71/AERO DATA/12

THE SPILLAGE DRAG AND EXTERNAL DRAG COEFFICIENT, ARROW I, IA.

NOV. 1958

W.B.McCARTER

AVRO AIRCRAFT LIMITED

MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

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THE DRAG OF THE ARROW 1

PART I SPILLAGE DRAG

PART II MINIMUM EXTERNAL DRAG COEFFICIENT

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AVRO AIRCRAFT LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT

REPORT NO.	71/AERO DATA/12
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SUMMARY

Included in this composite report is a new revised spillage drag characteristic which shows good agreement with both the Arrow I Intake Test results and for the Vigilante, an aircraft with similar intake geometry.

In addition, a new external drag curve is suggested which is consistent with all available W.T.M. and F.F.R.M. tests of the Arrow I and is in good agreement with that of the Hustler, and aircraft with definite family characteristics.

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AIRCRAFT:

GEOMETRY

Ref. P/Geom/32

30° conical nose.
J75 intakes, 5.6 sq. ft. throat.
39" cylindrical ejector.
L.E. droop, 5% notch, 10% extensions.
3 1/2% cambered wing, 1225 sq. ft. area, MAC 362.6 in.
4% tail.

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REPORT NO. 71/AERO DATA/12
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AIRCRAFT:

NOTATION

F_N	Net thrust force	lb.
D	Drag force	lb.
p	static pressure	psi
V	velocity	fps
m	mass flow	lb/sec.
q	$\frac{\gamma p_\infty M_\infty^2}{2}$	psi
M	Mach Number	–
γ	ratio of specific heats	–
C_D	drag coefficient	
A_c	capture area, projected frontal area of intake including compression surface	
S	lifting surface area	
C_p	pressure coefficient $\frac{p - p_\infty}{q_\infty}$	

Subscript

∞	at infinity
1	lip station
2	throat station
3	engine compressor face station
4	plane of engine final nozzle
*	choking
o	capture
f	friction, fin, fuselage (portion affected by streamtube only)
c	entering stream tube contour, the dividing streamline
R	external compression ramp
l	lip
d	duct
X	external
e	engine
w	wing

REFERENCES

1. RAE Dept. Aero 2380 — L.E. Fraenkel — External Drag of Some Pitot Type Intakes at Supersonic Speeds. Part I. — June 1950

2. NACA RM E56J01 — J.L. Allen — Performance of a Blunt lip side Intake with Ramp Bleed, Bypass, and a Long Constant Area Duct Ahead of the Engine M.66, M 1.5 to M 2.1 — Oct. 1956

3. NACA RM A56C06 — W.F. Davis F.T. Gowen — Change with Mass Flow Ratio of the Cowl Pressure Drag of Normal Shock Inlets at Supersonic Speeds — June 1956

4. P/Power/63 — W.B. McCarter — Design of the Ramp Bleed — Apr. 1956

5. NACA TN 3213 — G.E. Solomon — Transonic Flow Post Cone Cylinders — Sept. 1954

6. NACA TN 3457 — W.E. Moeckel — Estimation of Lip Forces at Subsonic and Supersonic Speeds — June 1955

7. RAE TN Aero 2315 — J. Seddon — Note on Spillage Drag of Pitot Type Air Intakes at Transonic Speeds — Aug. 1954

8. Douglas Report SM13747 — F.W. Graham — Notes on the Drag of Scoop and Blunt Bodies — Apr. 1950

9. NACA RM E51B13 — M. Sibulkin — Theoretical and Experimental Investigation of Additive Drag — May 1957

10. P/Aero Data/72 — J.A. Stein — Drag Analysis of CF-105 FFRM No. 6 — June 1956

11. P/Aero Data/66 — J.A. Stein — Drag Analysis of CF-105 FFRM — Oct. 1955

12. P/Wind Tunnel/131 — J. Clark — Summary of Wind Tunnel Testing on the CF-105 — Sept. 1956

13. P/Aero Data/70 — J. Stein — Drag Connections to the C105 Langley W/T Model — May 1956

14. P/Wind Tunnel/130 — D.J. Foster — Unitary Wind Tunnel Reduction Data — Dec. 1956

15.	P/Aero Data/58 R. Skulsky	C105 Drag Analysis of June 1955 CAL Wind Tunnel Tests	Oct. 1955
16.	P/F.F. Models/47 D. Ewart	C105 Summary of the First Seven Free Flight Model Tests and Results	Aug. 1957
17.	NACA RM L57G29 S. Hoffman	Free Flight Investigation of the Drag of a model of a 60° Delta-Wing Bomber (B58)	Sept. 1957
18.	NACA TN 4201 W.E. Stoney Jr.	Collection of Zero-Lift Drag Data on Bodies of Revolution from Free-Flight Investigations	Jan. 1958
19.	S.E. Hoerner	Fluid Dynamic Drag	Aug. 1957
20.	NAE-122 J.G. Laberge	Minimum Drag on the C105 1/80 Scale Model	May 1955
21.	NAE-AE-46h J.G. Laberge	Minimum Drag on the C105 1/80 Scale Model	Oct. 1956
22.	AFTR-AFFTC-54-14	Drag Coefficient, YF-102, USAF No. 52-7995	Dec. 1954
23.	NACA TN E53H19	Investigation at M 1.5, 1.7 of a Twin-Duct Side-Intake System with Two-Dimensional 6° Compression Ramps (F102-A)	
24.	NAE-AE-46d J.G. Laberge D.W. Boyer	Supersonic Wind Tunnel Test of the 1/40 Scale Intake Model of the C105 Aircraft	Oct. 1955
25.	7/Perf/41 D. Hague	Spillage Drag Calculations on the Arrow 1 Aircraft at M 1.5 by the Area Rule Method	Dec. 1958
26.	NAE-AE-46f J.A. Van du Bliek	High Speed Wind Tunnel Tests on a 1/50 Scale Half Model of the Avro CF105 Aircraft	June 1958
27.	P/Aero Data/74 S. Singer	Final Induced and Trim Drag From FFRM and W.T.M. Tests	Oct. 1956

INTRODUCTION

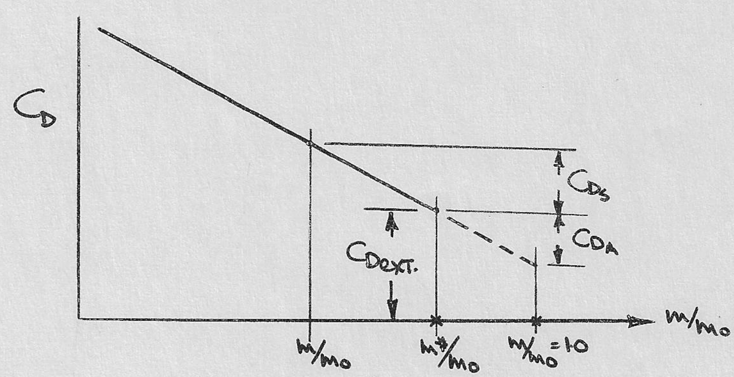
The drag of an aircraft is probably the most controversial force that is required to assess A/C capability. Controversial because it is difficult to estimate with accuracy because of interference effects, and difficult to measure with accuracy with a W/T model, Free Flight model, or even on a full scale flight test aircraft because of the relative order of the necessary corrections.

Now that all model testing is complete this review of aircraft drag is an attempt to correlate all the many contributions, witting and unwitting, to the drag picture. This review was attempted to give greater confidence in the drag used to predict Arrow 1 performance, on the eve of its first flight, and at the same time allow an extrapolation for use on the Arrow 3.

Drag is normally broken down into two parts, external and internal. By definition on the Arrow 1, 2 the demarcation is the dividing streamline when the intake is running full, i.e. a choked throat. This definition allows the simplicity of a fixed external drag (i.e. does not alter with engine RPM) and minimizes the corrections to drag models where the intake flow can be simply fixed at the choke value.

Because of the increase in throat area between the Arrow 1 and Arrow 2, the external drag would by this definition be decreased. However for simplicity on the Arrow 2, the same external drag is used and the internal drag reduced.

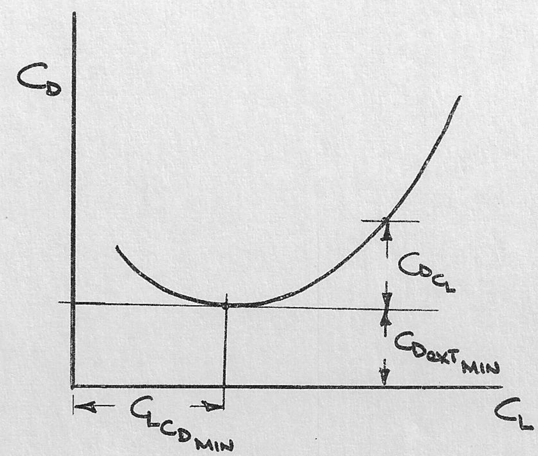
It follows, then, that the drag at zero lift is made up of two parts, a fixed portion, the external drag, and a portion varying with engine RPM, the internal drag, which in turn is usually termed in the literature additive or spillage drag.



In this report the spillage drag is defined as the drag increment due to spilled intake flows where the intake is running at less than choke mass flow (i.e. the internal drag) and the additive drag is defined as that portion of the external drag due to spilled intake flows when the intake is running full.

INTRODUCTION (Cont'd)

The contributions to the total drag of an aircraft, then, are minimum external drag (which by definition includes additive drag), drag due to lift, (g forces) trim drag (control forces) and spillage drag. It is implied that the contributions can be separated out and do not affect one another, i.e. for example that there is no change in C_{D_s} or $C_{D_{trim}}$ with C_L.



$$C_D = C_{D_{ext_{min}}} + C_{D_{C_L}} + C_{D_{trim}} + C_{D_s}$$

when $C_{D_{min}} = f(M)$

$$C_{D_{C_L}} = f(M, C_L) = f(M, n, W, p_\infty)$$

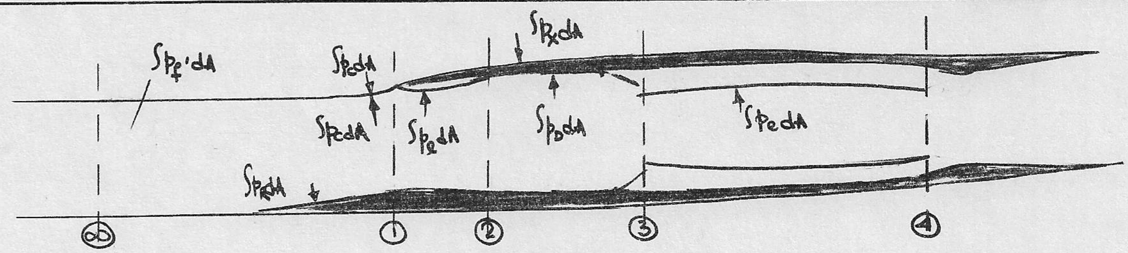
$$C_{D_{trim}} = f(M, \delta)$$

$$C_{D_s} = f(M, m)$$

This report is confined to $C_{D_{ext_{min}}}$ and C_{D_s}.

Because $C_{D_{ext_{min}}}$ has been measured with a cold jet and since the contribution of the ejector to internal thrust has been measured with a cold jet in quiescent air, the mutual effect of the jet on boattail pressures and of the external stream on ejector internal pressures has not been analysed in this report. This effect will be significant only at Mach Numbers above and below the ejector design point (i.e. the design point is the Mach Number where the exit static pressure is just ambient). For our aircraft with an ejector design joint of $M \doteq 1.7$ this will be especially significant at transonic speeds where because of mixed subsonic-supersonic flows the solution can be given only by very expensive test. This afterbody drag analysis is to be given in a separate report.

The assistance of D. Garland, D. Ewart and G. Dimock of the Stability and Control Section is gratefully acknowledged. The assessment of model data, which was, primarily, stability and control model data, would have been impossible without their assistance and critical comments.



Net thrust as defined in engine brochure:

$F_N{}' = m_e (V_4 - V_\infty) + (p_4 - p_\infty) A_4$

$= +\int p_c \, dA - \int p_R \, dA + \int p_d \, dA - \int p_1 \, dA + \int p_e \, dA$

Internal Drag

$D_{int} = -\int p_c \, dA + \int p_1 \, dA - \int p_d \, dA + \int p_R \, dA + \int p_f{}' dA$

External Drag

$D_{ext} = +\int p_c \, dA + \int p_x \, dA$

True Net Thrust of Engine:

$F_N{}'' = m(V_4 - V_3) + (p_4 - p_\infty) A_4 - (p_3 - p_\infty) A_3$

$= +\int p_e \, dA$

$\therefore$ Total internal force:

$F_N{}'' - D_{int} = m (V_4 - V_3) + (p_4 - p_\infty) A_4 - (p_3 - p_\infty) A_3 + \int p_1 \, dA - \int p_e dA$
$+ \int p_d \, dA - \int p_R \, dA - \int p_f{}' dA$

$= F_N{}' + m(V_\infty - V_3) - (p_3 - p_\infty) A_3 + \int p_c \, dA - \int p_1 \, dA$
$+ \int p_d \, dA - \int p_R \, dA - \int p_f{}' dA$

But $m (V_3 - V_\infty) + (p_3 - p_\infty) A_3 = \int p_c \, dA - \int p_1 \, dA + \int p_d \, dA - \int p_R \, dA - \int p_f{}' dA$

$\therefore F_N{}'' - D_{int} = F_N{}'$

However, included in the external drag is the term $\int p_c \, dA$.

Now: $m(V_1 - V_\infty) + (p_1 - p_\infty) A_1 = \int p_c \, dA - \int p_R \, dA - \int p_f{}' dA$

$\therefore \int p_c \, dA = m(V_1 - V_\infty) + (p_1 - p_\infty) A_1 + \int p_R \, dA + \int p_f{}' dA$

By the definition given in the introduction for the division of this contribution to the aircraft drag we have:

$$\Delta D_s = (\int p_c \, dA)_{m=m^*} - (\int p_c \, dA)_m$$

$$= m^* (V_1^* - V_\infty) - m(V_1 - V_\infty) + (p_1^* - p_1) A_1$$

$$+ \int (p_R^* - p_R) dA + \int (p_f'^* - p_f') dA - \left[\Delta \int p_x \, dA\right]_{m^* - m}$$

$$\Delta \, Dext = \int (p_c \, dA)_{m=mo} - \int (p_c \, dA)_{m=m^*}$$

$$= m_o (V_1^o - V_\infty) - m^* (V_1^* - V_\infty) + (p_1^o - p_1^*) A_1$$

$$+ \int (p_R^o - p_R^*) dA + \int (p_f^{1o} - p_f^{1*}) dA - \left[\Delta \int p_x \, dA\right]_{mo - m^*}$$

Where each has been increased by a term signifying the change of external pressures attendant on change in stream tube contour, commonly termed the lip suction relief. This is a small thrust due to increased lip suction on rounded lips due to local velocity increases at reduced intake mass flow.

In coefficient form:

$$C_{D_s} = \Delta D_s / q S_w \qquad \frac{m_o}{q_\infty} = \frac{2A_c}{V_\infty}$$

$$C_{D_s} = \frac{m^*}{m_o} \left(\frac{V_1^*}{V_\infty} - 1\right) \frac{2A_c}{S_w} - \frac{m}{m_o} \left(\frac{V_1}{V_\infty} - 1\right) \frac{2A_c}{S_w} + \left(\frac{p_1^*}{p_\infty} - \frac{p_1}{p_\infty}\right) \frac{2}{\gamma M_\infty^2} \frac{A_1}{S_w}$$

$$+ \left(\frac{p_R}{p_W} - \frac{p_R}{p_\infty}\right) \frac{2}{\gamma M_\infty^2} \frac{(A_c - A_1)}{S_W} + \left(\frac{p_f}{p_\infty} - \frac{p_f}{p_\infty}\right) \frac{2}{\gamma M_\infty^2} dA - C_{D_{L_S}}$$

For an open nosed pitot intake this reduces to:

$$A_c = A_i$$

$$C_{D_s} = \frac{m^*}{m_o} \left(\frac{V_1^*}{V_\infty} - 1\right) \frac{2A_c}{S_w} - \frac{m}{m_o} \left(\frac{V_1}{V_\infty} - 1\right) \frac{2A_c}{S_w} + \left(\frac{p_1^*}{p_\infty} - \frac{p_1}{p_\infty}\right) \frac{2}{\partial M_\infty^2} \frac{A_c}{S_w}$$

$$= \frac{2A_c}{S_w} \quad \frac{m^*}{m_o} \left(\frac{V_1^*}{V_\infty} - 1\right) - \frac{m}{m_o} \left(\frac{V_1}{V_\infty} - 1\right) + \left(\frac{p_1^*}{p_\infty} - \frac{p_1}{p_\infty}\right) \frac{2}{\partial M_\infty^2} - C_{D_{L_s}}$$

Fraenkel in Ref. No. 1 has indicated that for an open-nosed intake with full lip suction:

$$C_{D_s} = 2 \frac{A_c}{S_w} \left[\frac{m^*}{m_o} \left(\frac{V_w}{V_\infty} - 1\right) - \frac{m}{m_o} \left(\frac{V_w}{V_\infty} - 1\right) \right]$$

$$\text{or} \quad \frac{\partial C_{D_s}}{\partial m / m_o} = -2 \frac{A_c}{S_w} \left(\frac{V_w}{V_\infty} - 1\right)$$

or for a thin, sharp, open-nosed intake:

$$\frac{\partial C_{D_S}}{\partial m/mo} = 2\,\frac{A_c}{S_W}\left(\frac{V_1}{V_\infty} - 1\right)$$

This agrees with Ref. Nos. 6,7,8,9.

For the C105 intake, the assumptions made to fit a curve through the experimental points at M 1.5, 1.7, 2.0 of Ref. No. 2 are:

a) All spillage is through the normal shock only, with slope given for $m = m^*$.
b) Change of ramp drag to be found from Schlieren photos of our intake at Lewis.
c) Change of fuselage drag with engine mass flow assumed negligible.
d) Lip suction to be taken from Ref. No. 3.

$$\therefore \frac{\partial C_{D_S}}{\partial m/mo} = 2\,\frac{A_c}{S_W}\left(\frac{V_1}{V_\infty} - 1\right)_{m=m^*} + \frac{\partial C_{D_R}}{\partial m/mo} + \frac{\partial C_{D_{L_S}}}{\partial m/mo}$$

(a) First term is shown on sheet 16.
(b) $C_{D_R} = \left(\frac{p_{R2} - p_{R1}}{p_\infty}\right)\frac{2}{\partial M_\infty^{\,2}}\;\frac{d \times k \sin\delta}{144\;S_W} = C_p\,\frac{A'}{S_W}$

where p_{R_2} = ramp static pressure behind terminal normal shock wave.
p_{R_1} = ramp static pressure behind oblique shock wave from leading edge of ramp.
d = distance terminal normal shock waves forward, in inches, with reduced mass flow from choking.
k = depth of ramp, inches (50").
δ = ramp angle. (12°)
$C_p = \frac{p_{R2} - p_{R1}}{p_\infty}\;\frac{2}{\partial M_\infty^{\,2}}$

A' = area over which increased pressure through T.N.S.W. acts $= \left(\frac{dk \sin\delta}{144}\right)$ □"

M_∞	p_{R1}/p_∞	p_{R2}/p_{R1}	$\frac{p_{R2}-p_{R1}}{p_\infty}$	$\frac{2}{\partial M_\infty^2}$	C_p
1.495	2.03	1.0	0		
1.51	1.92	1.0	0	.628	0
1.6	1.828	1.375	.685	.5575	.382
1.7	1.825	1.70	1.278	.494	.631
1.8	1.83	2.02	1.867	.441	.823
1.9	1.863	2.35	2.515	.3955	.994
2.0	1.89	2.67	3.16	.357	1.13
2.5	2.09	4.50	7.31	.2285	1.671
3.0	2.34	6.60	13.10	.1588	2.08

$$\frac{p_{R2} - p_{R1}}{p_\infty} = \frac{p_{R1}}{p_\infty}\left(\frac{p_{R2}}{p_{R1}} - 1\right)$$

$$\frac{\partial C_{D_R}}{\partial m/mo} = \left(\frac{C_p}{S_w}\right)\left(\frac{k \sin\delta}{144}\right)\frac{\partial d}{\partial m/mo}$$

where $\partial d/\partial\, m/_{mo}$ may be found from Ref. No. 4, Shts.16-18

M	$\frac{\partial d}{\partial m/mo}$	C_p (Schlieren)	$\frac{\partial C_{D_r}}{\partial m/mo}$
1.5	55.9	.337	.033
1.7	22.5	.631	.025
2.0	21.4	1.130	.043

As indicated on sheet 17, the curve for $\partial C_{D_R}/\partial\, m/_{mo}$ is given above M 1.7 by factoring the C_p curve and extrapolating to M 1.0.

(c) $\frac{\partial C_{D_{L_s}}}{\partial m/mo}$ Ref: No. 3, Fig. No. 12.
Altered by factor $A_c/A_f = \frac{10.18}{40.98} = .248$

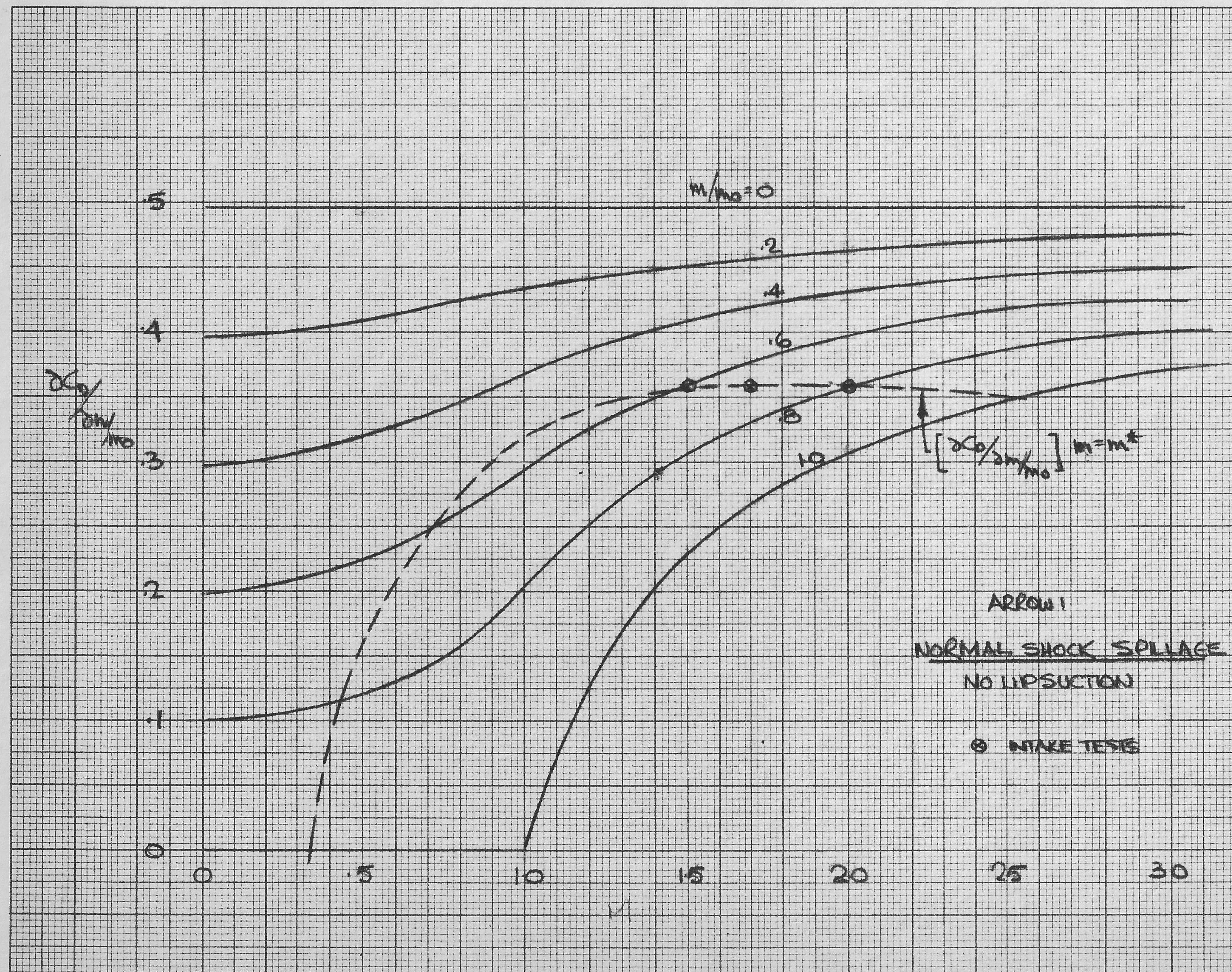
M_{os}	$\partial C_{D_{L_s}}/\partial m/mo$ Model No. 3	Model No. 6
1.34	.050	-
1.41	.050	.053
1.51	.030	.052
1.6	.040	.048
1.71	.037	.055
1.88	.029	-

