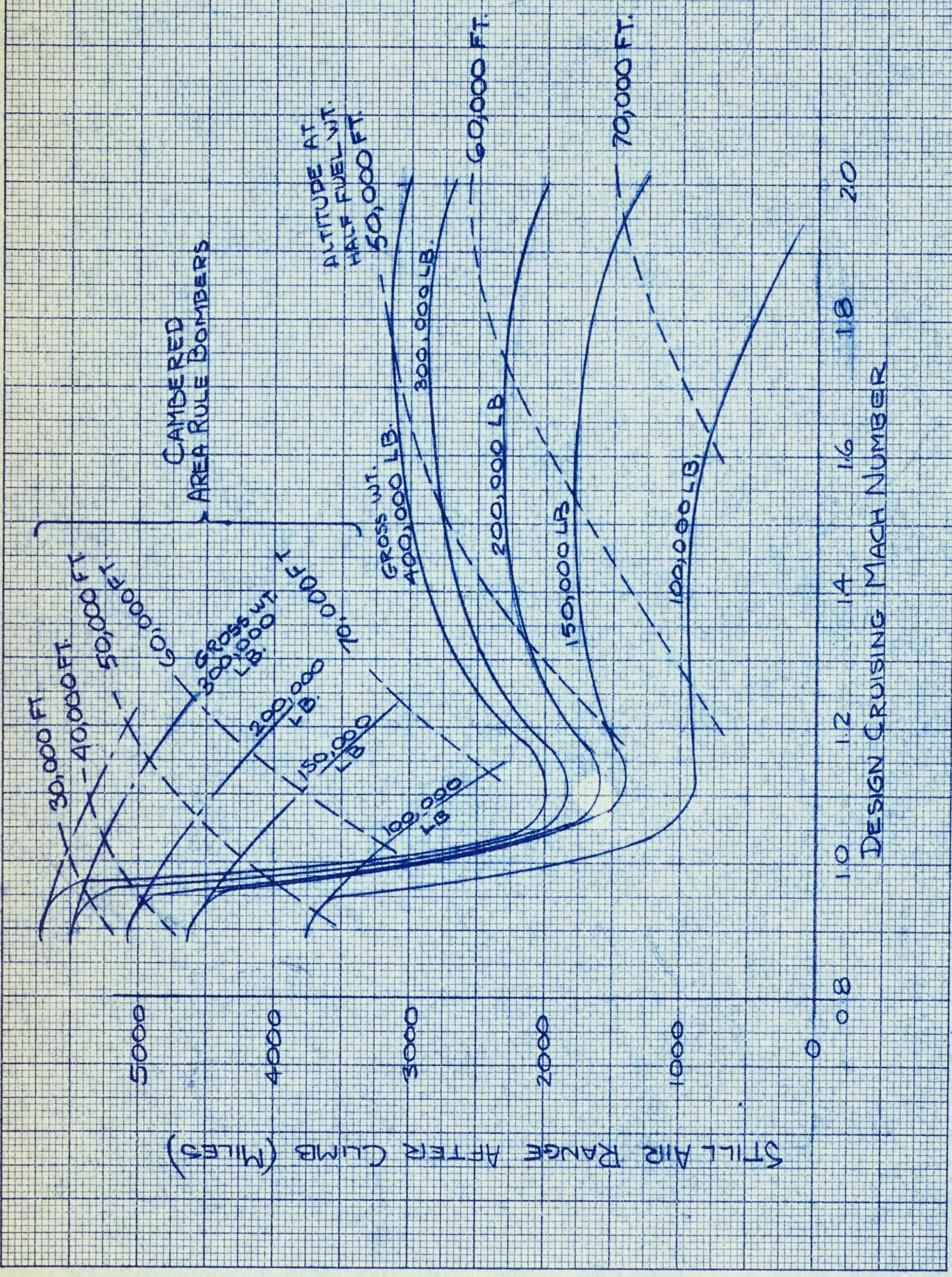
4. Callan, M.M., Templin, R.J. — A Collection and Analysis of Experimental Data on the Aerodynamic Characteristics of Finite Wings. Part II. Zero Lift Drag of Swept and Straight Wings at Supersonic Speeds. NAE LR-108, July 1954

5. Whitcomb, R.T., Fischetti, T.L. — Development of a Supersonic Area Rule and and Application to the Design of a Wing-Body Combination having High Lift-to-Drag Ratios. NACA RM L52H31a, Oct., 1953.

RANGE, WEIGHT, CRUISING ALTITUDE OF SUPERSONIC BOMBERS

FIG 1

PAYLOAD = 10,000 LB.



BOMBER CONFIGURATIONS

FIG. 2