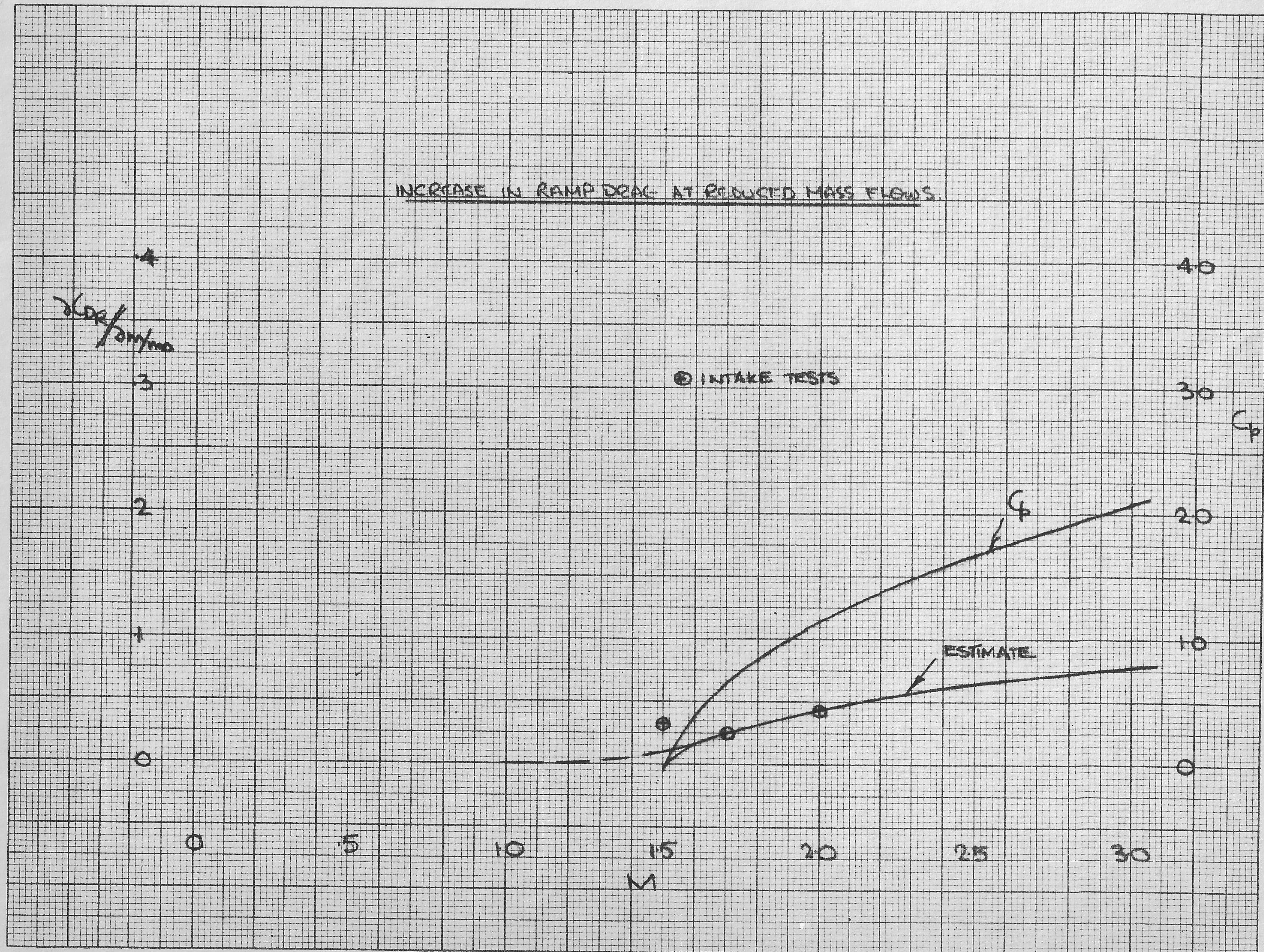
As indicated on sheet 18, the results of Ref. No. 3 are of the same order as given by the theory of Fraenkel in Ref. No. 1 for $m/mo = .9$.

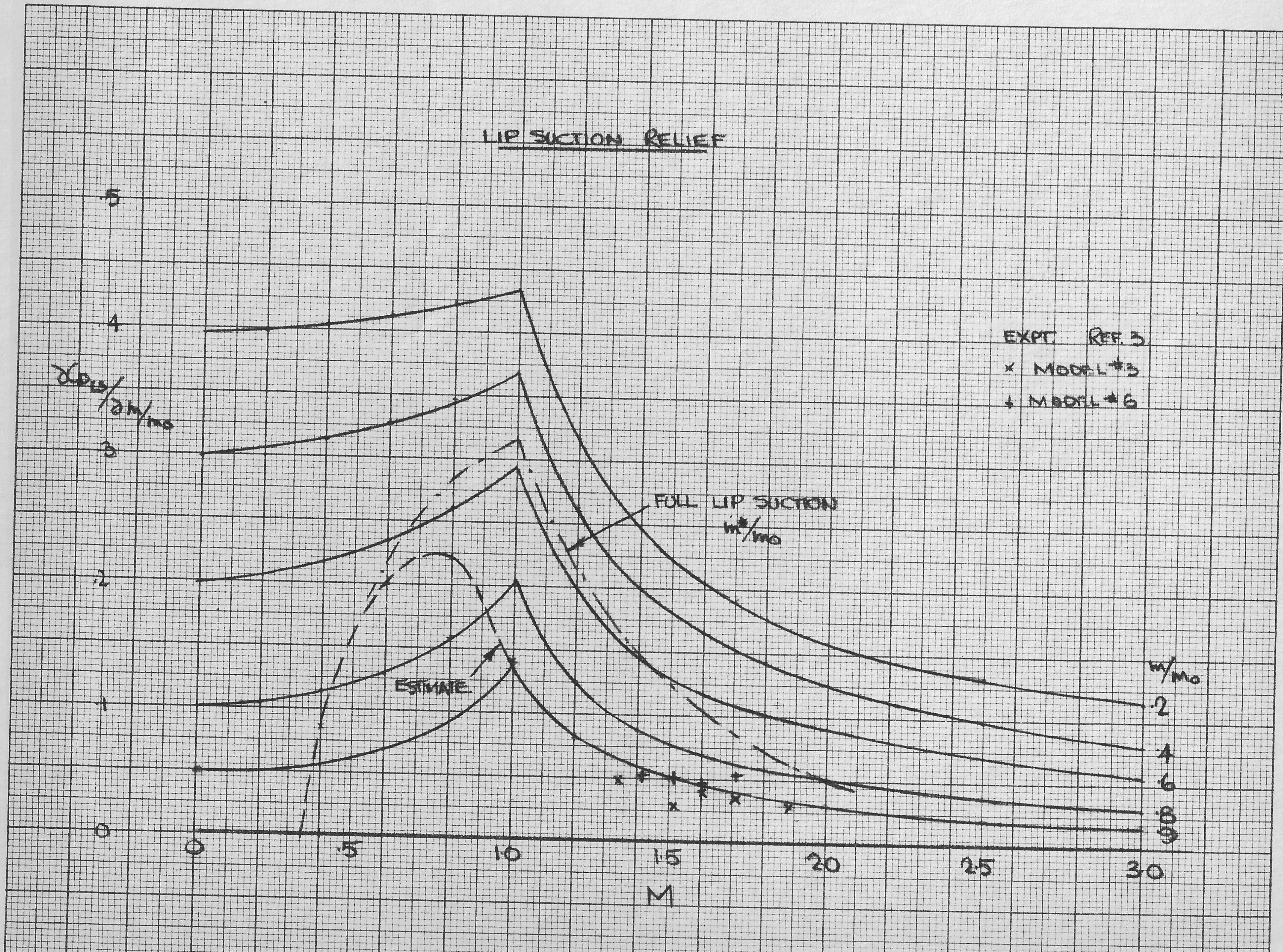
$$\text{where } \frac{\partial C_{D_{L_s}}}{\partial m/mo} = 2\,\frac{A_c}{S_w}\left(\frac{V_1}{V_\infty} - \frac{V_w}{V_\infty}\right)_{m/mo\,=\,.9}$$

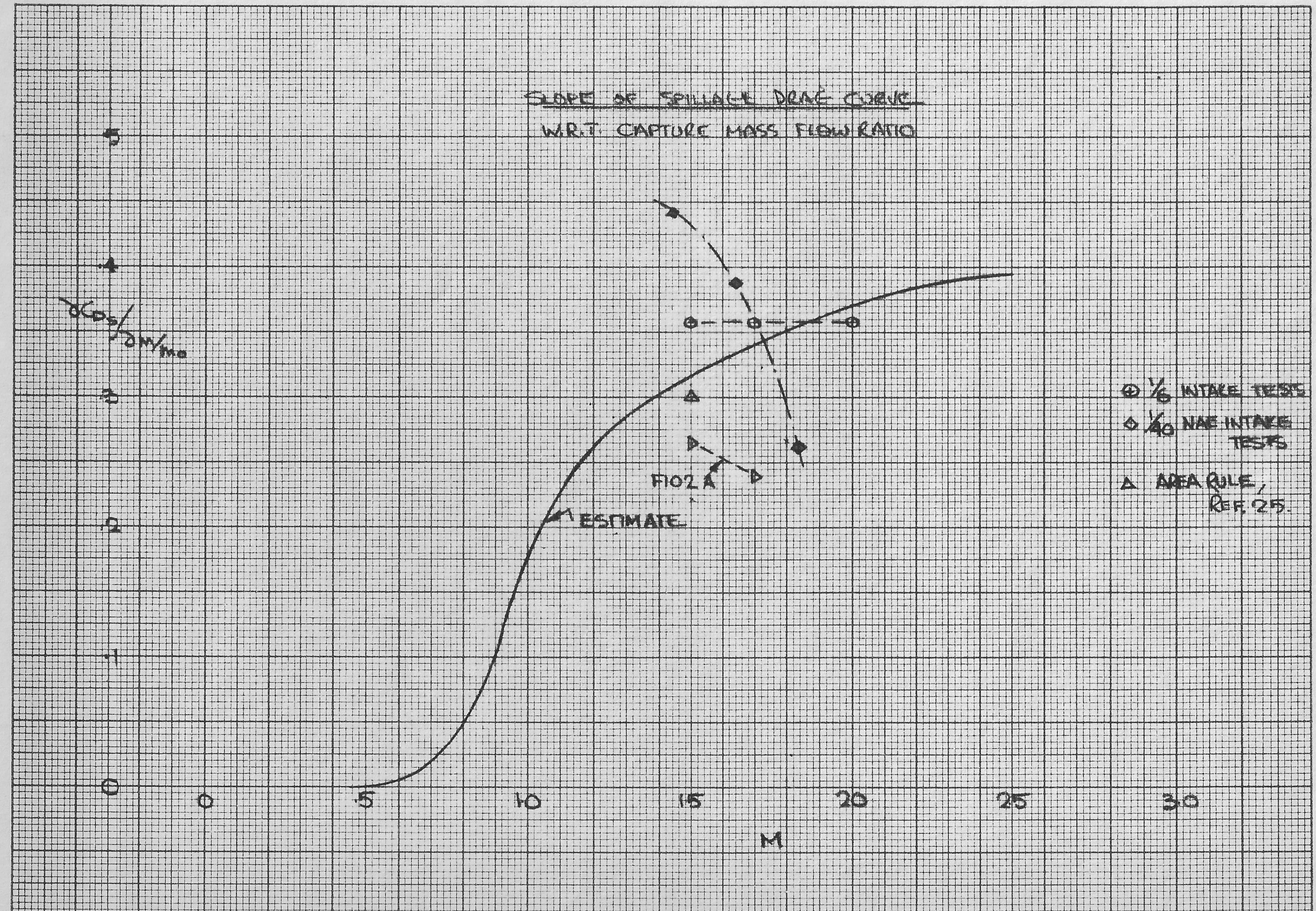
Summary

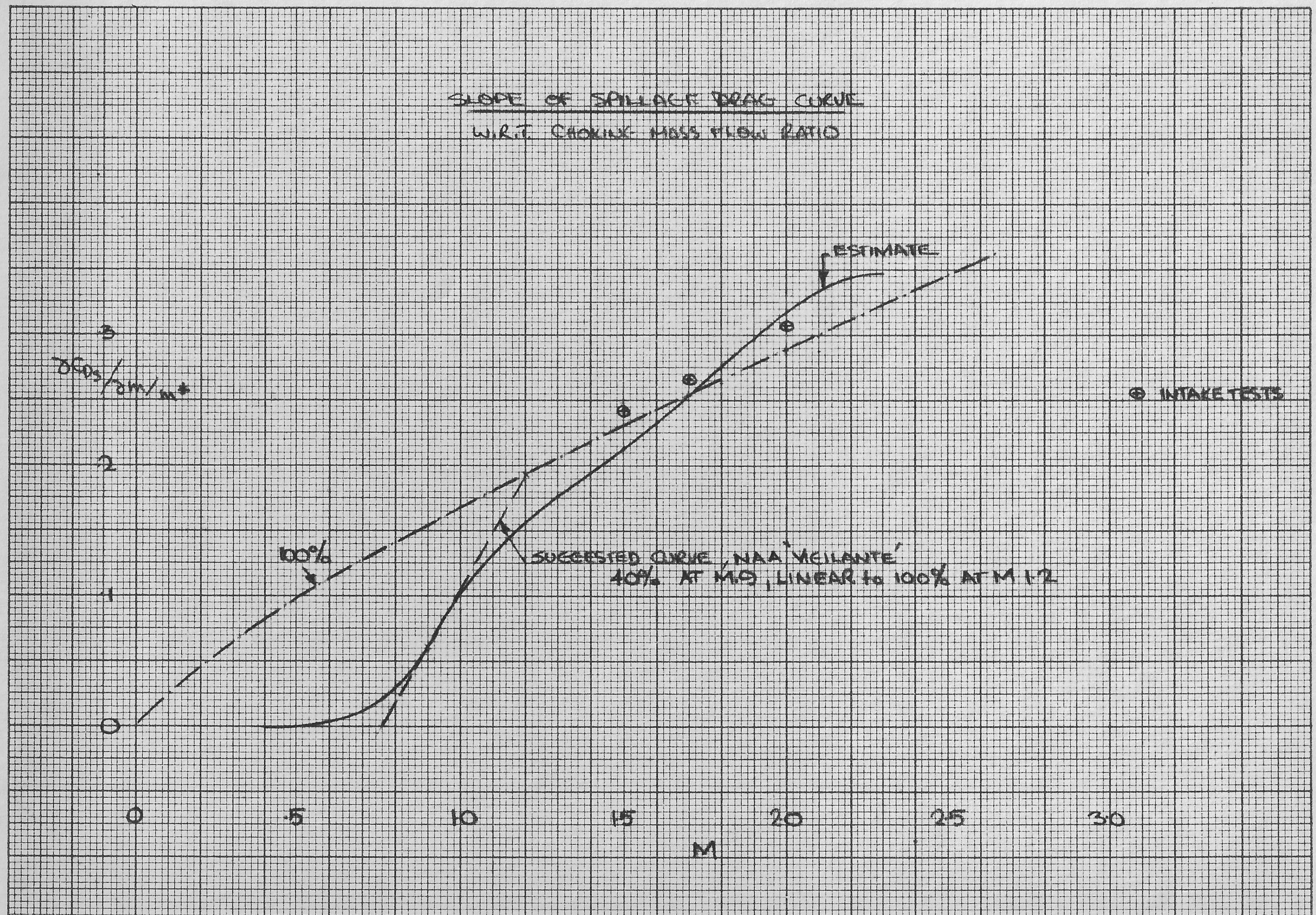
M_∞	$\partial C_{D_s}/\partial m^*/mo$	$\frac{\partial C_{D_r}}{\partial m/mo}$	$\frac{\partial C_{D_{L_s}}}{\partial m/mo}$	$\partial C_{D_s}/\partial m/mo$
.34	0	0	0	0
.4	.086	0	.086	0
.6	.205	0	.200	.005
.8	.272	0	.222	.050
1.0	.317	0	.138	.179
1.2	.342	.001	.083	.260
1.4	.355	.006	.060	.301
1.6	.358	.017	.046	.329
1.8	.358	.031	.036	.353
2.0	.358	.043	.030	.371
2.5	.350	.065	.020	.395


m/mo = 0
·2
·4
·6
·8
1·0
$\partial C_D / \partial m/m_0$
$[\partial C_D / \partial m/m_0]_{m=m^*}$
·5
·4
·3
·2
·1
0
0
·5
1·0
1·5
2·0
2·5
3·0
M
ARROW I
NORMAL SHOCK SPILLAGE
NO LIPSUCTION
⊗ INTAKE TESTS


INCREASE IN RAMP DRAG AT REDUCED MASS FLOWS
⊕ INTAKE TESTS
$\partial C_{DR} / \partial m/m_0$
·4
·3
·2
·1
0
C_p
4·0
3·0
2·0
1·0
0
C_p
ESTIMATE
0
·5
1·0
1·5
2·0
2·5
3·0
M


LIP SUCTION RELIEF
EXPT. REF. 3
× MODEL #3
+ MODEL #6
FULL LIP SUCTION
m*/mo
ESTIMATE
m/mo
·2
·4
·6
·8
·9
0
·1
·2
·3
·4
·5
0
·5
1·0
1·5
2·0
2·5
3·0
M


SLOPE OF SPILLAGE DRAG CURVE
W.R.T. CAPTURE MASS FLOW RATIO
∂CDS/∂m/mo
.5
.4
.3
.2
.1
0
0
.5
10
15
20
25
30
M
ESTIMATE
F102A
⊕ 1/6 INTAKE TESTS
◇ 1/40 NAE INTAKE TESTS
△ AREA RULE, REF. 25.


SLOPE OF SPILLAGE DRAG CURVE
W.R.T. CHOKING MASS FLOW RATIO
ESTIMATE
⊕ INTAKE TESTS
100%
SUGGESTED CURVE, NAA 'VIGILANTE'
40% AT M.9, LINEAR to 100% AT M 1.2
0
1
2
3
0
·5
1·0
1·5
2·0
2·5
3·0
M

Experimental Data leading to the estimate of $C_{D_{Min}}$ for the Arrow I with extended L.E., Droop and Notch.

Data analysed:

	Source	M	Symbol	Remarks
(a)	.04 CAL	.5 (discreet M) .8 .9 .95 .98 1.0 1.05 1.1 1.15 1.23	X	May 1955 3'x4' Transonic Tunnel
(b)	.03 Langley	1.41	◇	April 1956 4'x4' SSWT
(c)	.03 Langley	1.6 1.8 2.0	▣	4'x4' Unitary Tunnel July 1956
(d)	Area Rule	1.5	▲	F.A. Woodward March, 1957
(e)	Free Flight, NACA, Wallops Island, Virginia	1.0 to 1.6 (Continuous M)	⊖	May 9, 1956

(a) .04 Scale Cornell

$$C_{D_{Ext}} = C_{D_{X_{Sting}}} + C_{D_C} + C_{D_M} - C_{D_B} - C_{D_S} - \Delta C_{D_N}$$

$C_{D_{X_{Sting}}}$ and C_{D_C} were measured on the model. Unfortunately, however, because the model was essentially for evaluation of stability and control data, only 'representative' internal flows were used.

$$C_{D_c} = \frac{(p_c - p_\infty)}{q_\infty \; S_W} = C_p \left(\frac{A_c}{S_W}\right) = C_p \left(\frac{.0069}{1.103}\right)$$

Ratio taken from .03 scale model

M	C_p (Measured)	C_{D_c}
.50	.233	+.0015
.80	.268	+.0017
.90	.280	+.0018
.95	.297	+.0019
.98	.300	+.0019
1.00	.301	+.0019
1.05	.280	+.0018
1.10	.317	+.0020
1.15	.316	+.0020
1.23	.335	+.0021

$$C_{D_m} = \frac{m(V_4 - V_\infty)}{q_\infty\ S_w} + \frac{(p_4 - p_\infty)}{q\ S_w}$$

$$= \frac{m}{mo}\left(M_4 \frac{a_4}{a_o} - M_\infty \frac{a_\infty}{a_o}\right) \frac{2}{M_\infty{}^2} \frac{A_c}{S_w} \frac{H_\infty}{p_\infty} \varphi(M_\infty)$$

$$+\left(\frac{p_4}{H_4}\frac{H_4}{p_\infty} - 1\right)\frac{A_4}{S_w}\frac{2}{\partial M_\infty{}^2}$$

$$= \frac{m}{mo}\left(\frac{M_4}{M_\infty}\left(\frac{a_4}{a_o}\right)\left(\frac{a_o}{a_\infty}\right) - 1\right) 2\frac{A_c}{S_w} + \left(\frac{p_4}{H_4}\frac{H_4}{p_\infty} - 1\right)\left(\frac{A_4}{S_w}\right)\frac{2}{\partial M_\infty^2}$$

I (momentum term) II (pressure term)

If we assume the exit area is choked:

$M_4 = 1.0$

$m/mo = .5787\left(\frac{A_4}{A_c}\right)\left(\frac{H_4}{H_\infty}\right) \varphi(M_\infty)$

$a_4/a_o = .9129$

$p_4/H_4 = .5283$ $\qquad p_4/p_\infty = \frac{p_4}{H_4}\frac{H_4}{H_\infty}\frac{H_\infty}{p_\infty}$

$A_4/A_c = .655$ (taken from .03 Scale Model) $\qquad \frac{A_4}{S_w} = \frac{2 \times .85}{1.103 \times 144} = .0107$

$H_4/H_\infty = \left(\frac{H_4}{H_i}\right)\left(\frac{H_i}{H_\infty}\right)$ where $H_i/H_\infty \doteqdot .99$

Assume H_4/H_i losses are the sum of frictional and **expansion** losses, where

$$(H_4/H_i)_f = 1 - f\ L/D\ \frac{q}{H_\infty}\left(\frac{H_\infty}{H_i}\right)$$

$$= 1 - f\left(\frac{L}{D}\right)\frac{\partial}{2}\left(\frac{p_\infty}{H_\infty}\right) M_\infty{}^2 \left(\frac{H_\infty}{H_i}\right)$$

M_∞	q_∞	R_N/ft. x 10^6	R_N/D	f	p_∞/H_∞	$(H_4/H_i)_f$	$(H_4/H_i)_{exp}$	H_4/H_i
.5	123	1.23	148000	.018	.843	.962	.937	.899
.8	262	1.68	202000	"	.656	.924	.875	.799
.9	305	1.73	208000	"	.591	.913	.858	.771
.95	324	1.79	215000	"	.560	.909	.850	.759
.98	335	"	"	"	.541	.904	.843	.747
1.0	341	"	"	"	.528	.904	.843	.747
1.05	357	1.81	217000	"	.498	.901	.837	.738
1.1	371	1.84	221000	"	.468	.897	.832	.729
1.15	382	"	"	"	.440	.895	.827	.722
1.2	389	"	"	"	.412	.893	.824	.717
1.23	393	"	"	"	.396	.892	.822	.714

$\frac{L}{D} \doteqdot 14.2$

Expansion Losses:

$(H_4/H_i)_{exp} = 1 - .42g$, as estimated by J. Stein.

M_∞	$\varphi(M_\infty)$	a_∞/a_o	m/m_o	$\frac{M_4 A_4}{M_\infty A_\infty} - 1$	I	p_4/p_∞	$\frac{A_4}{S_W}\ \frac{2}{\partial M_\infty^2}$	II	C_{D_M}
.5	.432	.976	.780	.870	.0222	.557	0612	-0271	-0049
.8	.557	.942	.537	.211	.0037	.638	0239	-0087	-0050
.9	.574	.928	.504	.092	.0015	.683	0189	-0060	-0045
.95	.577	.920	.493	.045	.0007	.710	0169	-0049	-0042
.98	.578	.916	.485	.016	.0003	.722	0159	-0044	-0041
1	.579	.913	.484	0	0	.740	0153	-0040	-0040
1.05	.579	.905	.478	-.038	-.0006	.776	0139	-0031	-0037
1.1	.574	.897	.476	-.075	-.0012	.814	0126	-0023	-0035
1.15	.571	.890	.474	-.107	-.0017	.858	0116	-0016	-0033
1.2	.562	.881	.478	-.137	-.0021	.911	0106	-0009	-0030
1.23	.560	.876	.478	-.153	-.0024	.943	0101	-0006	-0030

$$C_{D_s} = \frac{\partial C_{D_s}}{\partial m/mo} \left(m^*/mo - m/mo \right) \frac{A_f}{S_w} \times 2$$

M	$\partial C_{D_s} / \partial m/mo$	m^*/mo	$\Delta m/mo$	C_{D_s}
.5	0	.716	-.064	0
.8	.050	.554	.017	0001
.9	.100	.540	.036	0002
.95	.137	.536	.043	0004
.98	.160	.535	.050	0005
1	.179	.534	.050	0006
1.05	.201	.535	.057	0007
1.1	.226	.538	.062	0009
1.15	.245	.542	.068	0011
1.2	.260	.548	.070	0012
1.23	.266	.552	.074	0013

The base drag correction will be assumed to be identical to that given by the FFRM No. 6 above M = 1.0, and zero below M = 1.0.

ΔC_D due to modification of nose from a 50° included angle to 30°.

Assumption made is that above M = 1.41 the Taylor - Maccoll Theory is used but factored through the one experimental value from the 03 CAL model.

Below M = 1.0 the experimental evidence of Ref. 5 was used.

The cone base area assumed had a diameter of 40".

M	p_s/H_∞ $\theta = 25°$	$\theta = 15°$	$\Delta p_s/H_\infty$	p_∞/H_∞	ΔC_{D_N}
1.33	.671	.463	.208	.347	.0035
1.5	.521	.376	.145	.272	.0024
1.7	.415	.293	.122	.203	.0021
1.9	.332	.228	.104	.149	.0020
2.1	.265	.177	.088	.109	.0019

$$\Delta C_{D_N} = \frac{\Delta(p_s - p_\infty)}{q} \quad \frac{A_N}{S_w}$$

$$= \left(\frac{\Delta p_s}{H_\infty}\right) \frac{2}{\partial M_\infty^{\,2}} \quad \frac{A_N}{S_w} \quad \left(\frac{H_\infty}{p_\infty}\right)$$

$$A_N/S_w = 8.73/1225$$

Similarly, as a check, use:

$$\Delta C_{D_N} = \Delta C_{p_N} \cdot A_N/S_w$$

M	C_{p_N} $\theta = 25°$	$\theta = 15°$	ΔC_{p_N}	ΔC_{D_N}
1.6	.537	.228	.309	0022
1.8	.499	.210	.289	0021
2.0	.474	.200	.274	0020

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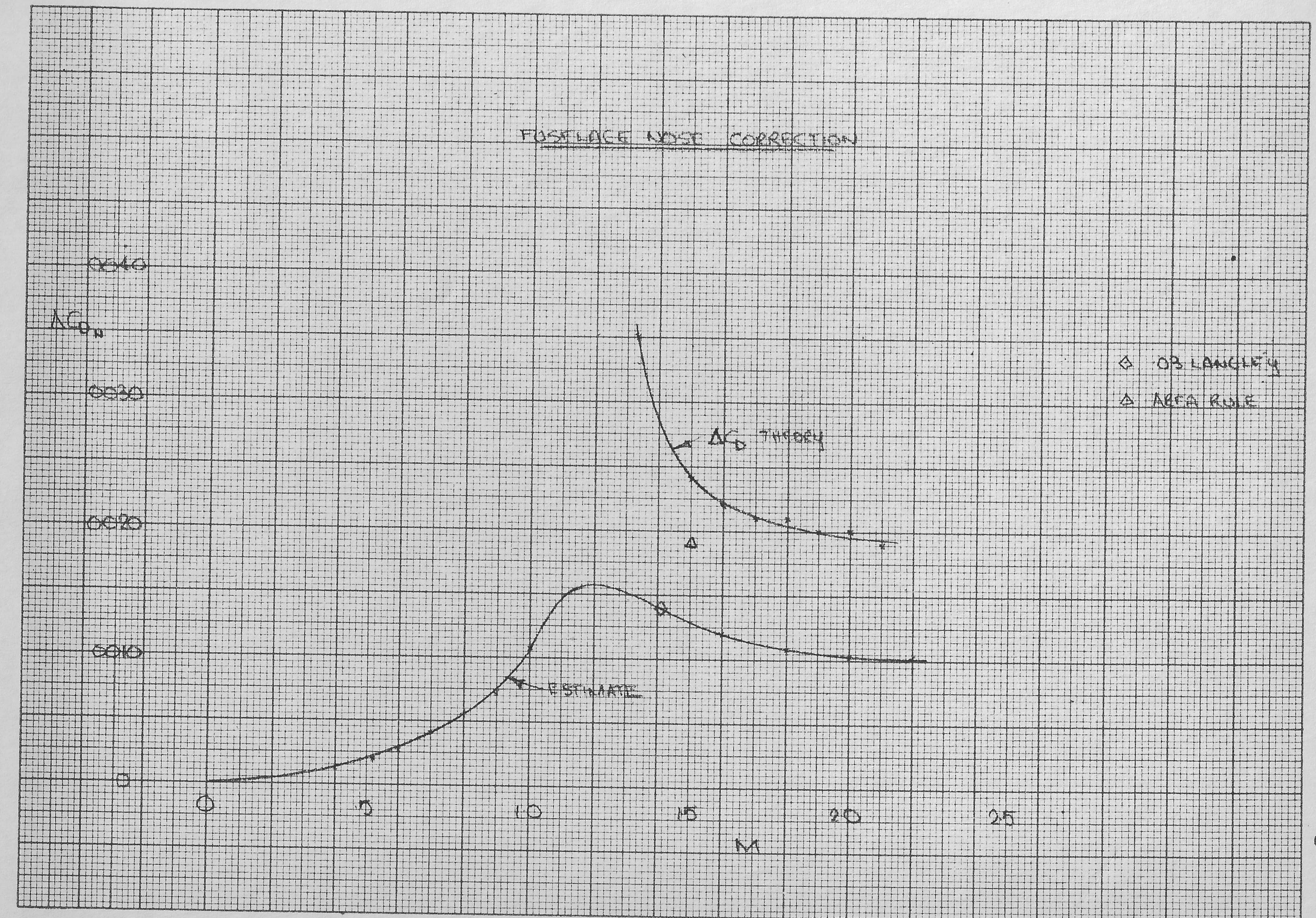
Summary for .04 CAL Model:

M_∞	$C_{D_{X_{sting}}}$	C_{D_c}	C_{D_M}	C_{D_B}	C_{D_s}	$\Delta C_{D_{Nose}}$	$C_{D_{ext}}$
.5	0167	.0015	-0049	0	0	-0002	0131
.8	0176	.0017	-0050	0	-0001	-0006	0136
.9	0174	.0018	-0045	0	-0002	-0008	0137
.95	0183	.0019	-0042	0	-0004	-0009	0147
.98	0198	.0019	-0041	0	-0005	-0010	0161
1	0218	.0019	-0040	0	-0006	-0011	0180
1.05	0258	.0018	-0037	-0001	-0007	-0013	0218
1.1	0249	.0020	-0035	-0001	-0009	-0014	0210
1.15	0250	.0020	-0033	-0002	-0011	-0015	0209
1.23	0267	.0021	-0030	-0002	-0013	-0015	0228


FUSELAGE NOSE CORRECTION
ΔC_{D_N}
.0040
.0030
.0020
.0010
0
0
.5
1.0
1.5
2.0
2.5
M
ΔC_D THEORY
ESTIMATE
◇ OB LANGLEY
△ AREA RULE

(b) .03 Scale, Langley Model.

$$C_{D_{ext}} = C_{D_{X_{Sting}}} + C_{D_c} + C_{D_M} - C_{D_B} - C_{D_s}$$

$$C_{D_{X_{Sting}}} = +.0215$$

$$C_{D_c} = \left(\frac{p_c - p_\infty}{qS}\right) A_c = \left(\frac{868.5 - 447.5}{684}\right) .0069 = .0043$$

$C_{D_B} = +.0004$, assumed same as for FFRM No. 6.

$$C_{D_M} = \frac{m\,(V_4 - V_\infty)}{qS} + \frac{(p_4 - p_\infty)}{qS} A_4\,\sigma$$

Assume, as for .04 Cornell Model, that the exit is choked.

$$\therefore\ m = \frac{.81\ H_4\ A_4}{a_o}\sigma$$

$$H_4 = H_i - f\,(L/D)q - .42q = \left(\frac{H_i}{H_\infty}\right)H_\infty - q(f\ L/D + .42)$$

$f = .018$ $\quad$ (R.N.)D = 229000

$L/D \doteqdot 14.2$

$$\therefore\ H_4 = .988 \times 1436 - 620\ (.018 \times 14.2 + .42)$$

$$= 1415 - 419$$

$$= 996 \text{ psf} \qquad \therefore\ H_4/H_i = .704$$

$A_4 = .85$ "/side
$a_o = 1162$ fps

$$\therefore\ m \doteqdot \frac{.81 \times 996 \times .85 \times 2}{144 \times 1162} = .00819 \text{ slugs/sec.}$$

$$V_4 = M_4 \left(\frac{a_4}{a_0}\right) a_0$$

$$= 1.0 \times .9129 \times 1162$$

$$= 1052 \text{ fps.}$$

$$V_\infty = M_\infty \left(\frac{a_\infty}{a_0}\right) a_0$$

$$= 1.41 \times .845 \times 1162$$

$$= 1388 \text{ fps.}$$

$$p_4 = .528 \times 996 = 526 \text{ psf.}$$

$$\therefore C_{D_M} = \frac{00819\ (1052 - 1388)}{684} + \frac{(526 - 447.5)}{684} \times \frac{.85 \times 2}{144}$$

$$= -0040 + .0014$$

$$= -0026$$

$$C_{D_S} = \frac{\partial C_{D_S}}{\partial m/mo} \left(m_i^*/mo - m/mo\right) \frac{A_f}{S_W} \times 2$$

$$m_i^* = .81 \frac{H_i\ A_i\ \sigma}{A_s} = \frac{.81 \times .986 \times 1436 \times .73 \times .97 \times 2}{1162 \times 144}$$

$$= .0097$$

$$m_o = \frac{\partial H_\infty\ A_c\ \varphi(M_\infty)}{A_s} = \frac{1.4 \times 1436 \times 1.3 \times 2 \times .516}{1162 \times 144}$$

$$= .0161$$

$$\therefore C_{D_S} = .306 \left(\frac{.0097}{.0161} - \frac{00819}{0161}\right) \frac{4098}{1225} \times 2$$

$$= .0019$$

$$\therefore C_{D_{ext}} = .0215 + .0043 - .0025 - .0004 - .0018$$

$$= .0211$$

(c) .03 Langley (Unitary)

$$C_{D_{ext}} = C_{D_{X_{Sting}}} + C_{D_c} + C_{D_M} - C_{D_B} - C_{D_s}$$

$C_{D_{X_{Sting}}}$ is found from plot of $C_{D_{X_{Sting}}}$ versus $\left[C_L - C_{L_{C_{D_{Min}}}}\right]^2$, sheets 60, 61

M = 1.6 Run 13

α	$C_{D_{X_{Sting}}}$	$C_L - C_{L_{C_{D_{Min}}}}$	$(C_L - C_{L_{C_{D_{Min}}}})^2$
-2.06	0231	-.1086	0118
-.99	0199	-.0583	0034
.06	0191	-.0092	0
2.19	0226	.0897	0080
4.33	0321	.1874	0351
-2.32	0247	-.1176	0138
-1.29	0215	-.0721	0052
-.27	0200	-.0251	0006
.90	0200	.0283	0008
1.87	0217	.0725	0052
3.02	0255	.1234	0152
3.88	0300	.1658	0275

M = 1.8

α	$C_{D_{X_{Sting}}}$	$C_L - C_{L_{C_{D_{Min}}}}$	$(C_L - C_{L_{C_{D_{Min}}}})^2$
-2.06	0226	-0865	0075
.09	0188	-0015	0
2.13	0223	0862	0074
4.31	0306	1724	0297
-3.35	0275	-1372	0188
-2.35	0238	-0956	0091
-1.26	0210	-0517	0027
-.29	0200	-0127	0002
.88	0202	0365	0013
1.95	0226	0806	0069
2.99	0259	1195	0143
3.87	0296	1560	0243

M = 2.0 Run 14

α	$C_{D_{X_{Sting}}}$	$C_L - C_{L_{C_{D_{Min}}}}$	$(C_L - C_{L_{C_{D_{Min}}}})^2$
-1.99	0208	-0666	0044
.06	0188	0096	0001
1.05	0200	0464	0021
2.15	0223	0871	0076
4.28	0313	1635	0267

M = 2.0 Run 14 (Cont'd)

α	$C_{D_{X_{Sting}}}$	$C_L - C_{L_{C_{D_{Min}}}}$	$(C_L - C_{D_{Min}})^2$
-2.24	0220	-0682	0046
-1.25	0201	-0302	0009
-.19	0194	0087	0001
.83	0202	0452	0020
1.89	0226	0832	0092
2.95	0261	1210	0146
3.79	0298	1490	0222

Giving:

M	$C_{D_{X_{Sting_{Min}}}}$
1.6	0193
1.8	0199
2.0	0192

$C_{D_c} = \dfrac{X_c}{qS_w}$ $\qquad q = \dfrac{\partial p_\infty M_\infty^2}{2}$ $\qquad S_w = 1.103$

where $X_c = \left[(X_1 - 2500)\ 1.081967 + X_2 - p_\infty\right] A_c$

where X_1 and X_2 are W/T readings

M	X_1	X_2	p_∞	A_c	X_c	q	C_{D_c}
1.6	1495	2134.5	395	.0069	4.49	708	0058
1.8	1421	2134.5	293.4		4.65	665	0063
2.0	1287	2132.4	214.5		4.18	601	0063

$$C_{D_M} = \frac{m}{m_o} \left(\frac{9129}{M_\infty} \frac{a_o}{a_\infty} - 1\right)^2 \frac{A_c}{S_w} + \left(\frac{p_4}{p_\infty} - 1\right) \frac{A_4}{S_w} \frac{2}{\gamma M_\infty^2}$$

I II

i.e. exit is assumed choked as was other W/T models.

$$\text{where } \frac{m}{m_o} = .5787 \frac{A_4}{A_c} \left(\frac{H_4}{H_\infty}\right) \varphi(M_\infty)$$

$$= .5787 \times .655 \times \left(\frac{H_4}{H_\infty}\right) \varphi(M_\infty)$$

$$H_4 = (X_3 - 2500)\, 1.069219 + X_2$$

where X_2, X_3 are W/T readings

M_∞	X_2	X_3	H_4	H_∞	$\frac{H_4}{H_i}$	$\varphi(M_\infty)$	m/m_o	a_∞/a_o	H_4/H_∞
1.6	2134.5	1600	1172	1680	.717	.4629	.572	.8133	.698
1.8	2134.5	1575	1145	1680	.722	.4022	.644	.7790	.682
2.0	2132.4	1421	980	1680	.658	.3429	.645	.7454	.583

M_∞	$\frac{9129}{M_\infty}\left(\frac{a_o}{a_\infty}\right)$	I	p_4/p_∞	II	C_{D_M}
1.6	.702	-.0056	1.57	+.0034	-.0022
1.8	.651	-.0074	2.06	+.0050	-.0024
2.0	.613	-.0082	2.41	+.0054	-.0028

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$$C_{D_B} = \left[(X_4 - 2500)\ 1.09173 + X_2 - p_\infty\right] \frac{.004722}{qS_W}$$

M_∞	X_4	X_2	p	q	C_{D_B}
1.6	820	2134.5	395	708	-0006
1.8	693	2134.5	293.4	665	-0008
2.0	638	2132.4	214.5	601	-0008

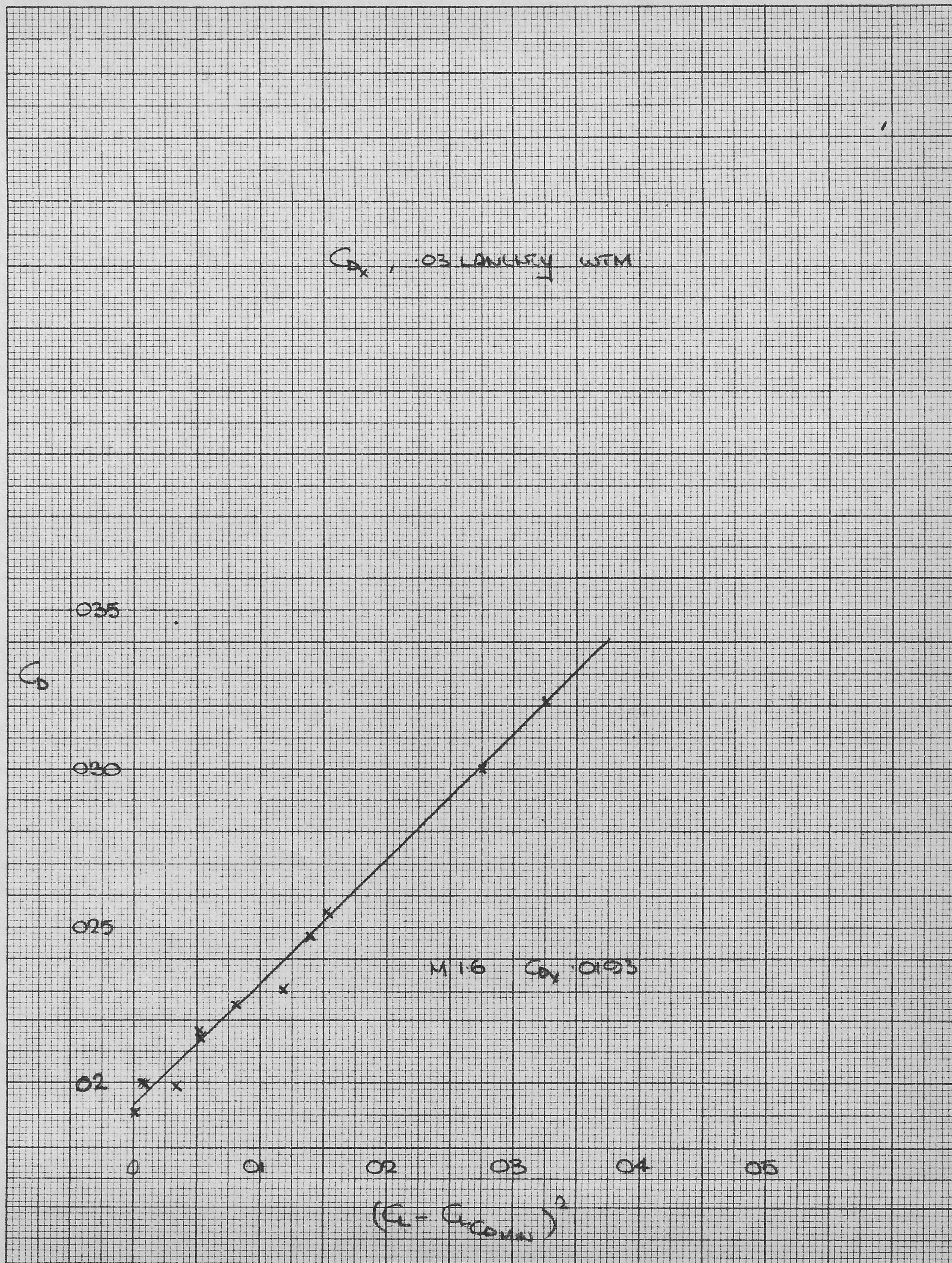
$$C_{D_s} = \frac{\partial C_{D_s}}{\partial m/mo} \left(m^*/mo - m/mo\right) \frac{A_f}{S_W} \times 2$$

$$m^*/mo = .5787 \frac{A_i}{A_c}\ \sigma \left(\frac{H_i}{H_\infty}\right) \varphi(M_\infty) \qquad \frac{A_i}{A_c} = \frac{5.6}{10.18} = \frac{.73}{1.3}$$

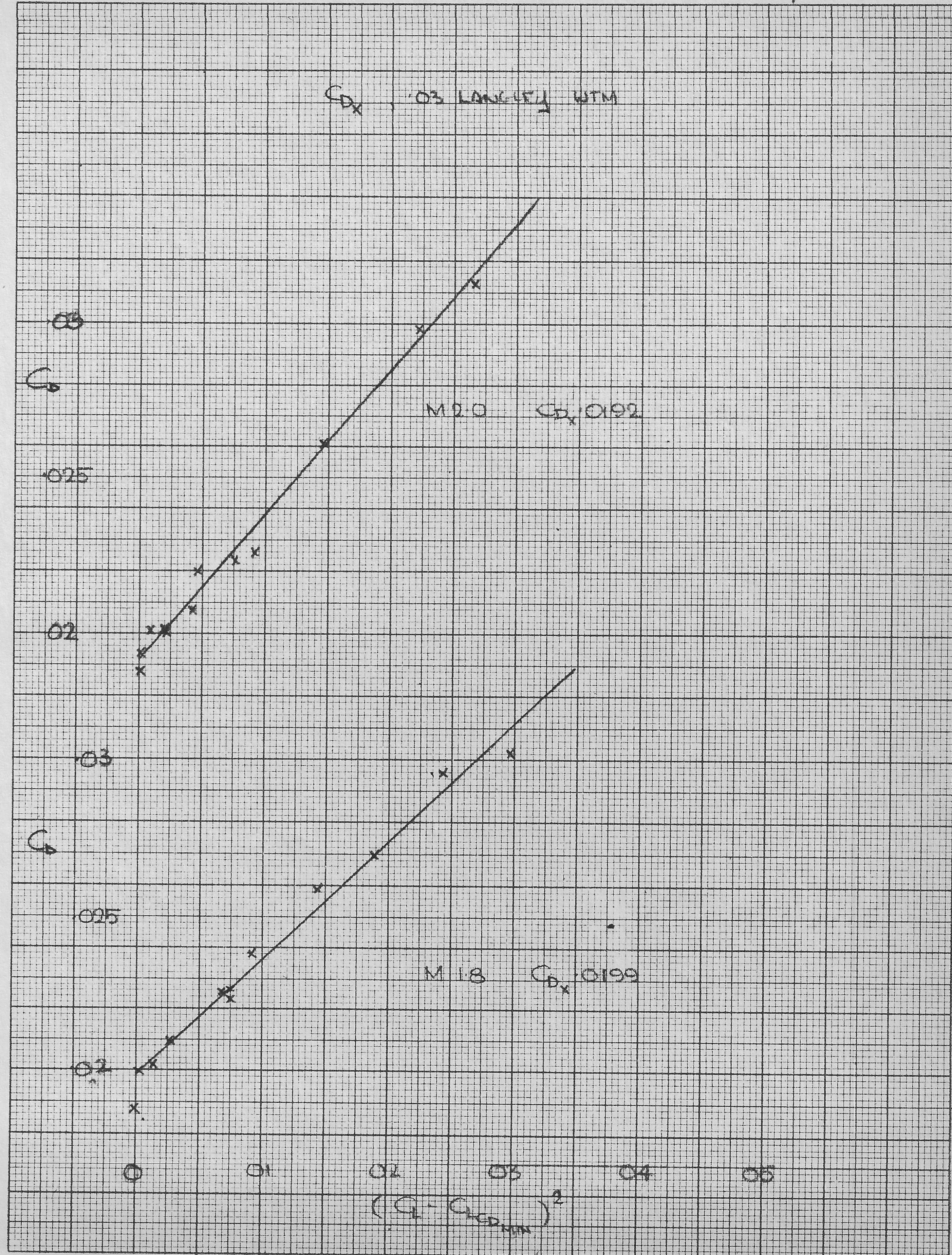
M_∞	σ	H_i/H_∞	$\varphi(M_\infty)$	m^*/mo	m/mo	$\partial C_{D_s}/\partial m/mo$	C_{D_s}
1.6	.97	.975	.4629	.661	.572	.335	.0020
1.8	.97	.945	.4022	.725	.644	.355	.0019
2.0	.97	.885	.3429	.801	.645	.372	.0038

Summary, .03 Langley

M	$C_{D_{X Sting}}$	C_{D_c}	C_{D_M}	C_{D_B}	C_{D_s}	$C_{D_{ext}}$
1.6	0193	0058	-0022	-0006	-0020	0203
1.8	0199	0063	-0024	-0008	-0019	0211
2.0	0192	0063	-0028	-0008	-0038	0203



CDx , ·03 LANGLEY WTM
CD
·035
·030
·025
·02
M 1·6 CDx ·0193
0
·1
·2
·3
·4
·5
(CL - CLCDmin)²



CDx , '03 LANGLEY WTM
·03
CD
·025
M 2·0 CDx ·0192
·02
·03
CD
·025
M 1·8 CDx ·0199
·02
0
·01
·02
·03
·04
·05
(CL - CLCDMIN)²

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(d) Area Rule, Summary of data available:

Ref:	Date	Config.	$C_{D_W} + C_{D_{INT}}$
5920/20/3	March 4/57	50° nose 45.1" ejector	0164
"	"	30° nose 45.7" ejector	0144
6244/20/J	March 18/57	30° nose 39" ejector	0163
	Oct. 23/57	30° nose 39" ejector	0161
5920/20/J	March 4/57	30° nose 39" ejector	0114
		" 43" "	0089
		" 45.7" "	0080
		" 57.3" "	0064

AIRCRAFT:		PREPARED BY	DATE
		CHECKED BY	DATE

Area Rule Estimate - Ref. IDM 6244/20/J
F.A. Woodward
March 18/57

$$C_{D_{ext}} = C_{D_{AR}} + C_{D_{SF}}$$

where:

$C_{D_{AR}}$ is the Area-Rule estimate for the Arrow, with J75 intake and exit, of wave and interference drag.

$C_{D_{SF}}$ is the additional drag contribution due to skin friction.

$$\therefore \quad C_{D_{ext}} = .0161 + .0065 = .0226$$

The drag decrement due to enlarged ejector, Arrow 1A configuration, is: 0161-0145 = 0016

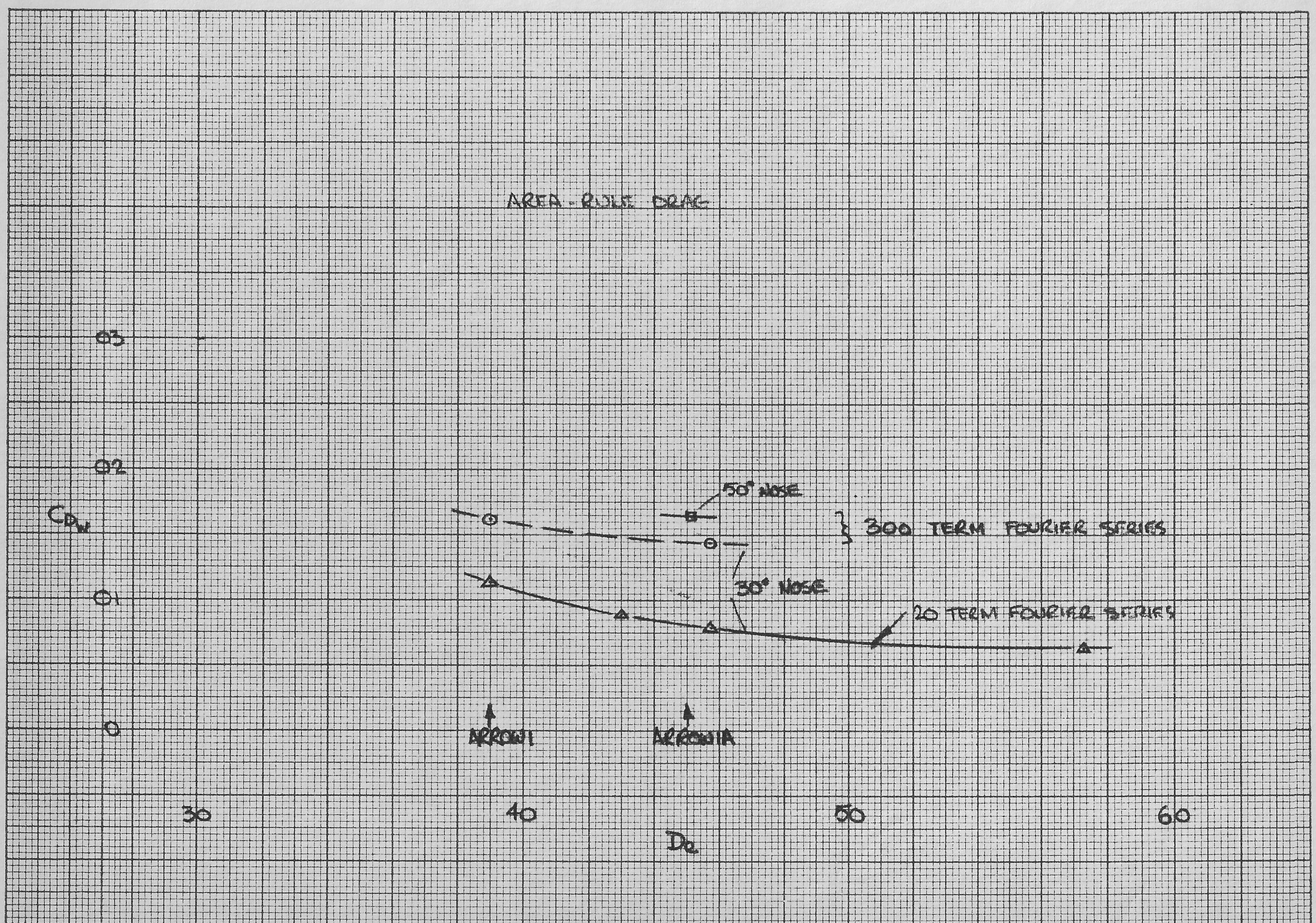
Ref. IDM Oct. 23/57

Effect of nose; reduction in conical angle of 50 to 30°.

$$\Delta C_{D_N} = .0164 - 0145 = .0019$$

Ref. IDM 5920/20/J
F.A. Woodward
March 4/57

From sheet 59 it is seen that approximately 70% of the Area-Rule estimate was realised in W.T.T. It is suggested, therefore, that the improvement in external drag due to enlarged ejector (Arrow 1A) is 70% of 0016, i.e. .0011.


AREA - RULE DRAG
50° NOSE
300 TERM FOURIER SERIES
30° NOSE
20 TERM FOURIER SERIES
ARROW I
ARROW IA
C_{D_W}
.03
.02
.01
0
30
40
50
60
Da

(e) Free Flight Rocket Model No. 6 1/8 scale.

$$C_{D_{ext}} = C_{D_X} - \Delta C_{D_{Pitot}} - \Delta C_{D_{Boom}} - \Delta C_{D_{Fin}} - \Delta C_{D_{Rakes}} - C_{D_B} - C_{D_M} - \Delta C_{D_{Trim}} - \Delta C_{D_i} - C_{D_s}$$

where:

C_{D_X}: total drag measured during deceleration of model.

ΔC_D pitot: Pitot correction.

$\Delta C_{D_{Boom}}$: Boom correction.

$\Delta C_{D_{Fin}}$: Oversize fin correction.

$\Delta C_{D_{Rakes}}$: Rake correction.

C_{D_B}: Base Area correction, $C_{D_B} = \left(\frac{p_B - p_\infty}{qS}\right) A_B$

C_{D_M}: Momentum correction, $C_{D_M} = \frac{m(V_4 - V_\infty) + (p_4 - p_\infty) A_4}{qS}$

$$m = \frac{\partial H_4 \, A_4 \sigma \, \varphi(M_4)}{a_o}$$

$$m_i^* = \frac{81 \, H_i \, A_i \sigma}{a_o}$$

$$m/m_i^* = \left(\frac{H_4}{H_i}\right)\left(\frac{A_4 \sigma}{A_i \sigma}\right) \frac{\varphi(M_4)}{.5787}$$

$A_4 \sigma \doteq 29.6$ $\quad A_4 = 31.8 \square''$ $\quad \sigma \doteq .93$
$A_i \sigma \doteq 24.4$ $\quad A_i = 25.2 \square''$ $\quad \sigma \doteq .97$

M_∞	p_4	H_4	p_∞	p_B	q_S	C_{D_B}	p_4/H_4	C_L
1.6	37.05	51.38	13.75	8.20	63138	-0005	.721	024
1.5	28.09	41.67	13.35	8.30	57554	-0005	.674	028
1.4	23.36	34.66	12.77	9.30	48080	-0004	.673	035
1.3	19.44	28.92	12.26	10.15	40003	-0003	.672	033
1.2	16.34	24.35	11.83	10.50	32921	-0002	"	019
1.15	15.20	22.50	11.67	10.75	29782	-0002	"	010
1.1	14.16	21.04	11.50	10.85	26796	-0001	"	002
1.05	13.30	19.40	11.30	11.08	24040	-0001	.685	-011
1.0	12.61	18.28	11.10	11.85	21475	0	.691	-042

M_∞	M_4	$\Delta C_{D_{Rake}}$	H_∞	a_o	$\varphi(M_4)$	H_i/H_∞	H_4/H_i	m/m_i *
1.6	.699	-0001	58.44	1368	.529	.975	.903	1.000
1.5	.773	"	48.60	1333	.552	.985	.871	1.006
1.4	.774	"	40.64	1303	.551	.986	.866	1.000
1.3	.775	"	33.97	1271	.551	.988	.861	.991
1.2	"	"	28.69	1254	"	.989	.857	.987
1.15	"	"	26.54	1240	"	.990	.857	.991
1.1	"	"	24.55	1230	"	"	.866	1.000
1.05	.758	"	22.69	1221	.548	"	.863	.992
1.0	.748	"	21.01	1210	.544	"	.879	1.001

$$\frac{H_4}{H_i} = \frac{H_4}{H_\infty}\left(\frac{H_\infty}{H_i}\right)$$

It is seen that the intake is choked, and thus no spillage drag correction need be applied, i.e. $C_{D_S} = 0$.

$V_4 = M_4 \; a_4 = M_4 \left(\frac{a_4}{a_o}\right) a_{\ominus} \qquad V_\infty = M_\infty \left(\frac{a_\infty}{a_o}\right) a_{\ominus}$

M	a_4/a_o	V_4	V_∞	m^*	$m^*(V_4 - V_\infty)$	$A_4 \; (p_4 - p_\infty)$	D_M	C_{D_M}
1.6	.954	912	1779	.825	715	690	25	0004
1.5	.945	975	1662	.701	481	439	42	0007
1.4	"	952	1546	.605	359	314	45	0009
1.3	.945	931	1429	.521	260	212	48	0012
1.2	"	919	1325	.448	182	133	49	0015
1.15	"	902	1269	.419	154	106	48	0016
1.1	"	898	1214	.391	124	80	44	0016
1.05	.947	877	1160	.364	103	59	44	0018
1.0	.948	861	1105	.341	84	45	39	0018

$\Delta C_{D_{Trim}}$ – Trim drag correction, $\delta_e = 5°41'$ Ref. 27

A_B 5.5 sq./in.

ΔC_{D_i}: Induced drag correction, i.e. drag due to lift on model.
$\Delta C_{Di} = C_{DC_L} - C_{DC_{LC_{DMin}}}$ Ref. 27

M	$\Delta C_{D_{Trim}}$	C_{DC_L}	$C_{DC_{LC_{DMin}}}$	ΔC_{D_i}	C_{D_X}	H_4/H_∞
1.6	.0033	0230	0228	0002	.0270	.880
1.5	.0029	0230	0228	0002	.0273	.857
1.4	.0028	0233	0231	0002	.0278	.853
1.3	.0037	0232	0231	0001	.0282	.851
1.2	.0048	0231	0231	0	.0284	.848
1.15	.0048	0230	0229	0001	.0284	.848
1.1	.0049	0223	0221	0002	.0282	.858
1.05	.0058	0207	0206	0001	.0273	.854
1.0	.0100	0165	0160	0005	.0226	.870

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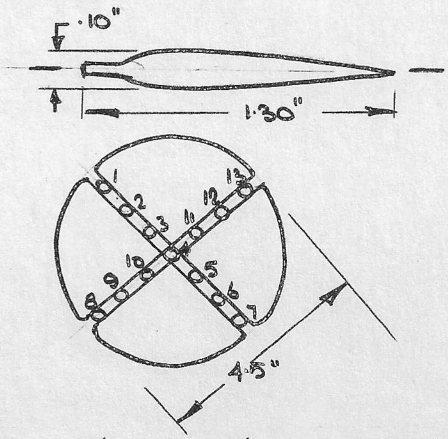
AIRCRAFT:

Pitot Pressure Rake



13 pitots imbedded in an aerofoil shape.

$$t/_c = .1/_{1.3} = 7.7\%$$

$$C_{D_{Rake}} = \frac{C_{D_o} \; q4 \; A4}{q_\infty \; S_W} \qquad .70 \leq M_4 \leq .78$$

$$R.N. = \frac{V \; x}{\nu} \doteqdot \frac{.75 \times 1117 \times 1.3}{1.564 \quad 12} \times 10^4 = 6 \times 10^5$$

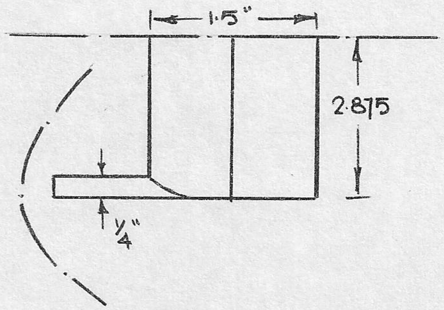
$$C_{D_{Rake/A/C}} = .0047 \times \frac{\partial}{2} \; p_4 \; \frac{M4^2}{qs} \; (4.5 \times 1.30 \times 8)$$

$$\doteqdot .0867 \; \frac{p4}{qs}$$

With interference effects use $C_{D_{Rake/\,A/C}} = .10 \; \frac{p4}{qs}$

M	p4	qs	$C_{D_{Rake/A/C}}$
1.6	37.05	63138	– .0001
1.5	28.09	57554	"
1.4	23.36	48080	"
1.3	19.44	40003	"
1.2	16.34	32921	"
1.15	15.20	29782	"
1.1	14.16	26796	"
1.05	13.30	24040	"
1.0	12.61	21474	"

Pitot Drag



Assume drag of pitot is equivalent to an isolated cylinder and double-wedge aerofoil.

a) Cylinder C_{D_o} ref. Hoerner Fig. XVI-14

$$C_D = \frac{C_{D_o}\, q_\infty\, A}{q_\infty\, S_w} = C_{D_o}\frac{A}{S_w} = C_{D_o}\frac{\pi}{4} \times \frac{.25^2}{1225} \times \frac{64}{144}$$

$$= 1.78\, C_{D_o} \times 10^{-5}$$

b) Double Wedge $t/c = 16.67\%$

C_{D_o} ref. Hoerner Fig. XVII-11

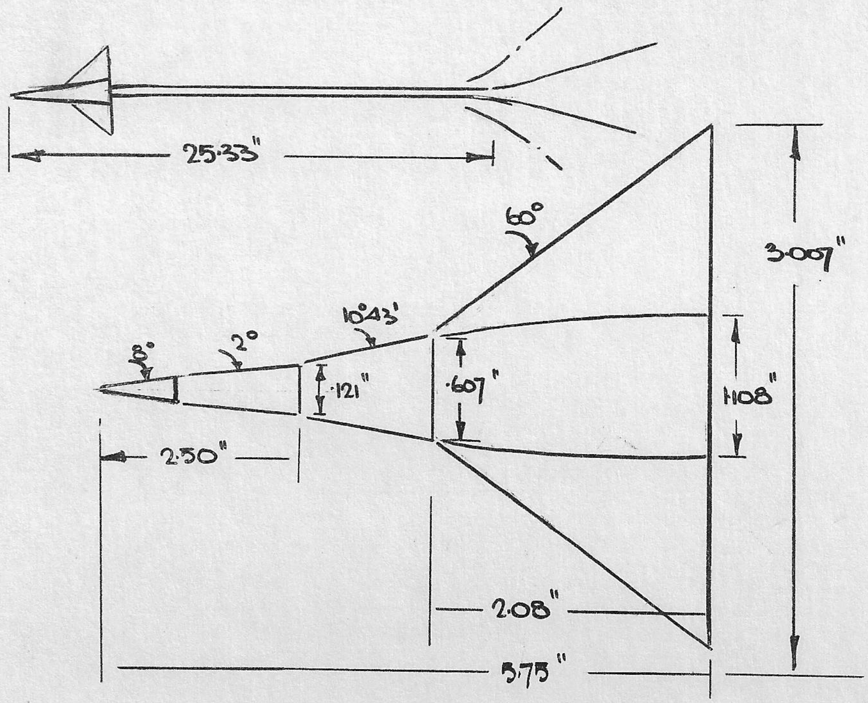
$$C_D = C_{D_o}\frac{q_\infty\, A}{q_\infty\, S_w} = C_{D_o}\frac{A}{S_w} = C_{D_o}\frac{1.5 \times 2.875 \times 64}{1225 \times 144}$$

$$= 1.56\, C_{D_o} \times 10^{-3}$$

M	C_{D_o}(cyl)	C_{D_o}(dbl.wedge) 10% t/c	C_D(cyl)	C_D(D.W.)	C_D pitot
1.6	1.6	03	00003	00013	0002
1.5	1.55	035	00003	00015	0002
1.4	1.5	04	00003	00017	0002
1.3	1.45	055	00003	00024	0003
1.2	1.4	065	00003	00028	0003
1.15	1.35	075	00002	00032	0003
1.1	1.3	08	00002	00035	0004
1.05	1.25	0825	00002	00036	0004
1.0	1.2	085	00002	00037	0004

Boom Correction

Enlarged Tip Cone:



Assume the drag due to build-up of boundary layer on the splike is balanced by the reduction in cone angle. It is assumed, therefore, that the drag of the boom is equivalent to an isolated tip cone.

$$C_D = C_{D_o} \frac{A}{S} \doteq C_{D_o} \frac{\pi}{4} \times \frac{1.108^2}{2760} = 3.48 \times 10^{-4} C_{D_o}$$

M	C_{D_o}	$C_{D_{Boom}}$
1.6	.38	0001
1.5	.40	0001
1.4	.42	0001
1.3	.45	0002
1.2	.47	0002
1.15	.48	0002
1.1	.50	0002
1.05	.47	0002
1.0	.42	0001

C_{D_o} ref. Hoerner Fig. XVI-39

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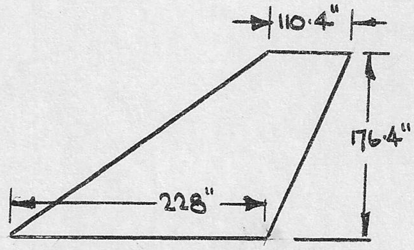
AIRCRAFT:

Oversize Fin

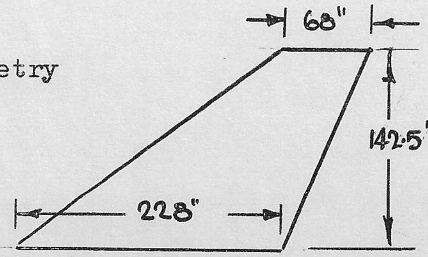
Ref. P. Geom. 32

Model Geometry



$$A_f = \left(\frac{228 + 110.4}{2}\right)\frac{176.4}{144} = 208^{\square'}$$

Aircraft (Arrow 1) Geometry



$$A_f = \left(\frac{228 + 68}{1225}\right)\frac{142.5}{144} = 147^{\square'}$$

$$\Delta C_{D_w} = C_{D_w}\left(\frac{208-147}{1225}\right) = 4.98 \times 10^{-2}\, C_{D_w}$$

C_{D_w} Ref. TN 4201 Fig. 5

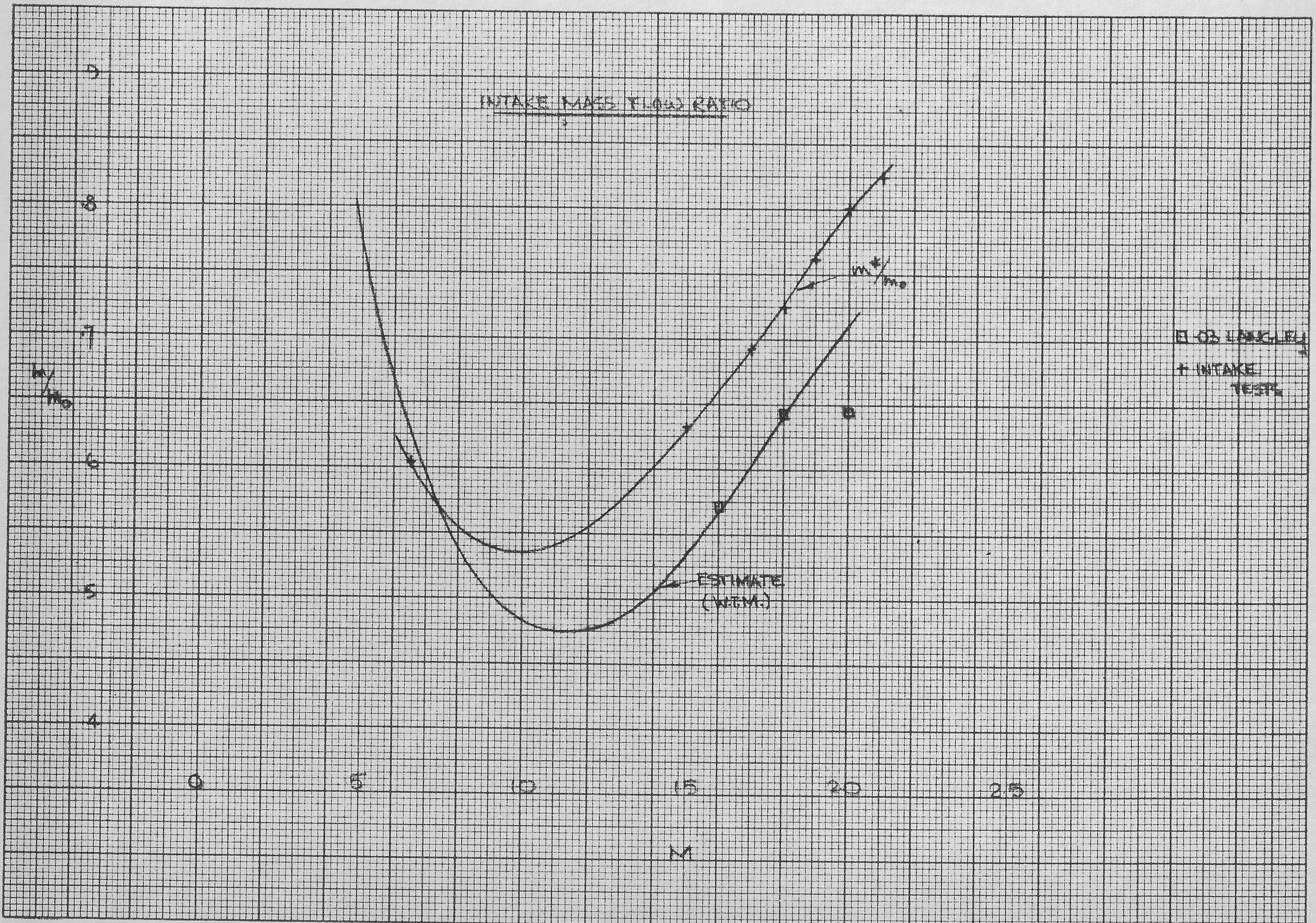
$$\Delta C_{D_f} = C_{D_f}\left(\frac{208-147}{1225}\right)^2 = 9.96 \times 10^{-2}\, C_{D_f}$$

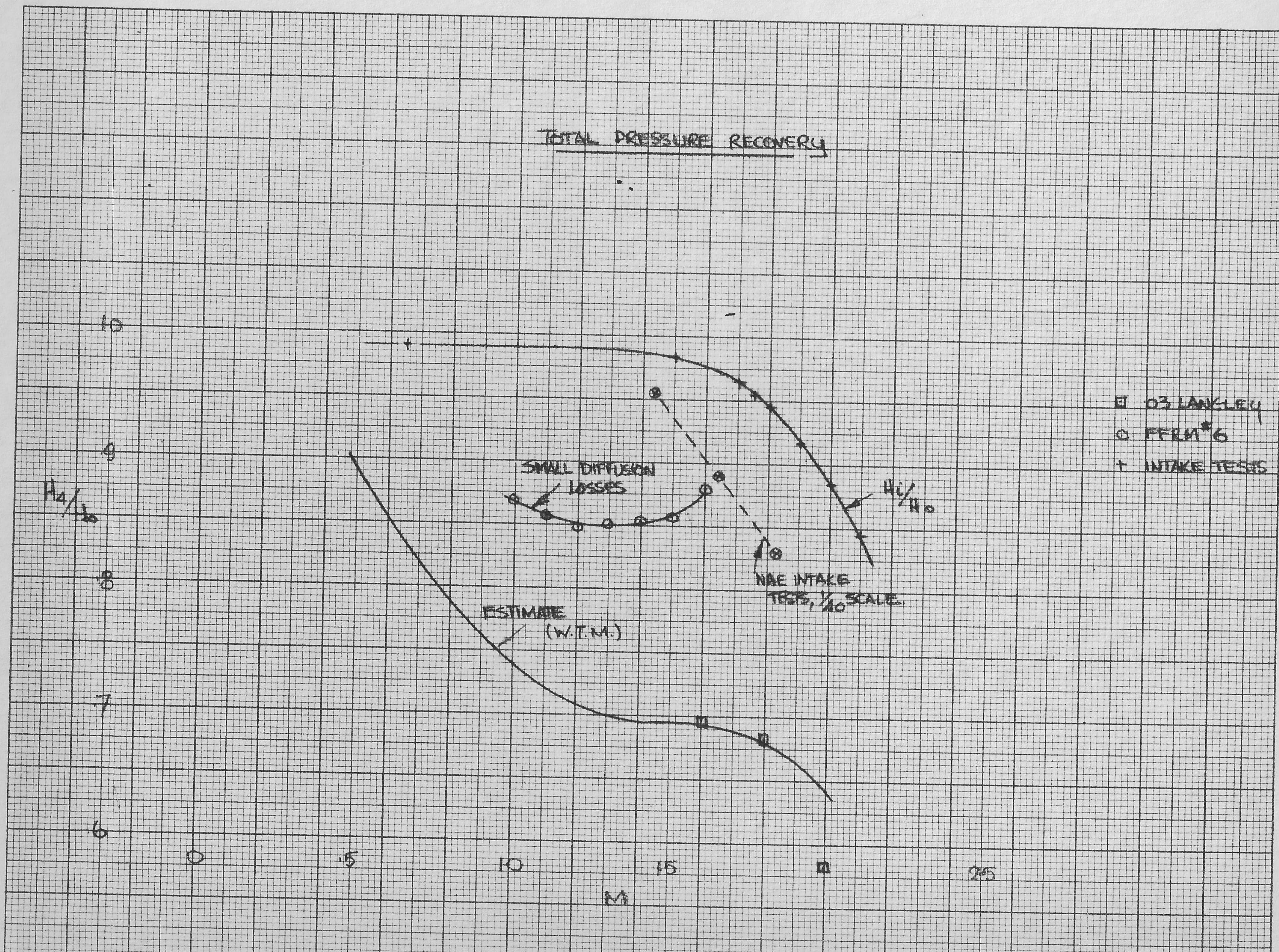
C_{D_f} Ref. TN 4201 Fig. 2

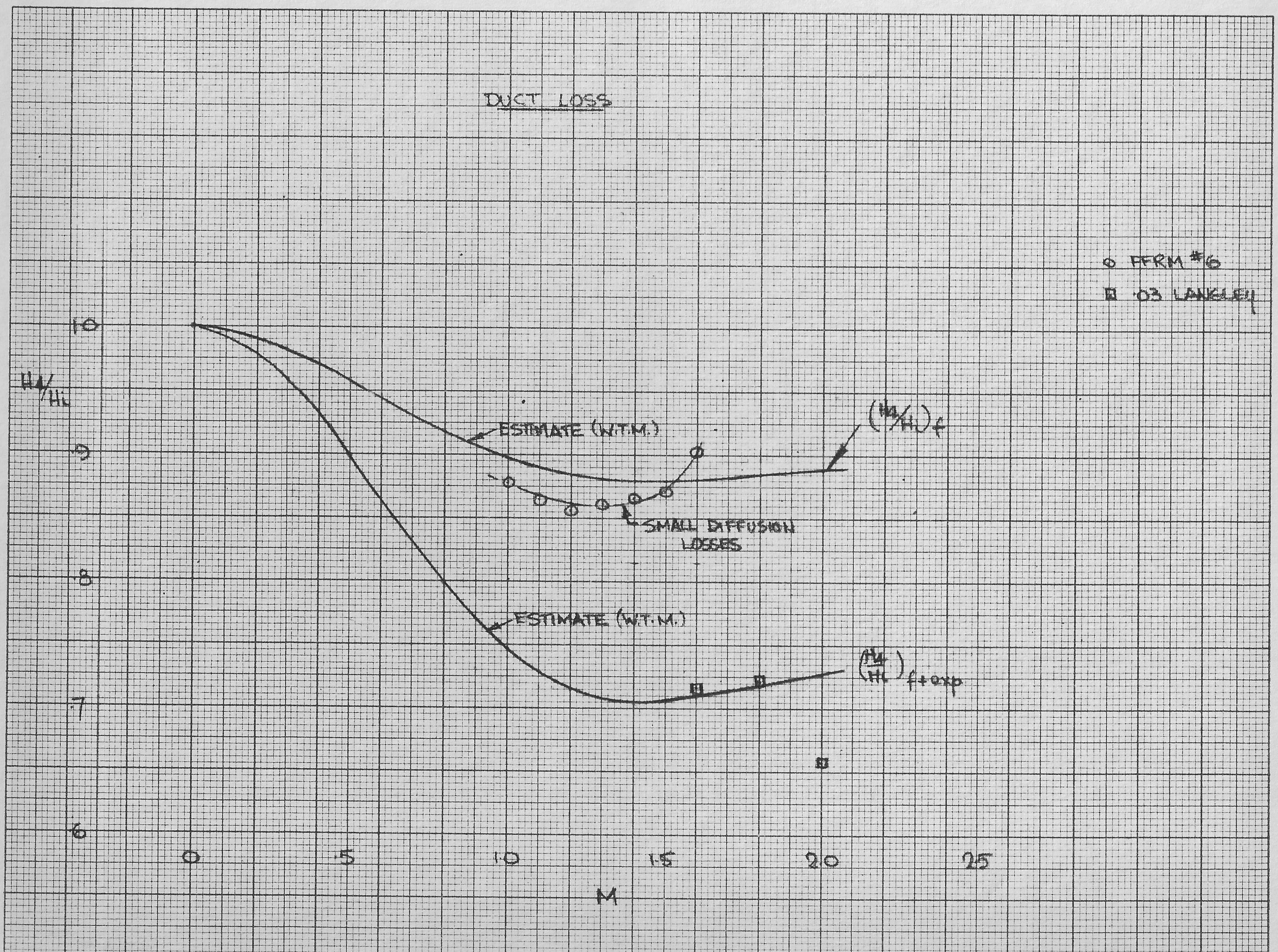
M	R.N.	C_{D_w}	ΔC_{D_w}	C_f	ΔC_{D_f}	$\Delta C_{D_{Fin}}$
1.6	44.2	0037	00018	00190	00019	00037
1.5	39.6	"	"	00197	00020	00038
1.4	35.6	"	"	00204	00020	00039
1.3	31.7	"	"	00212	00021	00040
1.2	28.0	"	"	00219	00022	00040
1.1	24.5	"	"	00228	00023	00041
1.0	21.5	0054	00027	00240	00024	00051

Summary, FFRM No. 6:

M_∞	C_{D_X}	$\Delta C_{D_{Pitot}}$	$\Delta C_{D_{Boom}}$	$\Delta C_{D_{Fin}}$	$\Delta C_{D_{Rakes}}$	C_{D_B}	C_{D_M}	$\Delta C_{D_{Trim}}$	ΔC_{D_i}	$C_{D_{Ext}}$
1.6	0270	-.0002	-.0001	-.0004	-.0001	-0005	-0004	-0033	-0002	0217
1.5	0273	"	"	"	"	-0005	-0007	-0029	-0002	0221
1.4	0271	"	"	"	"	-0004	-0009	-0028	-0002	0226
1.3	0282	-.0003	-.0002	"	"	-0003	-0012	-0037	-0001	0220
1.2	0284	"	"	"	"	-0002	-0015	-0048	0	0210
1.15	0284	"	"	"	"	-0002	-0016	-0048	-0001	0208
1.1	0282	-.0004	"	"	"	-0001	-0016	-0049	-0002	0205
1.05	0273	"	"	"	"	-0001	-0018	-0058	-0001	0186
1.0	0226	"	-.0001	-.0005	"	0	-0018	-0100	-0005	0094


INTAKE MASS FLOW RATIO
m/m₀
m*/m₀
ESTIMATE
(W.T.M.)
□ .03 LANGLEY
+ INTAKE
TESTS
M
0
5
10
15
20
25
4
5
6
7
8
9


TOTAL PRESSURE RECOVERY
10
9
8
7
6
H_4/H_0
0
.5
1.0
1.5
2.0
2.5
M
H_i/H_0
SMALL DIFFUSION LOSSES
NAE INTAKE TESTS, 1/40 SCALE.
ESTIMATE (W.T.M.)
⊡ 03 LANGLEY
○ FFRM #6
+ INTAKE TESTS


DUCT LOSS
o FFRM #6
▣ ·03 LANGLEY
H4/Hi
1·0
·9
·8
·7
·6
ESTIMATE (W.T.M.)
ESTIMATE (W.T.M.)
(H4/Hi)f
(H4/Hi)f+exp
SMALL DIFFUSION LOSSES
0
·5
1·0
1·5
2·0
2·5
M

51

Estimate of C_{D_M} and C_{D_s} for the W/T Models.

M	H_i/H_∞ Intake Tests	$\frac{2}{\partial M_\infty^2}$	p_∞/H_∞	a_∞/a_o	$\Delta H/H_\infty$	H_4/H_∞	$\varphi(M_\infty)$	m/mo	I
.5	.99	5.71	.843	.976	.0998	.890	.432	.781	0222
.8	.99	2.23	.656	.942	.199	.790	.557	.537	0037
1.0	.99	1.43	.528	.913	.250	.740	.579	.484	0
1.2	.989	.992	.412	.881	.281	.708	.562	.478	-0021
1.5	.985	.635	.272	.831	.290	.695	.492	.535	-0047
1.8	.945	.441	.174	.779	.267	.678	.402	.640	-0073
2.0	.885	.357	.128	.745	.242	.643	.343	.711	-0090

$$H_4 = H_i - q(f\,L/D + .42) = H_i - .676q$$

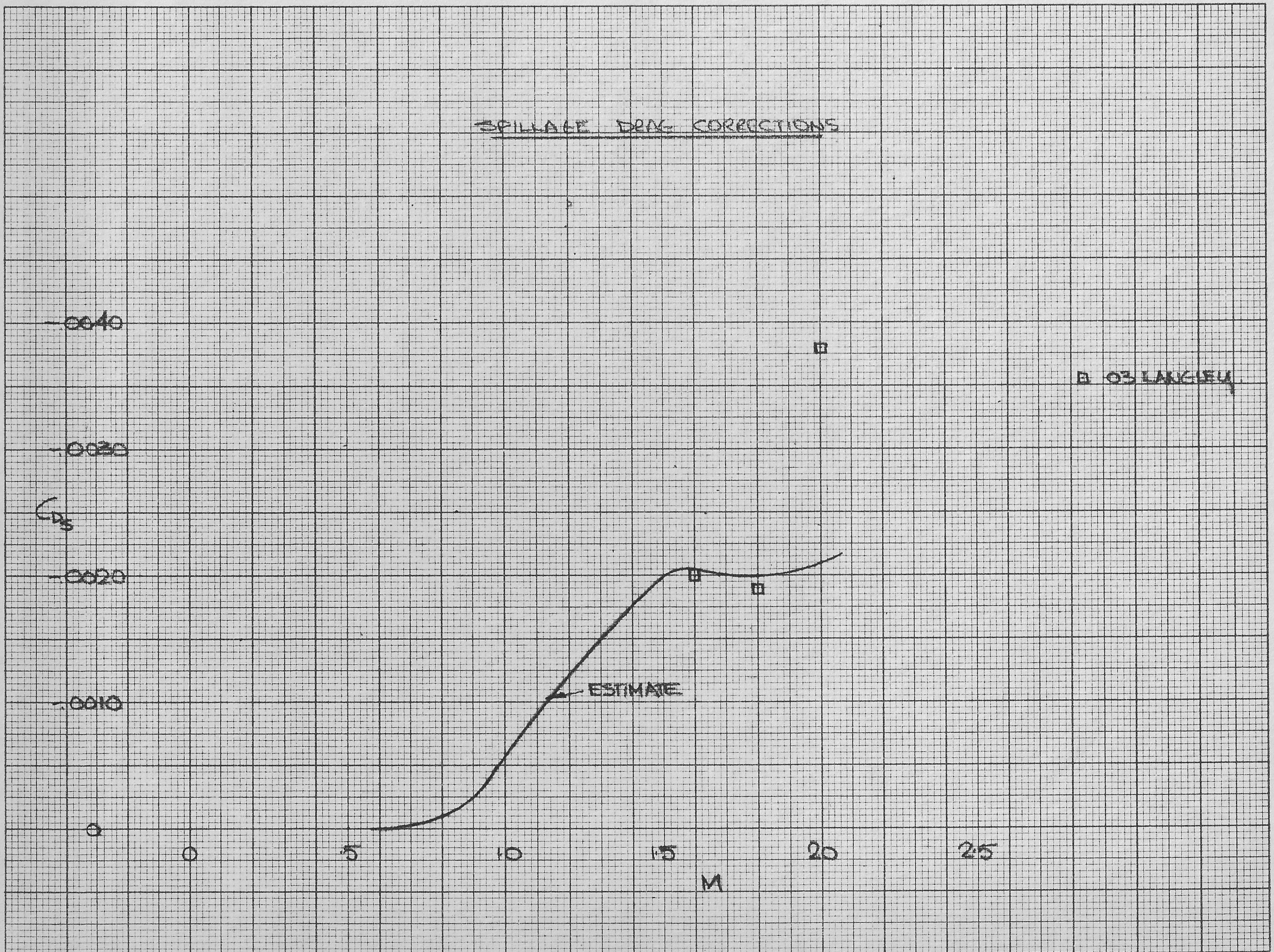
$$H_4/H_\infty = H_i/H_\infty - .676 \times \partial \left(\frac{p_\infty}{H_\infty}\right)\frac{M_\infty^2}{2}$$

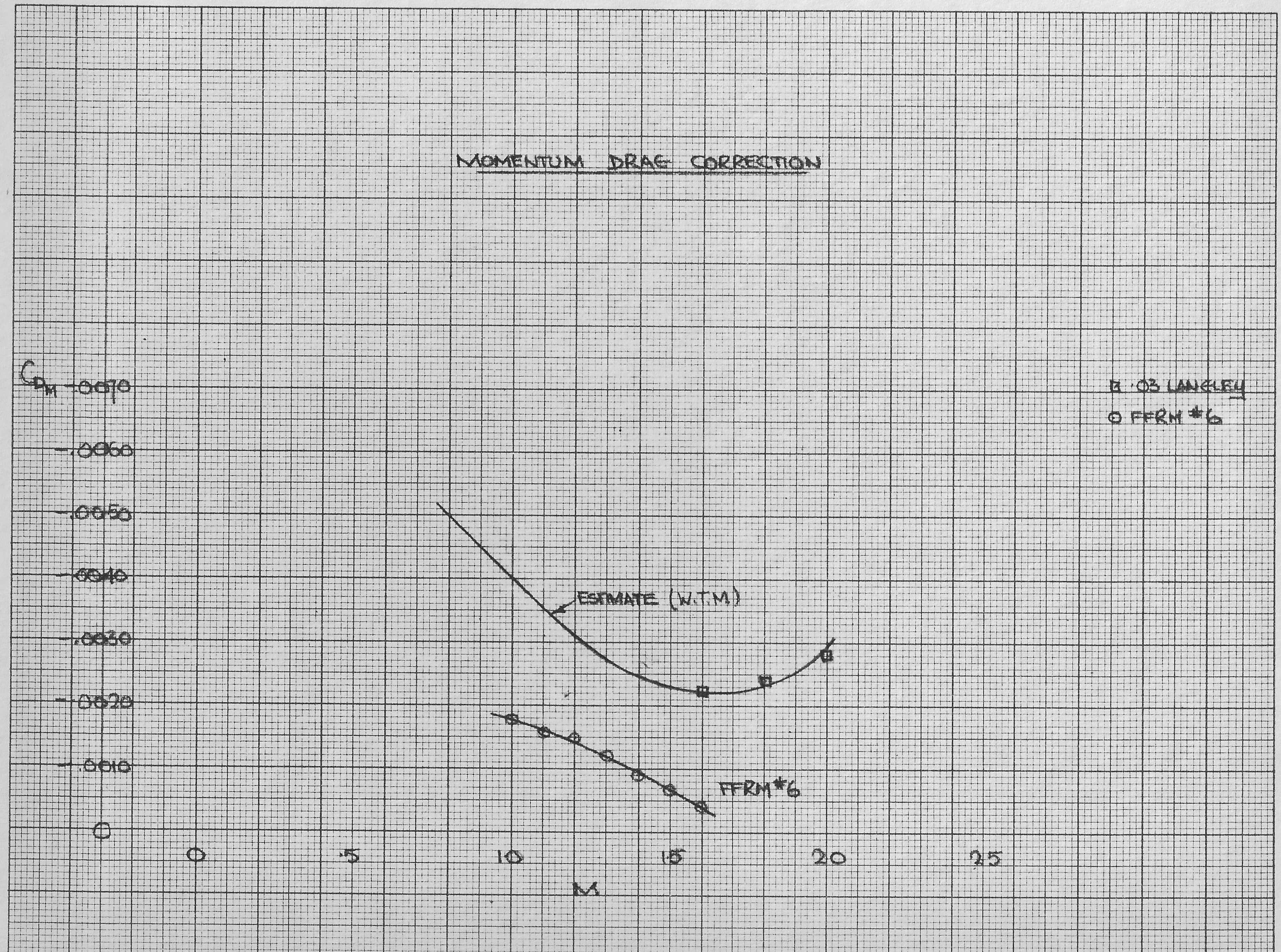
$$C_{D_M} = \frac{m}{mo}\underbrace{\left[\frac{.9129}{M_\infty}\left(\frac{a_o}{a_\infty}\right)^{-1}\right]}_{I}\frac{2A_c}{S_w} + \underbrace{\left\{.5283\left(\frac{H_\infty}{p_\infty}\right)\left(\frac{H_4}{H_\infty}\right)^{-1}\right\}}_{II}\frac{A_4}{S_w}\,\frac{2}{\partial M_\infty^2}$$

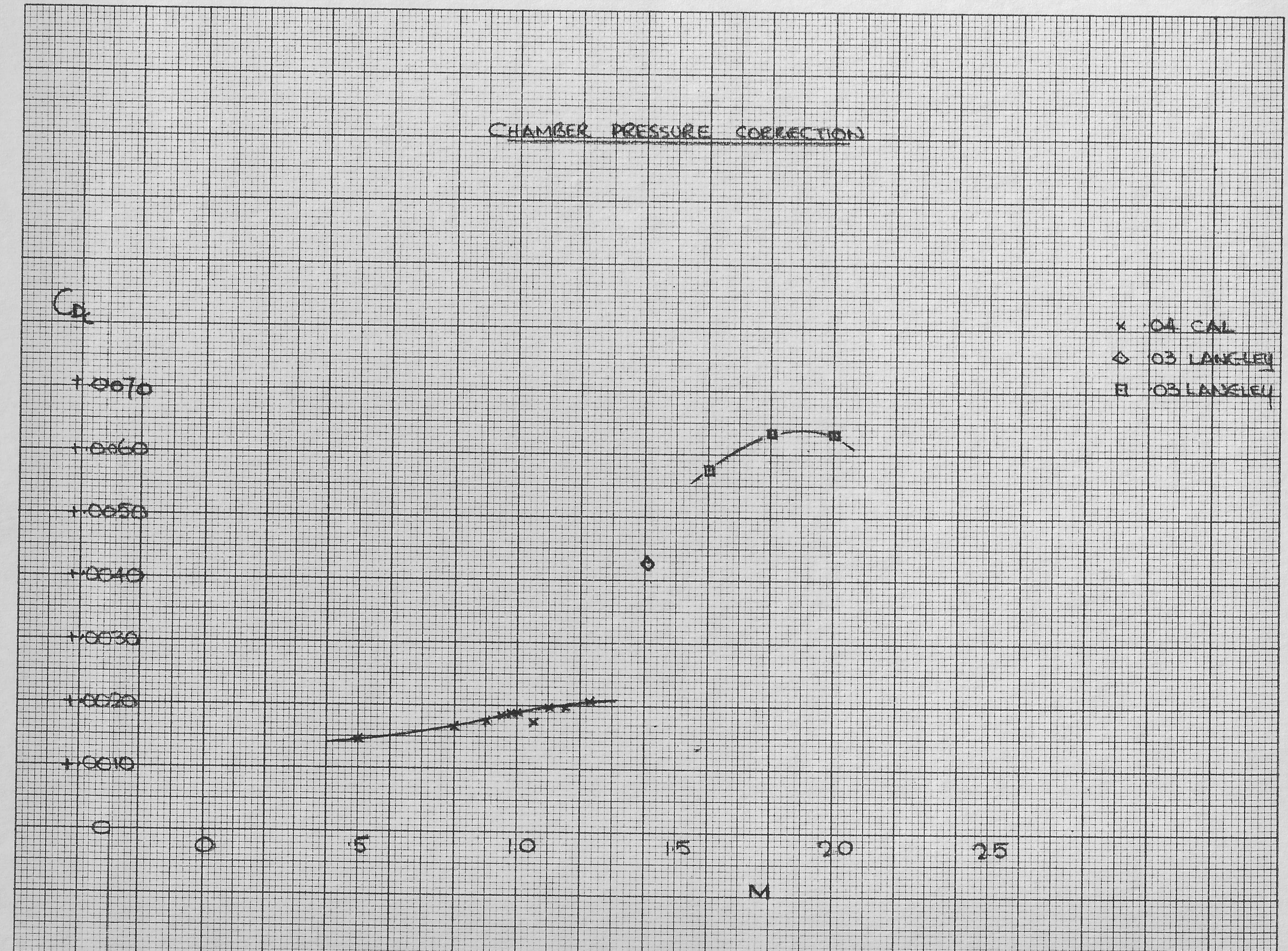
$$m/mo = .5787\left(\frac{A_4}{A_c}\right)\left(\frac{H_4}{H_\infty}\right)\varphi(M_\infty) = .379\left(\frac{H_4}{H_\infty}\right)\varphi(M_\infty)$$

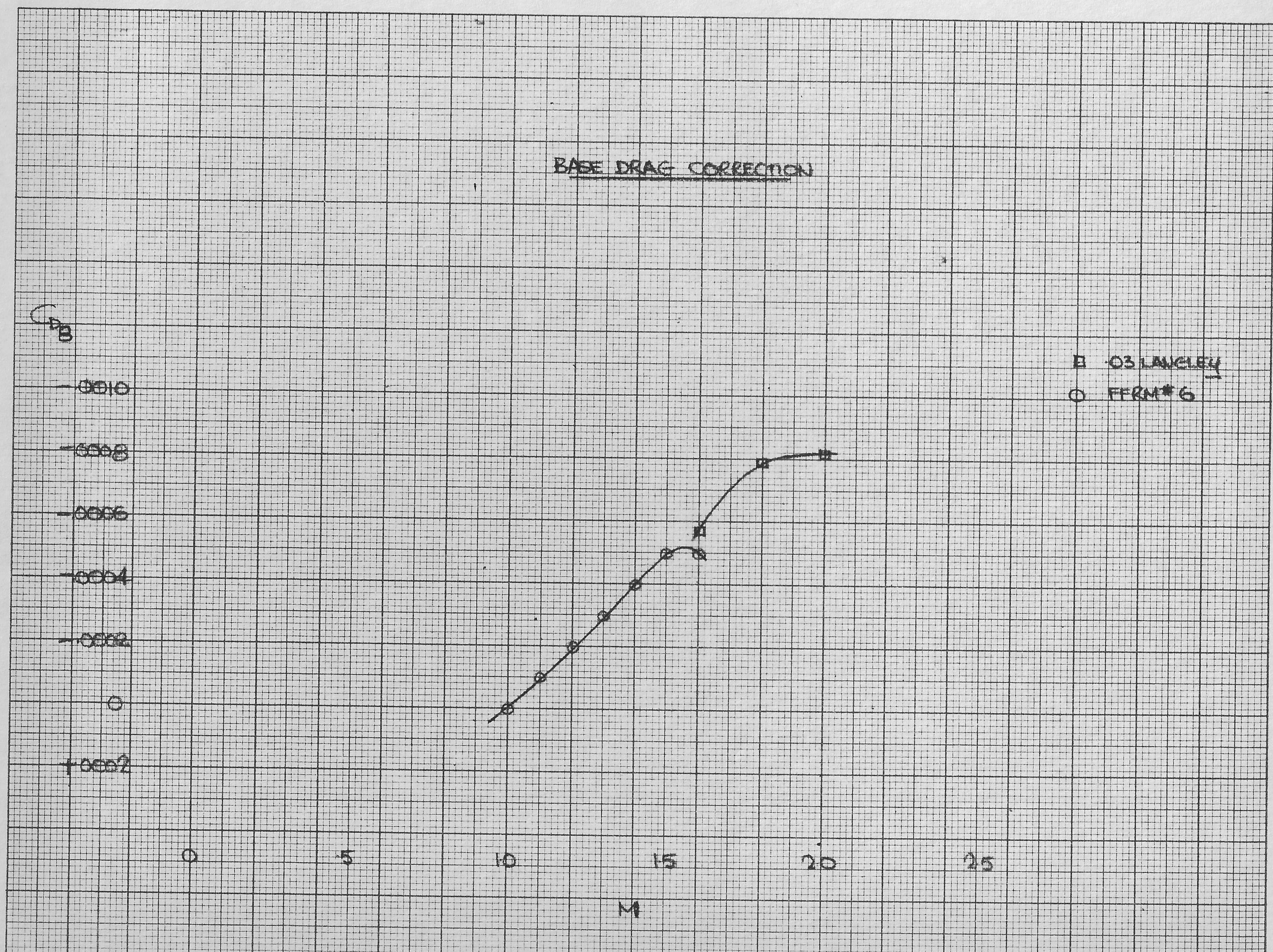
$$C_{D_s} = \frac{\partial C_{D_s}}{\partial m/mo}\left(m^{*}/mo - m/mo\right)\frac{A_f2}{S_w}$$

M	II	C_{D_M}	$\partial C_{Ds}/\partial m/mo$	m^*/mo	C_{Ds}
.5	-0270	-0048	0	.716	0
.8	-0087	-0050	.05	.554	0001
1.0	-0040	-0040	178	.534	0006
1.2	-0010	-0031	260	.548	0012
1.5	+0024	-0023	316	.631	0020
1.8	+0050	-0023	352	.725	0020
2.0	+0060	-0030	370	.801	0022


SPILLAGE DRAG CORRECTIONS
□ O3 LANGLEY
-.0040
-.0030
C_{D_S}
-.0020
-.0010
0
0
.5
1.0
1.5
2.0
2.5
M
ESTIMATE


MOMENTUM DRAG CORRECTION
C_{D_M}
.0070
.0060
.0050
.0040
.0030
.0020
.0010
0
0
.5
1.0
1.5
2.0
2.5
M
ESTIMATE (W.T.M)
FFRM #6
□ ·03 LANGLEY
○ FFRM #6


CHAMBER PRESSURE CORRECTION
C_{D_c}
+.0070
+.0060
+.0050
+.0040
+.0030
+.0020
+.0010
0
0
.5
1.0
1.5
2.0
2.5
M
× .04 CAL
◇ .03 LANGLEY
⊡ .03 LANGLEY



BASE DRAG CORRECTION
C_{D_B}
-.0010
-.0008
-.0006
-.0004
-.0002
0
+.0002
0
.5
1.0
1.5
2.0
2.5
M
.03 LANGLEY
FFRM # 6

Skin Friction Contribution (scale effect)

It is assumed herein that transition to a turbulent boundary layer occurs at transonic speeds due to shock development on all models of this investigation.

At subsonic speeds it is assumed that the fuselage and fin is turbulent but that the wing is laminar to 35% chord.

This assumed skin friction contribution to the measured drag is subtracted to give interference plus wave drag for the model (and for the aircraft). To this interference and wave drag is added the turbulent skin friction drag contribution of the aircraft. A constant aircraft skin friction drag contribution is used for convenience and is assessed at the critical design points - subsonic and supersonic cruise Mach Number and altitude.

For the aircraft:

$$R.N. = \frac{V\bar{\bar{C}}}{\nu} \doteqdot 50 \times 10^6 \text{ at M .92, 40000' and M 1.5, 50000'}$$

$$\therefore C_f = .002 \quad \text{Ref. T.N. 4201, Fig. 2}$$

$$\Delta C_{Df} = C_f \frac{A_{WET}}{S_W} = \frac{.002 \times 4000}{1225} = .0065$$

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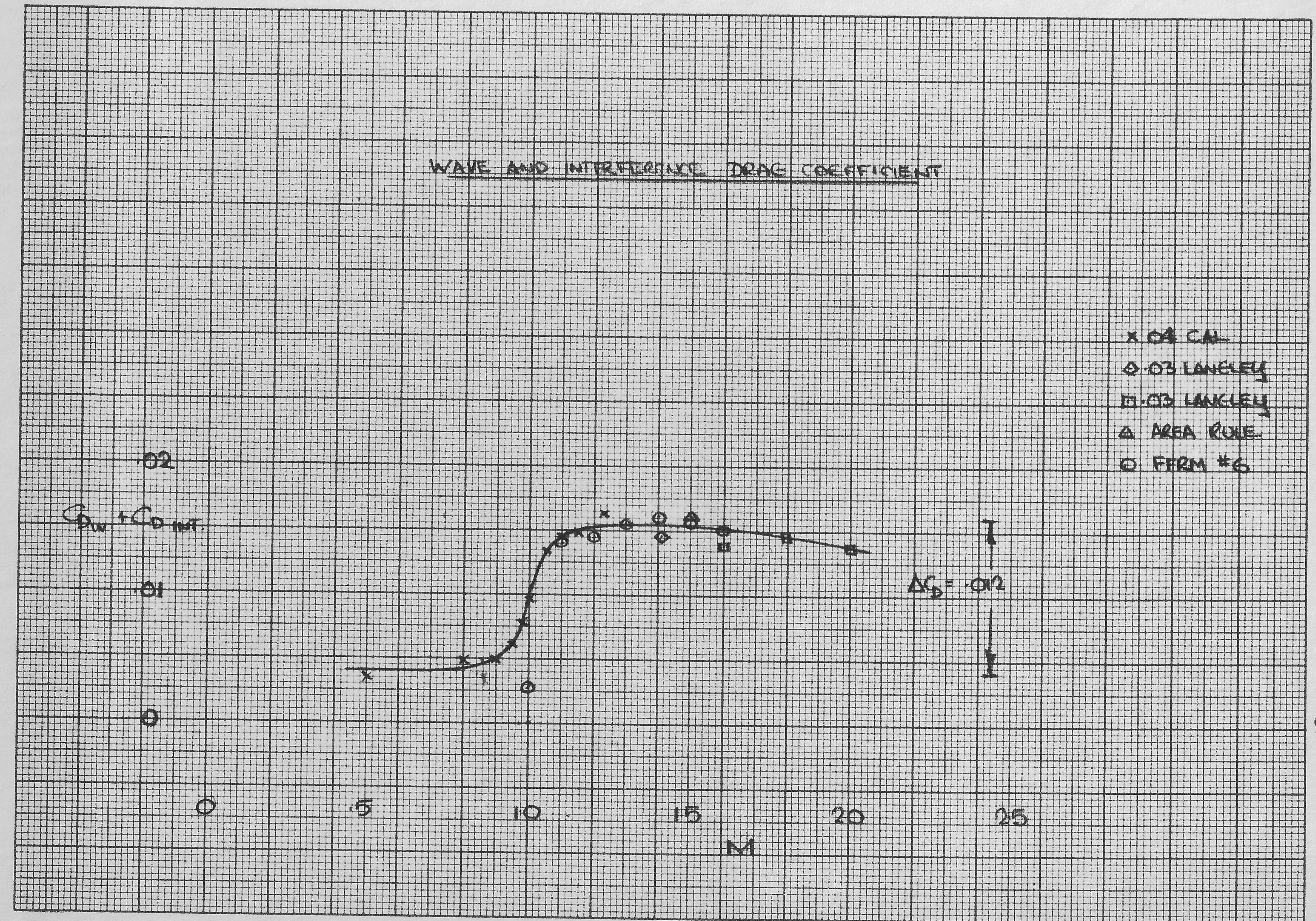
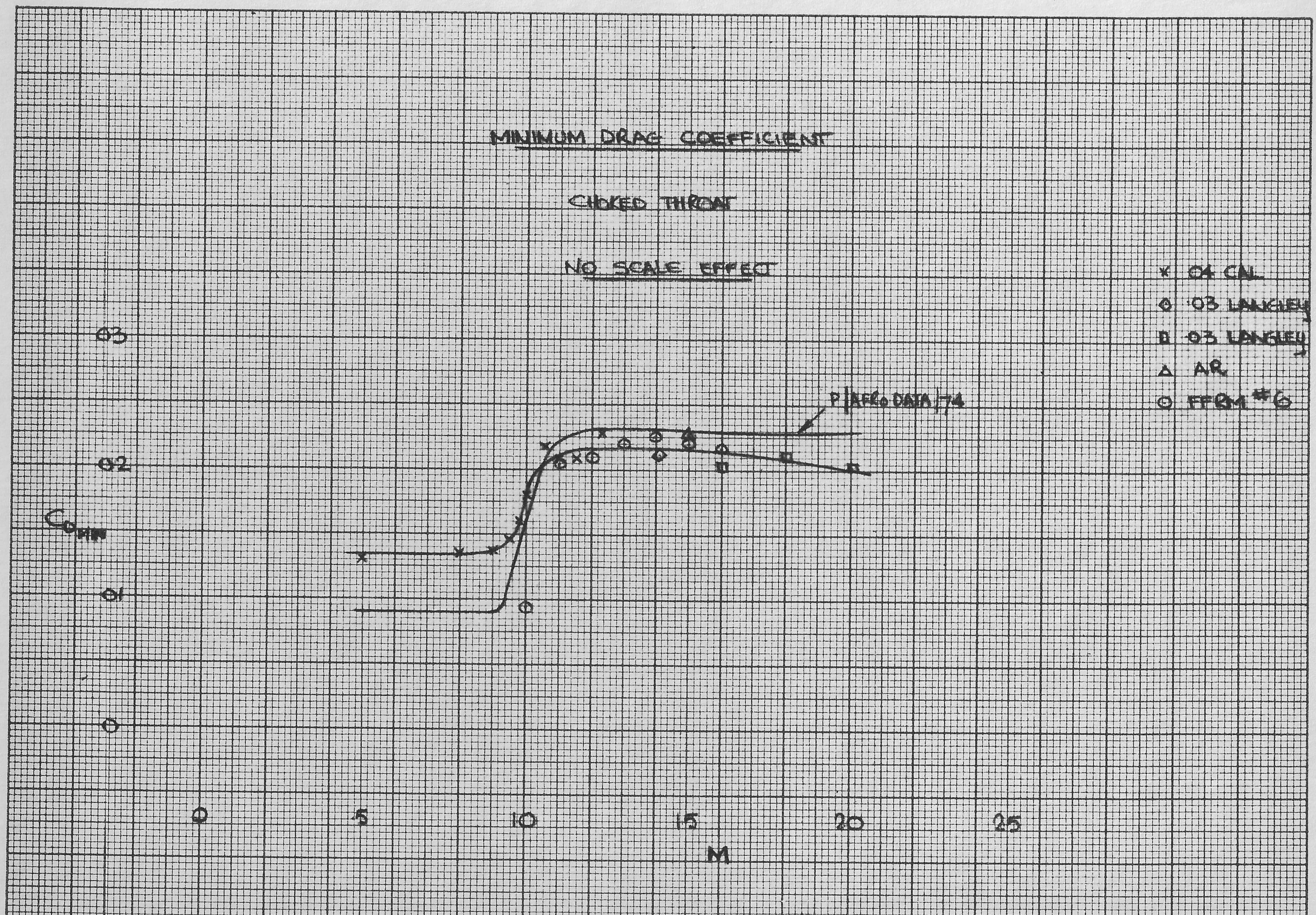
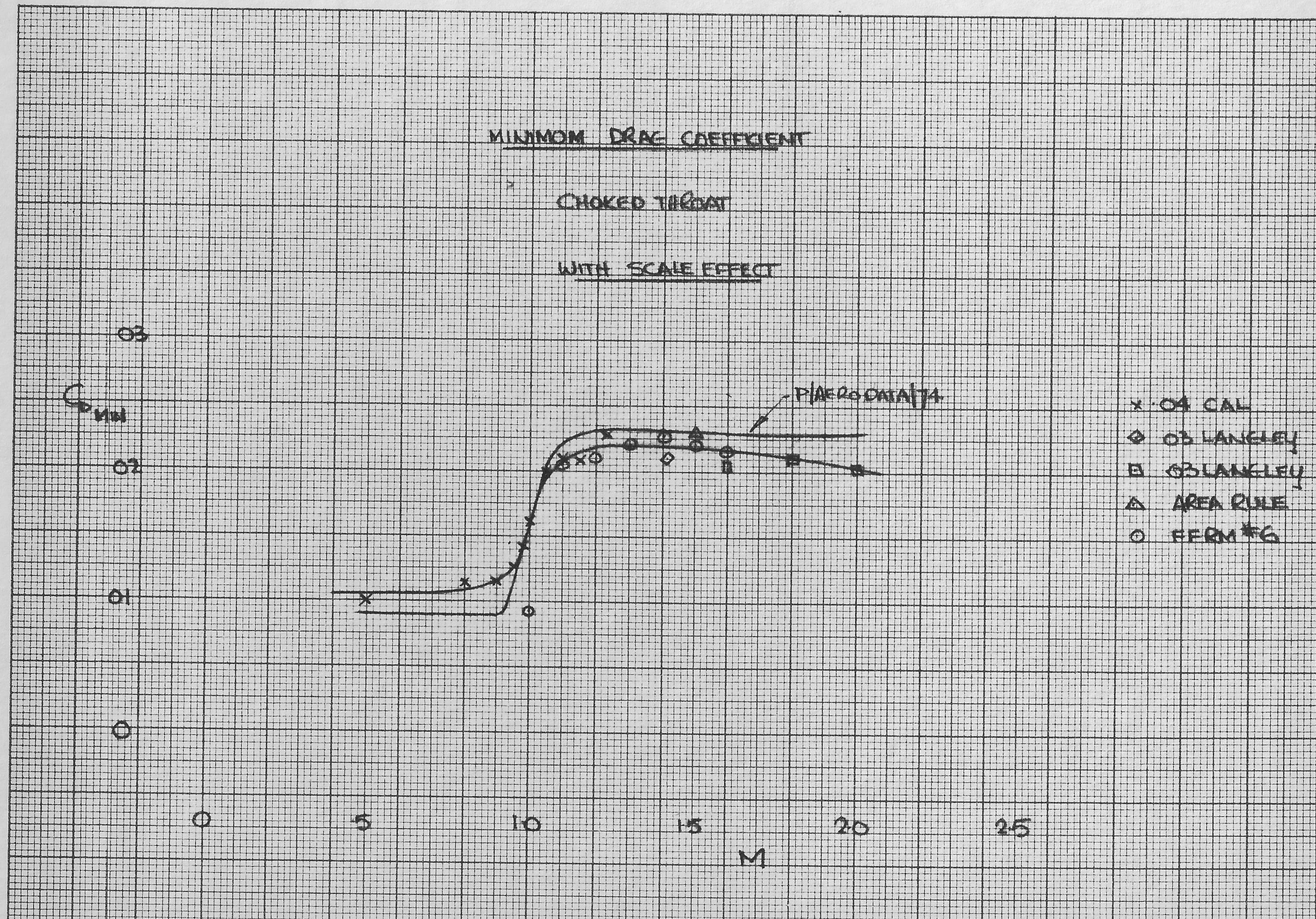
M	$C_{D_{ext}}$	R.N.($\bar{\bar{c}}$) 10^{-6}	C_f (turb)	C_f (lam)	ΔC_{D_f}	$C_{D_{w+int}}$	C_D
.5	0131	1.49	00390	00110	0096	0035	0100
.8	0136	2.03	00360	00094	0087	0049	0114
.9	0137	2.09	00360	00092	0087	0050	0115
.95	0147	2.16	00355	00090	0085	0062	0127
.98	0161	2.16	0035	"	0084	0077	0142
1.0	0180	2.16	0035	"	0084	0096	0161
1.05	0218	2.19	0035	–	0114	0134	0199
1.1	0210	2.22	0035	–	0114	0145	0210
1.15	0209	2.22	00345	–	0113	0144	0209
1.23	0228	2.22	00345	–	0113	0163	0228
1.41	0211	1.74	00350	–	0114	0146	0211
1.5	–	–	–	–	0065	0161	0226
1.6	0203	2.68	0032	–	0105	0138	0203
1.8	0211	2.50	00315	–	0103	0146	0211
2.0	0203	2.31	0031	–	0101	0138	0203
1.6	0217	44.2	00200	–	0065	0152	0217
1.5	0221	39.6	00200	–	0065	0156	0221
1.4	0226	35.6	00205	–	0067	0161	0226
1.3	0220	31.7	00215	–	0070	0155	0220
1.2	0210	28.0	00220	–	0072	0145	0210
1.1	0205	24.5	00230	–	0075	0140	0205
1.0	0094	21.5	00240	–	0078	0029	0094


WAVE AND INTERFERENCE DRAG COEFFICIENT
× .04 CAL.
◇ .03 LANGLEY
⊡ .03 LANGLEY
△ AREA RULE
⊙ FIRM #6
.02
$C_{D_W} + C_{D\,INT.}$
.01
0
$\Delta C_D = .012$
0
.5
1.0
1.5
2.0
2.5
M


MINIMUM DRAG COEFFICIENT
CHOKED THROAT
NO SCALE EFFECT
* .04 CAL
◇ .03 LANGLEY
▫ .03 LANGLEY
△ AR
⊙ FFRM #6
P/AERO DATA/74
.03
.02
.01
0
$C_{D_{MIN}}$
0
.5
1.0
1.5
2.0
2.5
M


MINIMUM DRAG COEFFICIENT
CHOKED THROAT
WITH SCALE EFFECT
P/AERO DATA/74
.03
$C_{D_{MIN}}$
.02
.01
0
0
.5
1.0
1.5
2.0
2.5
M
× .04 CAL
◇ .03 LANGLEY
□ .03 LANGLEY
△ AREA RULE
○ FFRM #6

(f) Estimate of P/Aero Data/27, Feb. 12, 1954. (A. Marshall)

M	$C_{D_{Min}}$	$\partial C_{D_s} / \partial m/mo$	m_i^* / mo	C_{D_A}	$C_{D_{ext}}$	ΔC_D
.5	0087	0	.716	0	0087	0
.7	"	.02	.582	00056	0093	0006
.9	"	.10	.540	00308	0118	0031
.94	"	.128	.537	00397	0127	0040
1.0	0137	.170	.555	00530	0190	0103
1.1	0164	.225	.541	00692	0233	0146
1.2	0159	.260	.555	00776	0237	0150
1.3	0154	.285	.575	00812	0235	0148
1.4	0151	.302	.602	00806	0232	0148
1.5	0148	.316	.632	00779	0226	0139
1.6	0148	.329	.661	00748	0223	0136
1.8	0147	.352	.725	00648	0212	0125
2.0	0147	.371	.801	00495	0197	0110

Adding on the drag of the intake due to spilled air of mass $m_o - m_i^*$

63

$$C_{D_A} = \frac{\partial C_{D_s}}{\partial m/mo} \left(1 - m_i^*/mo\right) 2 \frac{A_f}{S_w}$$

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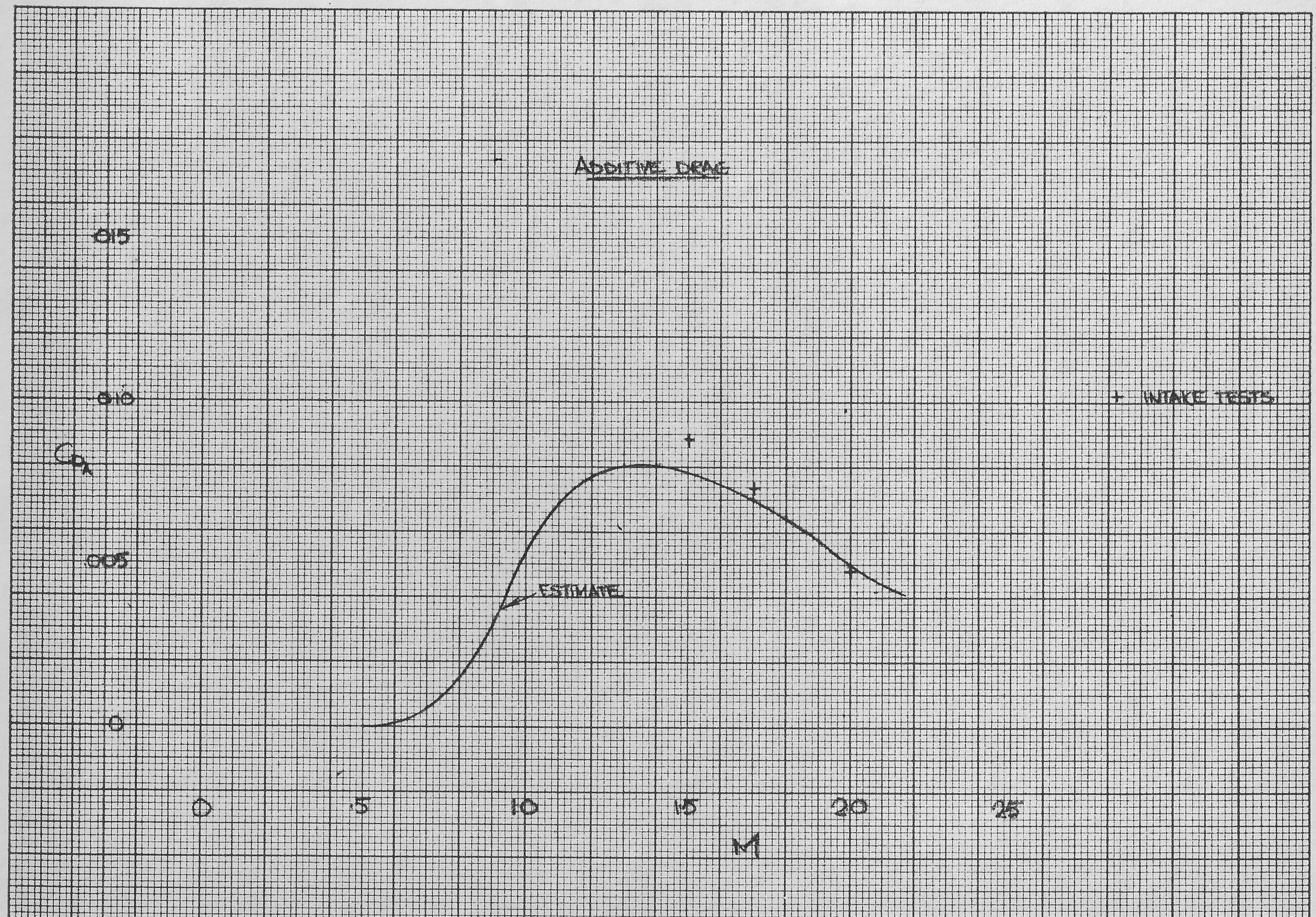
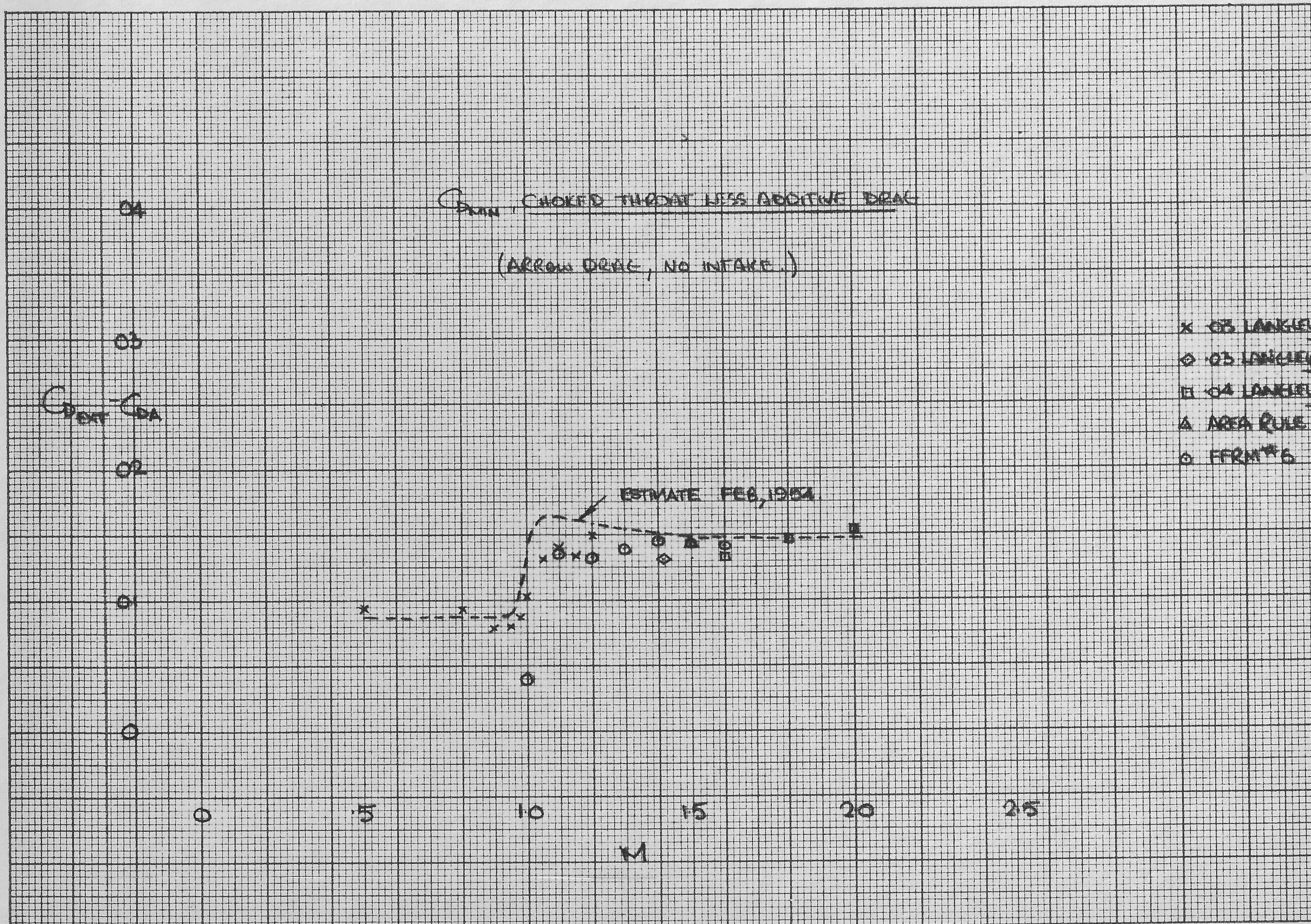
AIRCRAFT:

Additive Drag

M	$C_{D_{ext}}$	m_i^*/mo	$\partial C_{D_s}/\partial m/mo$	$\Delta m/mo$	C_{D_A}	(No Intake) $C_{D_{ext}}$
.5	0131	.716	0	.284	0	0131
.8	0136	.554	.05	.446	0015	0121
.9	0137	.540	.1	.460	0031	0106
.95	0147	.536	.137	.464	0043	0104
.98	0161	.535	.160	.465	0050	0111
1.0	0180	.535	.170	.465	0053	0127
1.05	0218	.537	.201	.463	0062	0156
1.1	0210	.541	.225	.459	0069	0141
1.15	0209	.548	.245	.452	0074	0135
1.23	0228	.561	.266	.439	0078	0150
1.41	0211	.605	.304	.395	0081	0130
1.6	0203	.681	.329	.319	0070	0133
1.8	0211	.725	.352	.275	0065	0146
2.0	0203	.801	.372	.199	0050	0153
1.5	0226	.632	.316	.368	0078	0149
1.6	0217	.681	.329	.319	0070	0147
1.5	0221	.632	.316	.368	0078	0143
1.4	0226	.602	.302	.398	0081	0145
1.3	0220	.575	.285	.425	0081	0139
1.2	0210	.555	.260	.445	0076	0134
1.1	0205	.541	.225	.459	0069	0136
1.0	0094	.535	.170	.465	0053	0041

The Additive Drag gives a measure of the sizeable drag contribution of the intake to the aircraft drag.


ADDITIVE DRAG
·015
·010
·005
0
C_{D_A}
ESTIMATE
+ INTAKE TESTS
0
·5
1·0
1·5
2·0
2·5
M



CDMIN, CHOKED THROAT LESS ADDITIVE DRAG
(ARROW DRAG, NO INTAKE.)
CDEXT - CDA
.04
.03
.02
.01
0
0
.5
1.0
1.5
2.0
2.5
M
ESTIMATE FEB, 1954.
× .03 LANGLEY
◇ .03 LANGLEY
⊞ .04 LANGLEY
△ AREA RULE
⊙ FFRM #6

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AIRCRAFT:

History of Estimated Drag of C105

J.Lucas
Sept. 13/51
P/Aero Data/4

M	
0 → .95	0090
1.0	0138
1.025 → 1.6	01575

J.Lucas
May 28/52
P/Aero Data/6

M	
0 → .95	0087
1.0	0130
1.05	0153
1.1	0154
1.15	0152
1.2	0148
1.3	0144
1.4	0141
1.5	0140
1.6	"
1.7	"

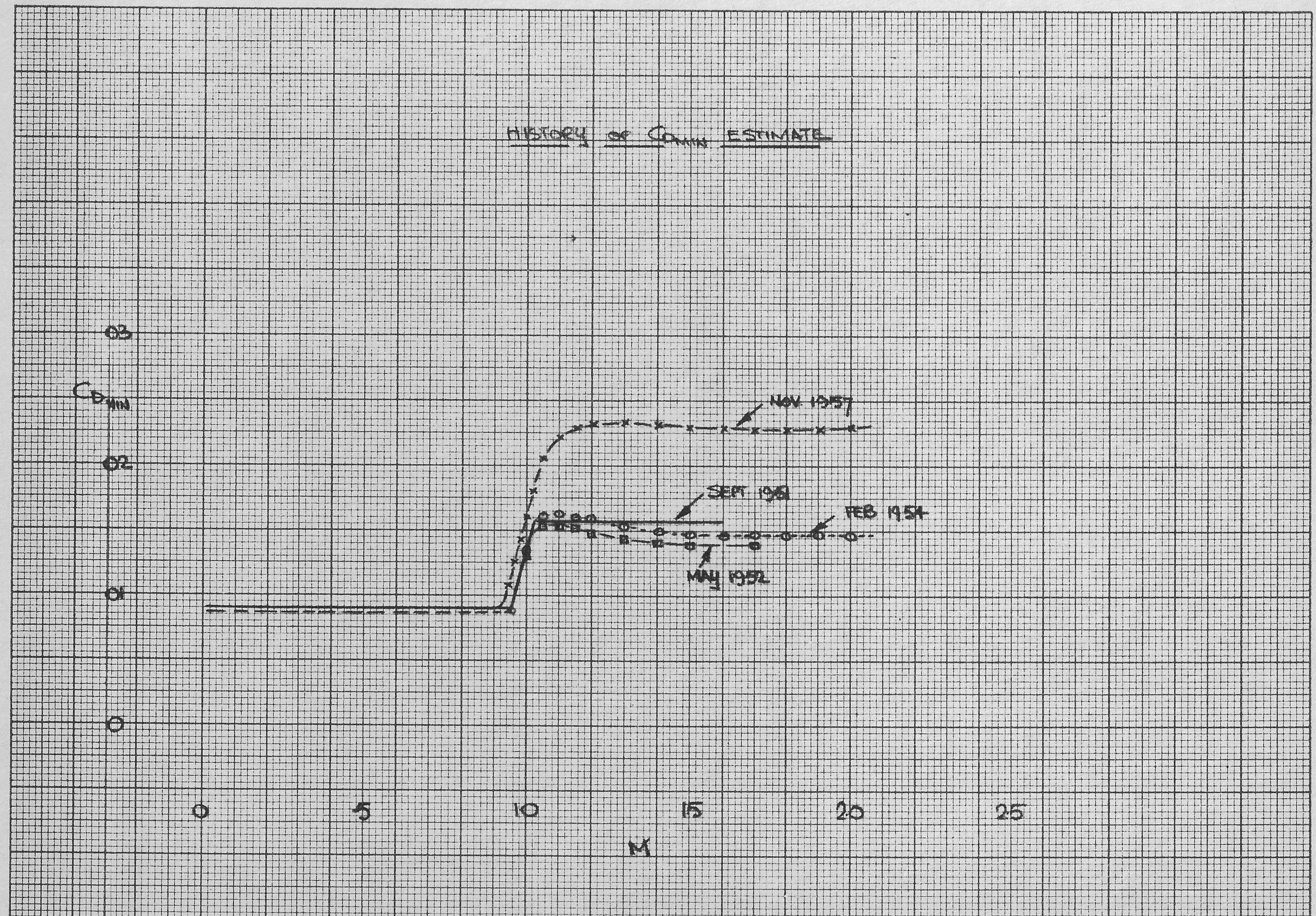
A.Marshall
P/Aero Data/27
Feb. 12/54

M	
0 → .95	00865
1.0	0136
1.05	0162
1.1	01635
1.15	0161
1.2	01585
1.3	0154
1.4	0150
1.5	0148
1.6	0147
1.7	"
1.8	"
1.9	"
2.0	"

Latest drag.

S.Singer
Oct. 1956
P/Aero Data/74

M	$C_{D_{Min}}$
≤ .89	0090
.92	0092
.96	0126
1.0	0160
1.05	0205
1.1	0222
1.15	0228
1.2	0231
1.3	0233
1.4	0232
1.5	0230
1.6	0229
1.7	0228
1.8	0228
1.9	0228
2.0	0230


HISTORY OF $C_{D_{MIN}}$ ESTIMATE
$C_{D_{MIN}}$
.03
.02
.01
0
NOV. 1957
SEPT 1951
FEB 1954
MAY 1952
0
5
10
15
20
25
M

NAE Contribution to Drag

1. NAE-122 31 May 1955 1/80 scale RN $1.15 \rightarrow 1.74 \times 10^6$

M	C_{D_o}	ΔC_D
.53	0083	0003
.57	0100	0020
.60	0087	0007
.63	0083	0003
.675	0078	-0002
.725	0078	-0002
.77	0078	-0002
.83	0083	0003
.845	0090	0010
1.35	0240	0160
1.46	0246	0166
1.57	0228	0148

ΔC_D is the supersonic drag rise.

2. NAE-AE-46h 31 Oct. 56 1/80 Scale R.N. $1.36 \rightarrow 1.68 \times 10^6$

M	C_{D_o}	ΔC_D
1.35	0237	0157
1.57	0212	0132
1.78	0210	0130
2.03	0202	0122

3. Estimate at M 1.4 given in NAE-LR-87, P/Aero Data/48, March 1954

	NAE	Avro
Body	0094	0065
Wing	0048	0047
Fin	0008	0010
Canopy	0014	0010
Miscellaneous	0009	0010
Spillage	0008	-
Friction	0062	0062
C_{D_o} =	0235	0204
ΔC_D =	0155	0124

The NAE estimate has been altered to give C_{D_o} based on 1225 sq. ft.

4. NAE-AE-46f October 28, 1958 1/50 Scale, Reflection Plane Model

R.N. 2.0 → 2.83

M	ΔC_D
1.22	0123
1.35	0138
1.57	0115
1.78	0120
2.03	0080

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AIRCRAFT:

5. F102-A t/c 4% $C_L = .07$ AFTR-AFFTC-54-14

M	C_D	C_{D_o}	ΔC_D
.694	0102	0092	-0006
.74	0112	0102	0004
.79	0108	0098	0
.803	0106	0096	-0002
.857	0109	0099	0001
.874	0107	0097	-0001
.9	0114	0104	0006
.92	0116	0106	0008
.925	0111	0101	0003
.925	0122	0112	0014
.939	0131	0121	0023
.95	0187	0177	0079
.95	0198	0188	0090
.975	0253	0243	0145
.104	0284	0274	0176
1.052	0287	0277	0179
1.066	0291	0281	0183
1.072	0291	0281	0183
1.099	02905	02805	01825
1.113	0287	0277	0179
1.130	0284	0274	0176
1.148	02795	02695	01715
1.167	0276	0266	0168
1.18	02735	02635	01655

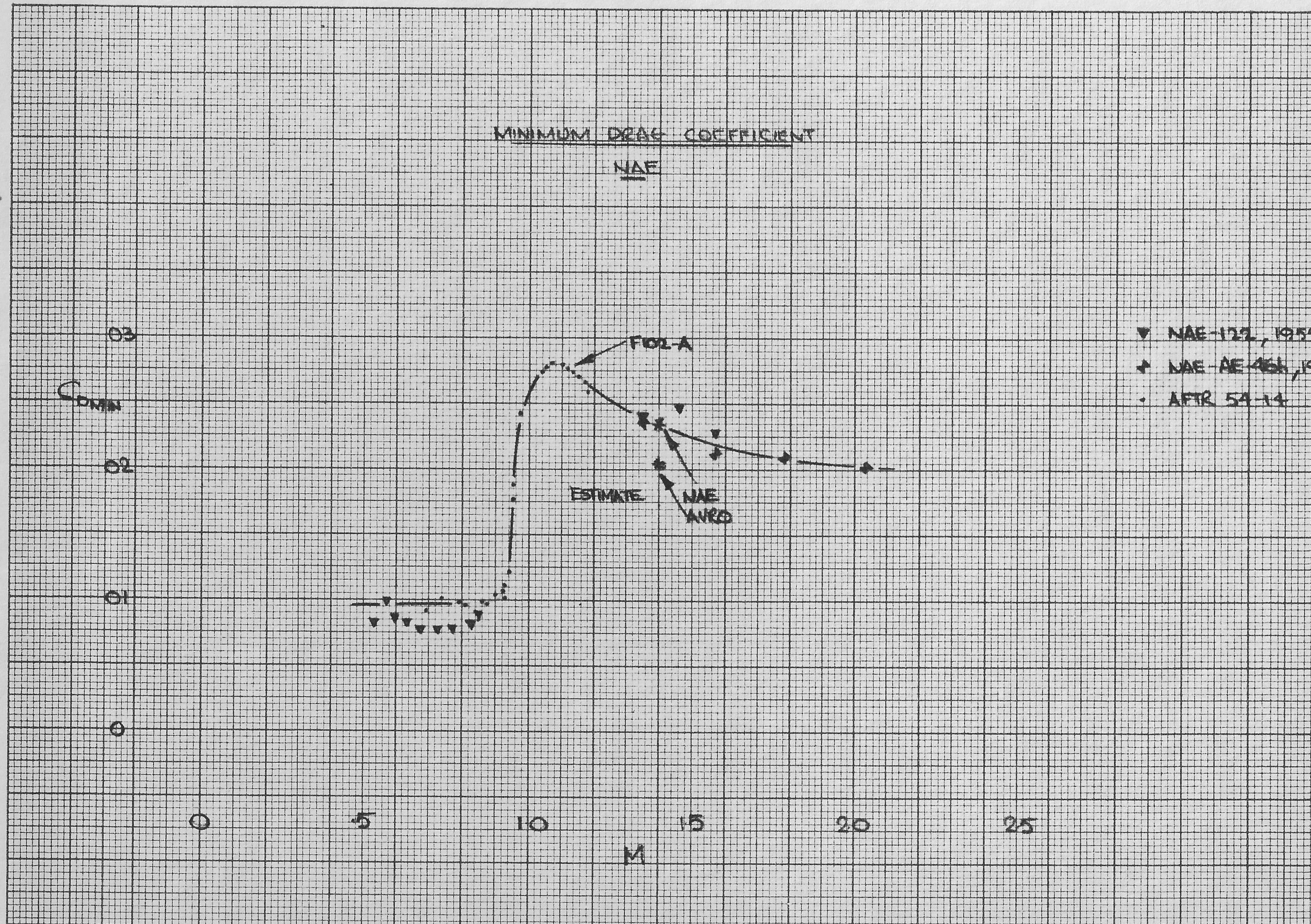
$C_{D_o} \div C_D - .0010$

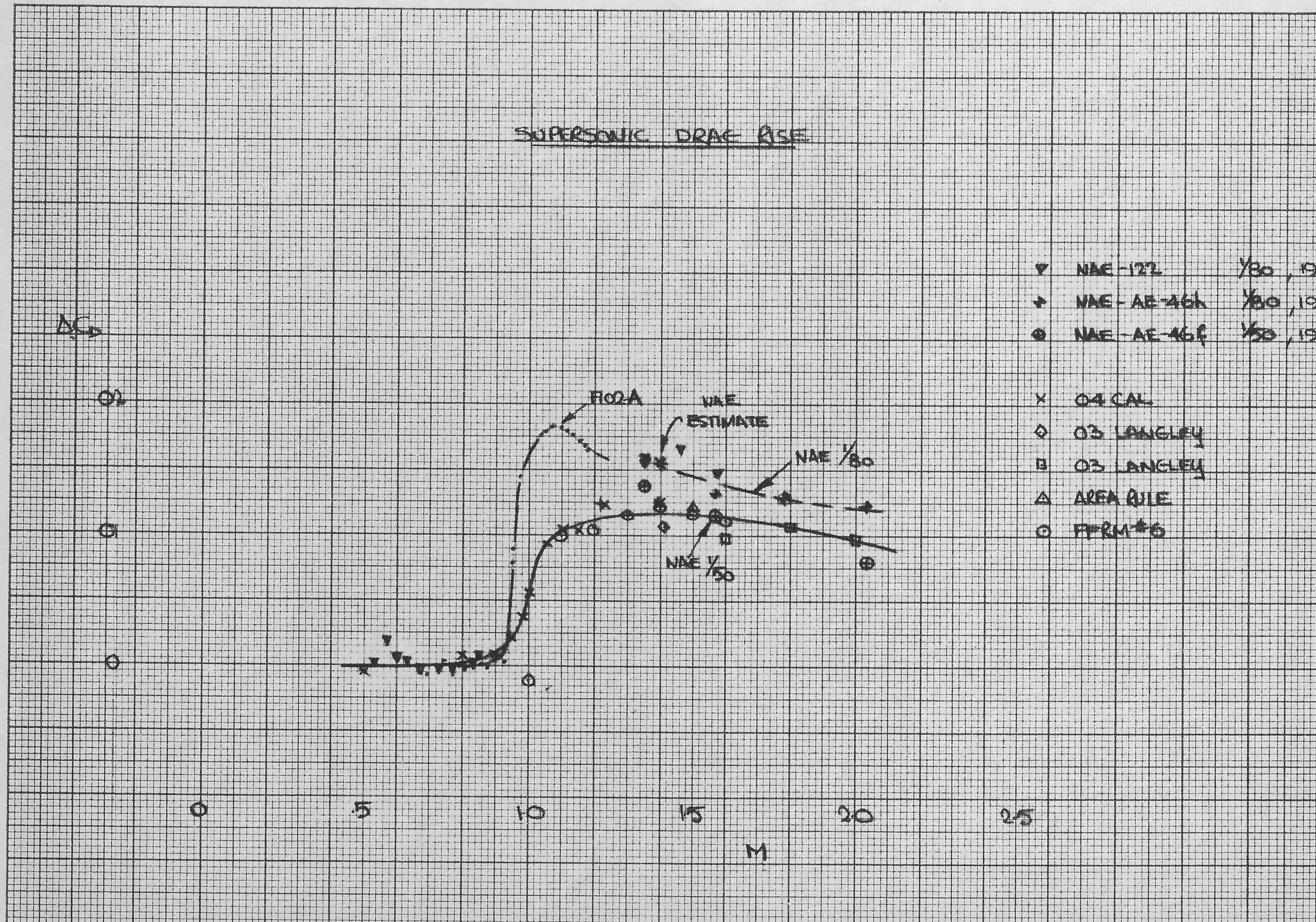
6. Avro WTM and FFRM Program:

	M	C_{D_o}	ΔC_D
.04 CAL	.5	0100	-0005
	.8	0114	0009
	.9	0115	0010
	.95	0127	0022
	.98	0142	0037
	1.0	0161	0056
	1.05	0199	0094
	1.1	0210	0105
	1.15	0209	0104
	1.23	0228	0123
.03 Langley	1.41	0211	0106
Area Rule	1.5	-	0121
.03 Langley	1.6	0203	0098
	1.8	0211	0106
	7.0	0203	0098
FFRM	1.6	0217	0112
	1.5	0221	0116
	1.4	0226	0121
	1.3	0220	0115
	1.2	0210	0105
	1.1	0205	0100
	1.0	0094	-0011

It can only be concluded from attached graphs that the NAE estimate and experimental data were in good agreement and of the right order for the early Arrow configuration.

Attention to detail, such as improved nose and ejector, as predicted by such methods as Area Rule and confirmed in FFRM tests has reduced the drag of the Arrow in the interim period to agree with Avro's early estimate.


MINIMUM DRAG COEFFICIENT
NAE
$C_{D_{MIN}}$
.03
.02
.01
0
0
.5
1.0
1.5
2.0
2.5
M
F102-A
ESTIMATE
NAE
AVRO
NAE-122, 1955
NAE-AE-, 1956
AFTR 54-14


SUPERSONIC DRAG RISE
ΔC_D
.02
.01
0
0
.5
1.0
1.5
2.0
2.5
M
F102A
NAE ESTIMATE
NAE 1/80
NAE 1/50
NAE-122 1/80, 1955
NAE-AE-46h 1/80, 1956
NAE-AE-46f 1/50, 1958
04 CAL
03 LANGLEY
03 LANGLEY
AREA RULE
FFRM #6

Comparison of Drag Coefficient as estimated from W.T.M. and FFRM to the equivalent body of revolution drag:

The equivalent body models are Model No. 131 of NACA TN 4201 and the Hustler, of NACA RM L57G29.

The two models must be connected for base drag, fin wave and friction drag, and body friction drag. This nets body pressure drag to which can be added the aircraft nominal friction drag.

$$C_D = C_{D_{Total}} - C_{D_B} - C_{D_{Fin}}$$

$$= C_{D_{Total}} \frac{A_X}{1225} - C_{p_B} \frac{A_B}{A_X} \frac{A_X}{1225} - C_{D_w} \left(\frac{S_{fin}}{A_X}\right) \frac{A_X}{1225} - C_{D_f} \times \frac{S_{fin}}{A} \frac{A}{1225}$$

for Model 131

$$C_D = \overset{I}{C_{D_{Total}} \frac{74}{1225}} - \overset{II}{C_{p_B} \; .20 \times \frac{74}{1225}} - \overset{III}{C_{D_N} \frac{11.86 \times 74}{2 \times 1225}} - \overset{IV}{C_f \frac{11.86 \times 74}{1225}}$$

$$\overset{V}{- C_f \times \frac{21.35 \times 74}{1225}} + .0065$$

for Hustler:

$$C_D = C_{D_{Total}} - \overset{I}{C_{p_B} \frac{A_B}{S_W}} - \overset{II}{C_{D_w} \frac{A_{Fin}}{S_W}} - \overset{III}{C_F \frac{A_{fin\ wet}}{S_W}}$$

$$S_W = 6.050$$

$$A_B = \frac{\pi \times 2.80^2}{4 \times 144} = .04275$$

$$A_{Fin} = \left(\frac{4.42 + 10.55}{2}\right)\left(\frac{18.20}{2} - 1.40\right) = .400\square'$$

x/L	x	A	A'	A'/A'max.	r/R
0	0	10.8	0	0	0
05	45	18.5	7.4	.1256	.354
.1	90	23.5	12.1	.2055	.453
.15	135	27.5	15.8	.268	.517
.2	180	33.3	21.3	.362	.602
.25	225	47.0	34.7	.589	.766
.3	270	53	40.5	.687	.828
.35	315	55.3	42.5	.722	.85
.4	360	57	43.9	.745	.864
.45	405	60.3	46.9	.796	.890
.5	450	64.0	50.3	.855	.923
.55	495	69.0	55.0	.933	.963
.6	540	72.3	58.0	.985	.99
.65	585	73.4	58.8	.999	.999
.7	630	73.8	58.9	1.0	1.0
.75	675	70.6	55.4	.941	.968
.8	720	65.0	49.6	.841	.917
.85	765	55.0	39.3	.667	.816
.9	810	34.0	18.0	.306	.553
.95	855	17.5	1.2	.204	.452
1.0	900	16.6	0	0	0

L = 900"

$$A \text{ free stream} = \left(\frac{H_i}{H_\infty}\right) A_i \times .97 = .99 \times 5.6 \times .97 = 10.75'$$

$$A \text{ stream tube, ejector} = \frac{\pi}{4}\,\frac{39^2}{144} = 16.57\,\square' \quad \text{Arrow 1}$$

$$= 22.1\,\square' \quad \text{Arrow 1A}$$

A' is Arrow cross-sectional area with stream tube area subtracted.

x/L	A/L²	$\sqrt{A/L^2}$	r/R	Model 131
	← Hustler		→	
0	0	0	0	0
.5	.04	.2	.1755	.27
1	.11	.331	.2905	.44
1.5	.23	.479	.420	.525
2	.37	.608	.534	.595
25	.52	.720	.631	.795
3	.69	.830	.728	.795
35	.76	.870	.763	.82
4	.83	.910	.799	.83
45	1.02	1.01	.886	.86
5	1.22	1.105	.97	.895
55	1.24	1.113	.977	.925
6	1.27	1.125	.987	.96
65	1.30	1.14	1.0	.985
7	1.21	1.10	.965	1.0
75	1.04	1.02	.895	.99
8	.93	.964	.845	.955
85	.82	.905	.794	.875
9	.55	.742	.651	.745
95	.25	.5	.438	.585
1.0	.15	.387	.340 – Base Drag	.455 – Base Drag

$$\frac{A}{L} = \frac{\pi r^2}{L^2}$$

$$\frac{r}{R} = \sqrt{\frac{A}{\pi}}\Big/R = \sqrt{\frac{A}{L^2}} \times \sqrt{\frac{L^2}{\pi}}\Big/R^2$$

$$= K\sqrt{\frac{A}{L^2}} \quad \text{where} \quad K = \sqrt{\frac{L^2}{R^2\pi}} = \frac{L}{R\sqrt{\pi}} = \sqrt{\frac{1}{1.3\times10^{-2}}}$$

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AIRCRAFT:

Model No. 131

M	$C_{D_{Total}}$	C_{p_B}	C_{D_w}	I	II	III	IV
.8	.146	.135	0	00882	00163	0	00223
.85	.150	"	"	00906	"	"	00222
.9	.161	"	"	00972	"	"	00218
.95	.212	.14	0004	01280	00169	00015	00216
1.0	.395	.20	0054	02385	00242	00194	00212
1.05	.425	.21	0038	02565	00254	00136	00207
1.1	.425	.20	0037	"	00242	00133	00204
1.15	.425	.19	"	"	00230	"	00200
1.2	.423	"	"	02555	"	"	00199
1.25	.420	"	"	02535	"	"	00197

M	V	C_D	R.N.	C_f
.8	00402	0074	4.5	00312
.85	00400	0077	4.8	00310
.9	00394	0085	5.1	00305
.95	00389	0114	5.4	00302
1.0	00382	0201	5.73	00296
1.05	00374	0224	6.03	00290
1.1	00367	0227	6.35	00285
1.15	00361	0232	6.65	00280
1.2	00358	0229	6.95	00278
1.25	00355	0227	7.3	00275

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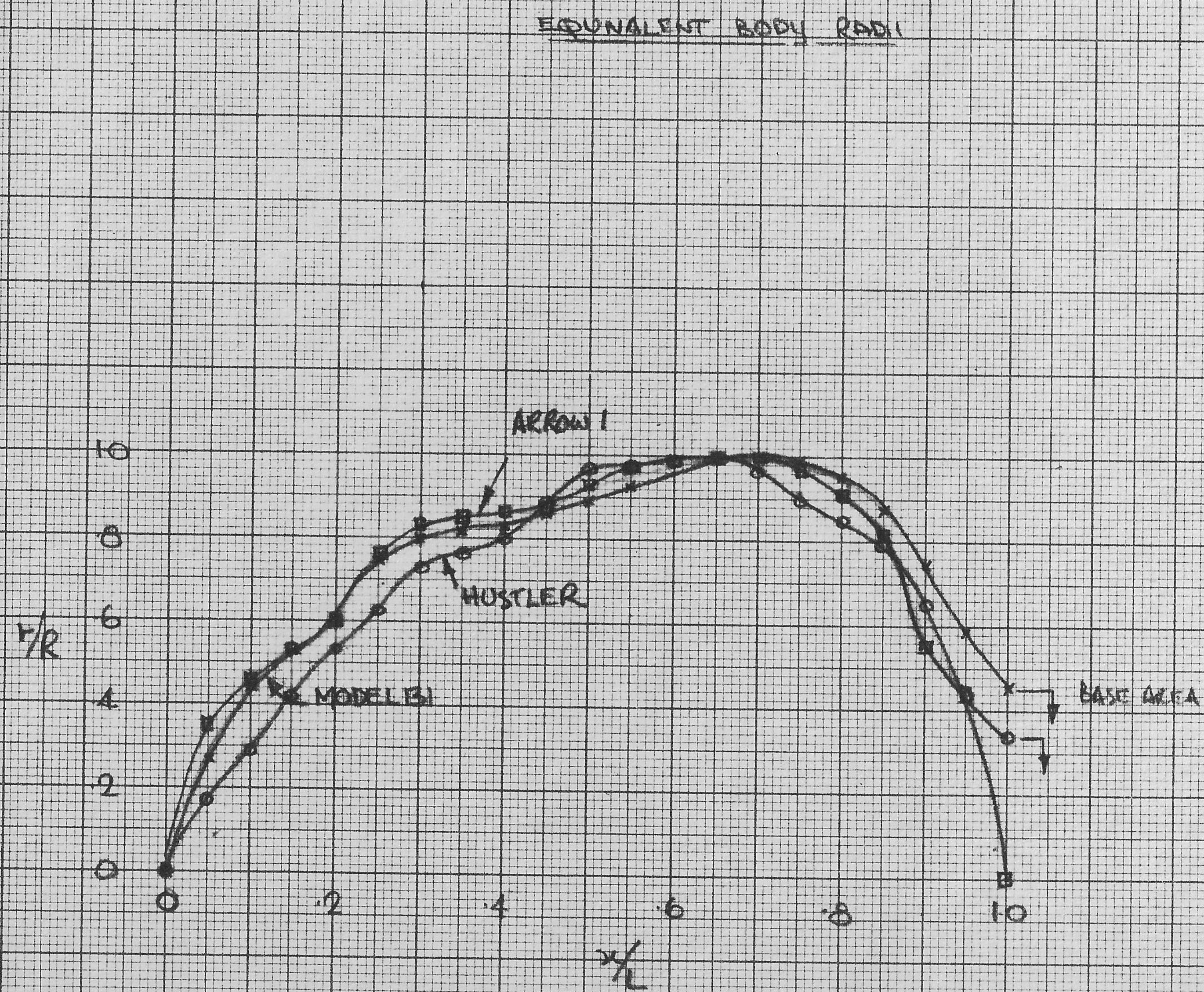
CHECKED BY	DATE

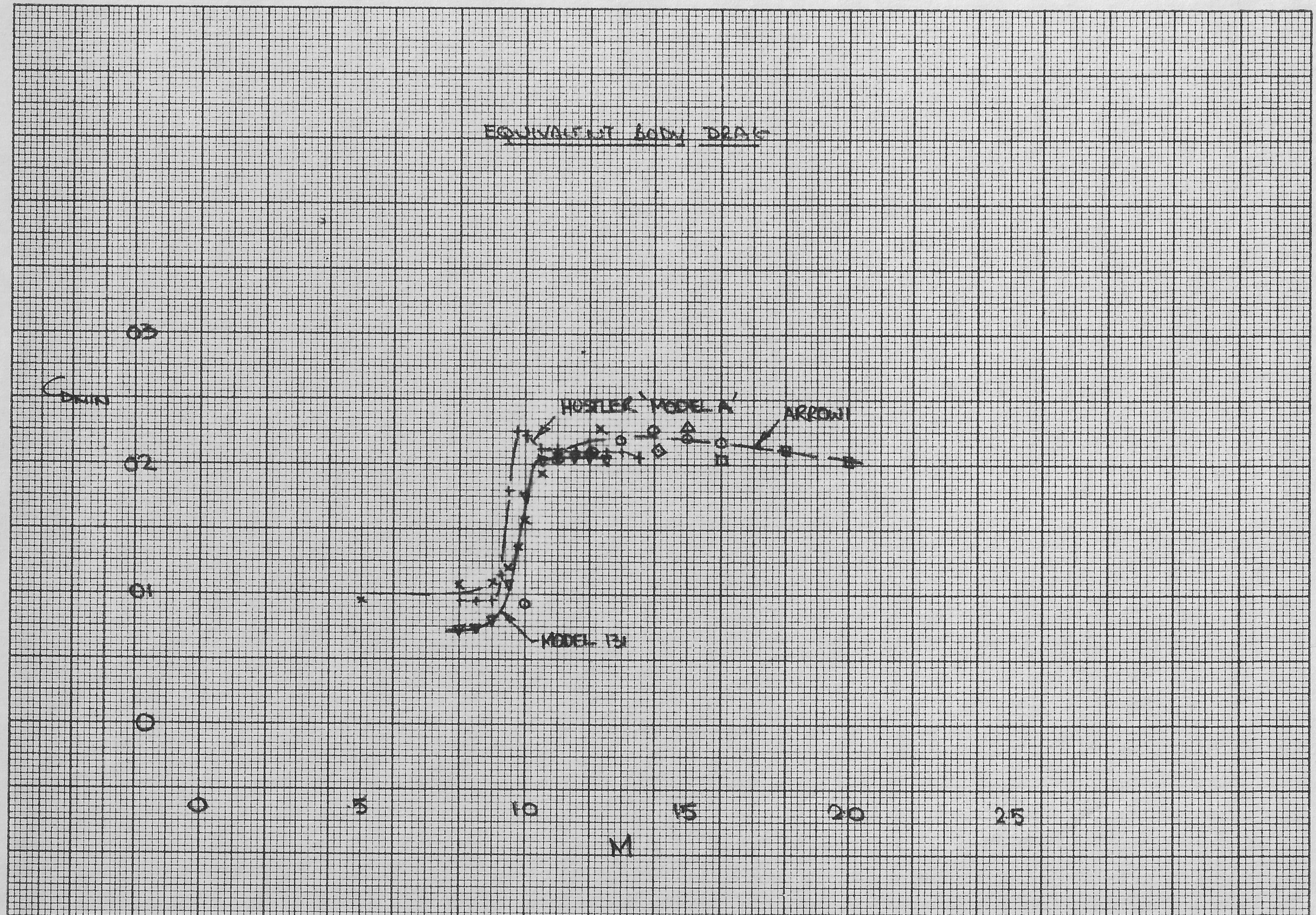
AIRCRAFT:

Hustler:

M	$C_{D_{Total}}$	C_{p_B}	C_{D_w}	I	II	R.N. x 10^6
.8	.0105	.135	0	.0010	0	9.3
.85	"	"	"	"	"	10.1
.9	"	"	"	"	"	11.0
.91	"	"	"	"	"	11.3
.925	.0125	"	.0001	"	"	11.5
.95	.0190	.14	.0004	"	"	12.0
.975	.0238	.175	.002	.0012	.0001	12.5
1.	.0240	.2	.0054	.0014	.0004	13.0
1.05	.0230	.21	.0038	.0015	.0003	13.7
1.1	.0228	.2	.0037	.0014	.0002	14.7
1.15	.0225	.19	"	.0013	"	15.5
1.2	"	"	"	"	"	16.5
1.25	"	"	"	"	"	17.4
1.3	"	"	"	"	"	18.2
1.35	.0221	.187	"	"	"	19.1

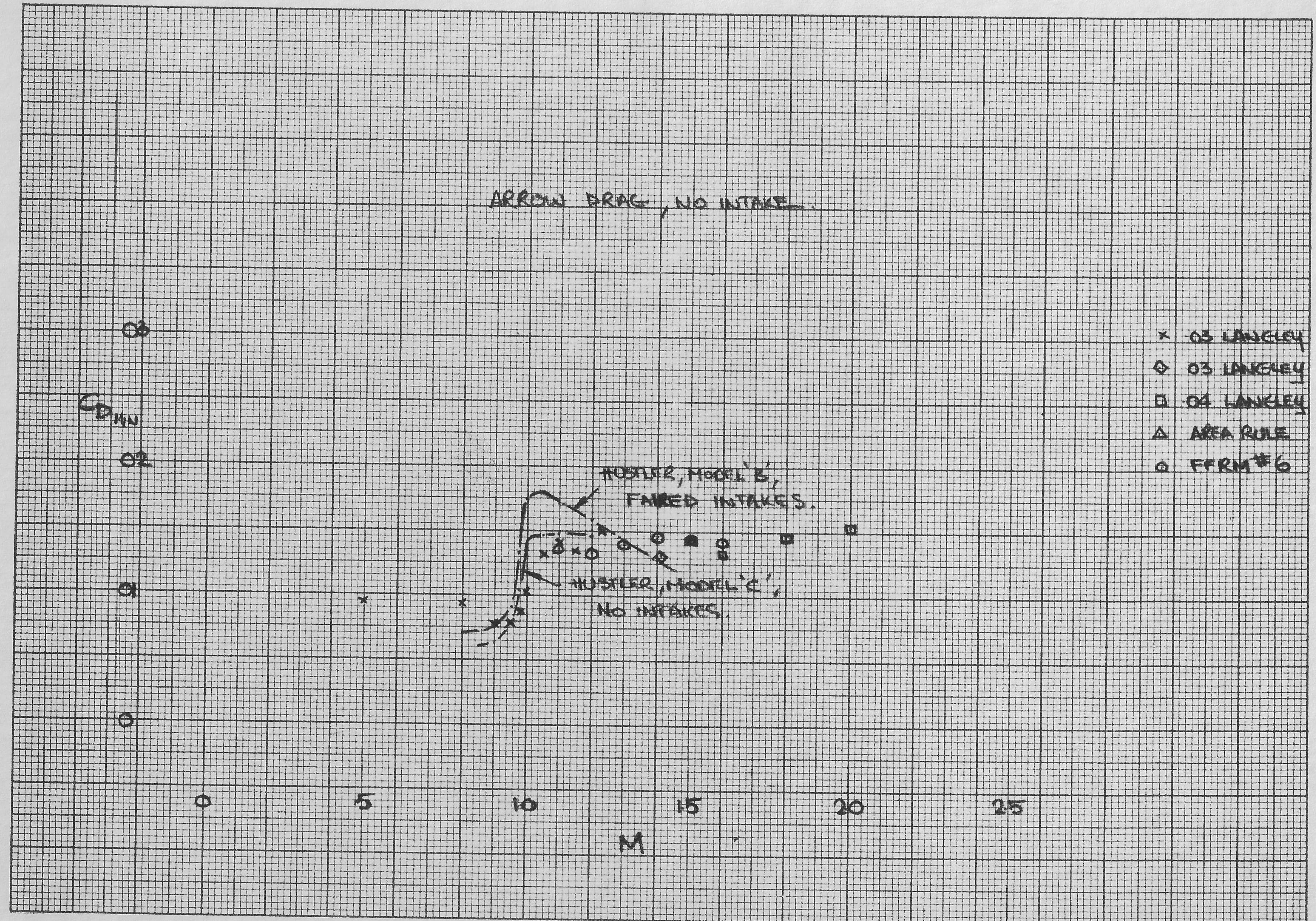
M	C_f	111	C_D
.8	.0028	00037	0091
.85	00275	00036	"
.9	0027	"	"
.91	"	"	"
.925	00262	00035	0111
.95	"	"	0175
.975	0026	00034	0222
1	00252	"	0219
1.05	00255	"	0209
1.1	0025	00033	"
1.15	00247	"	0207
1.2	00245	00032	"
1.25	0024	"	"
1.3	00238	00031	"
1.35	00235	"	0203


EQUIVALENT BODY RADII
ARROW I
HUSTLER
MODEL B1
BASE AREA
r/R
x/L
0
·2
·4
·6
·8
1·0
10
8
6
4
2
0


EQUIVALENT BODY DRAG
C D MIN
.03
.02
.01
0
0
.5
1.0
1.5
2.0
2.5
M
HOSTLER 'MODEL A'
ARROW I
MODEL B

Correction to Hustler Drag, No intake.

M	Model B $C_{D_{Total}}$	Model B ΔC_D	Model B C_D	Model C $C_{D_{Total}}$	Model C C_D
.8	0085	0014	0071	–	–
.85	0086	"	0072	0075	0061
.9	0090	"	0076	0078	0064
.95	0105	"	0091	0092	0078
.975	0150	0016	0134	0120	0104
1.0	0195	0021	0174	0162	0141
1.05	0201	"	0180	0168	0147
1.1	0190	0019	0171	"	0149
1.2	0177	0018	0159	0165	0147
1.3	0162	"	0144	–	–
1.4	0146	"	0128	–	–


ARROW DRAG, NO INTAKE.
.03
.02
.01
0
$C_{D_{MIN}}$
0
5
10
15
20
25
M
HUSTLER, MODEL 'B',
FAIRED INTAKES.
HUSTLER, MODEL 'C',
NO INTAKES.
× .03 LANGLEY
◇ .03 LANGLEY
□ .04 LANGLEY
△ AREA RULE
○ FFRM #6

Conclusions

Included in this composite report is a new spillage drag curve and new external drag curve.

I The new spillage drag curve was derived on the basis of the following assumptions;

1) All spillage is through the normal shock only, with slope given for $m = m^*$.
2) Change of ramp drag with reduced intake mass flow to be derived from Schlieren.
3) Change of fuselage drag with engine mass flow is negligible.
4) Lip suction relief to be taken from Ref. 3.

The new spillage drag curve shows good agreement with the Arrow 1 intake tests and also the curve as suggested by N.A.A. representatives from their intake tests on the 'Vigilante'.

II The new external drag curve was derived from all available drag data on W.T.M., FFRM and available Area-Rule estimates.

The drag data, when properly corrected by the methods outlined in the appendix, collapse on a single curve with acceptable scatter. This weighted curve indicates a reduction from the estimated external drag curve used in Arrow 1 performance analysis at Supersonic speeds but an increase at subsonic speeds.

When the additive drag due to the intake is subtracted from the aircraft drag very good agreement is made with the original Arrow external drag estimate made in Feb. 1954.

Comparison to FFRM drag data of the 'Hustler' and a model of similar cross-section shows good agreement. The large contribution to the aircraft drag due to intakes is again evident.

Because the WTM were primarily stability and control models certain assumptions outlined in the text had to be made in deriving the magnitude of the corrections to the measured drag. It is suggested that in future because of the powerful contribution of the intake to C_{M_0}, for example, (capture area is 28% of aircraft cross sectional area) that more attention be paid to internal flow.

The estimated improvement in external drag due to enlarged ejector of the Arrow 1A configuration is $\Delta C_D = .0011$.

The NAE estimate and experimental data were in excellent agreement and of the right order for the earliest Arrow configuration. Attention to detail, such as improved nose and ejector, as predicted by Area Rule and confirmed in FFRM tests has reduced the drag of the Arrow 1 in the interim period to agree with Avro's early estimate.

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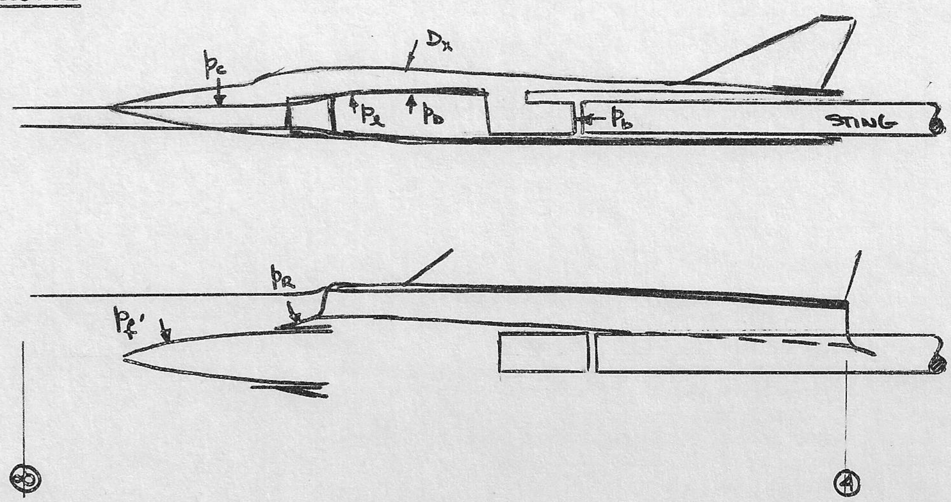
The method to correct the W/T, F/F, and intake models to give the external drag and spillage drag.

a) W/T Model



$$D_{ext} = +\int p_c dA + \Sigma D_x$$

$$D_{int} = -\int p_c dA + \int p_R A_r - \int p_d A_d + \int p_1 A_1 + \int p_f' dA$$

$$X_{Sting} = \Sigma D_x - \int p_D A_D + \int p_R A_R - \int p_b dA + \int p_1 A_1 + \int p_f' dA$$

We measure:

$$m^* (V_4 - V_\infty) + (p_4 - p_\infty) A_4 = \int p_1 dA - \int p_R A_R + \int p_D A_D - \int p_1 A_1 - \int p_f' dA$$

$$\therefore \quad D_{ext} = \int p_c dA + \Sigma D_x$$

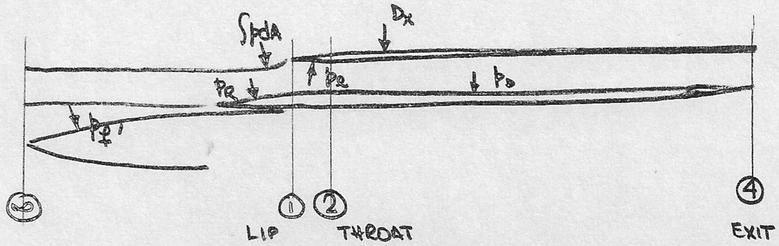
$$= m^* (V_4 - V_\infty) + (p_4 - p_\infty) A_4 + \int p_R A_R - \int p_d A_d + \int p_1 A_1 + \int p_f' dA + \Sigma D_x$$

$$= X_{Sting} + \int p_b dA + m^* (V_4 - V_\infty) + (p_4 - p_\infty) A_4$$

Or in coefficient form:

$$C_{D_{ext}} = C_{D_{X_{Sting}}} + C_{D_b} + C_{D_M} \qquad C_D = \frac{D}{q\, S_W}$$

(b) Free-Flight Models



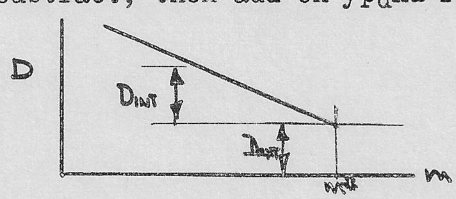
The free-flight model is not utilized to assess the spillage drag, but to give the total external drag, as defined i.e. the shock structure and internal flows should simulate choked thrust flows. The total change of momentum, choked flow, must then be subtracted to give the correct C_{D_x}.

where $m^* (V_4 - V_\infty) + (p_4 - p_\infty)A_4 = \int p\, dA - \int p_R A_R + \int p_d Ad - \int p_L A_L - \int p_f' dA$

But $m^* (V_4 - V_1) + p_4 A_4 - p_1 A_1 = \int p_d Ad - \int p_L A_L$

$m^* (V_4 - V_2) + p_4 A_4 - p_2 A_2 = \int p_d Ad$

Thus then is no correction, except that due to $\int p_d Ad$ being different than the A/C itself, i.e. calculate $\int p_d Ad$ for the model and subtract, then add on $\int p_d Ad$ for the A/C.



$$Dint = -\int p\, dA + \int p_R A_R - \int p_d Ad + \int p_1 A_1 + \int p_f' dA$$

$$Dext = +\int p\, dA + \Sigma D_x$$

∴ to find Dext when Dint + Dext is measured

$$Dext = (Dext + Dint) - Dint$$
$$= (Dext + Dint) + m^* (V_4 - V_\infty) + (p_4 - p_\infty)A_4$$

+ any correction due to model instrumentation (boom, pitots, rakes), differences in geometry (fin, camber, inlet & exit), trim drag, induced drag.

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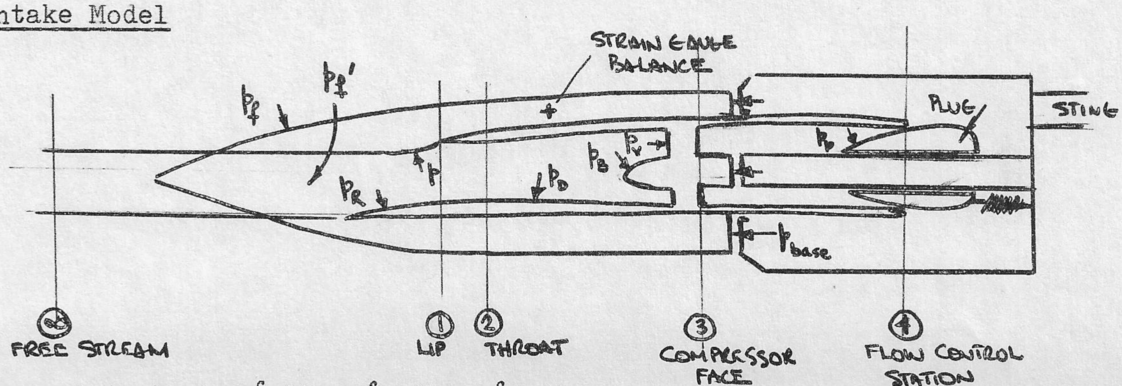
(c) <u>Intake Model</u>



We want to know $\Delta \int p dA = \int p_1 dA - \int p_2 dA$

where subscript 1 refers to choke throat m^*
2 " some reduced mass flow m

$$X_{SG} = \int p_I dA - \int p_d Ad + \int p_R A_R + \int p_B dA + \int p_V dA - \int p_{base} dA + \int p_f dA + \int p_f' dA$$

We measure: $m(V_3 - V_\infty) + (p_3 - p_\infty)A_3$ (We could measure the momentum change to stn. (4) but this would include $\int p_p dA$ which is carried by sting).

which equals: $-\int p dA + \int p_1 dA + \int p_B dA + \int p_V dA - \int p_d Ad + \int p_R A_R + \int p_f' dA$

$$\therefore \int p dA = \int p_1 dA + \int p_B dA + \int p_V dA - \int p_d Ad + \int p_R A_R + \int p_f' dA - m(V_3 - V_\infty) - (p_3 - p_\infty) A_3$$

$$= X_{SG} + \int p_{base} \, dA - \int p_f dA - m(V_3 - V_\infty) - (p_3 - p_\infty) A_3$$

$$\therefore \text{Spillage} = \int p_1 dA - \int p_2 dA$$

$$= \left[X_{SG_1} - X_{SG_2} \right] + \left[\int p_{base_1} dA - \int p_{base_2} dA \right] - m^*(V_3 - V_\infty) + m(V_3 - V_\infty) - (p_{3_1} - p_{3_2}) A_3$$

where $\int p_f dA$ is fuselage wing drag <u>un</u>affected by change in engine mass flow .

$\int p_f' dA$ " " affected " " "