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AIRLOADS - AVRO ARROW

The structural flight envelope is shown in figure 1. As can be seen, the lowest altitude at which the full range at supersonic Mach is met is 30,000 ft. At this altitude airload calculations were carried out for the full speed range ie.

M=2.0 , 1.6 , 1.2, 1.0 and .56

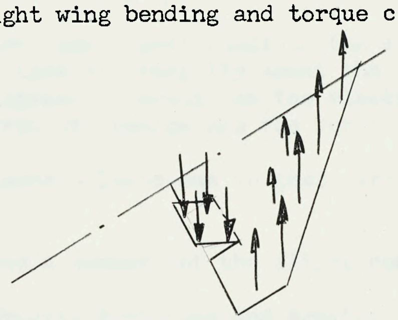
The symmetric flight manoeuvre envelope at this altitude is shown in figure 2. The fall off in design limit load factor with increasing temperature should be noted.

The subsonic and low supersonic loads were investigated under high air density conditions at sea level. The following speeds were considered:

M=1.09 , .80 , .60 , .453 and .272
Figure 3 shows the sea level manoeuvre envelope. It should be noted that all load factors quoted are based on an A.U.W. of 47,000 lb. and for higher weights " n W " is kept constant. That is the design load factor is lower.

At each altitude and Mach Number mentioned above airloads were calculated for some of the following manoeuvres:

(a) Steady pull-up to limit load factor -
Because of the special aircraft configuration this produced the largest in flight wing bending and torque conditions.



In achieving limit load factor the wing had to produce additional lift to overcome the large elevator balancing down load. In this manoeuvre pitching acceleration is zero.

(b) Steady push-down to negative limit load factor-
Not critical for design because at low load level.

(c) Checked pull-up to limit load factor-
Similar to the balanced pull-up except that the elevator load is dropped to the neutral level and the unbalanced aircraft allowed to pitch nose-down. Without the down elevator load the wing lift is somewhat smaller to reach the same normal load factor. This case produces large up bending in the rear of the wing structure. It also designs the fuselage rearend because the inertia effects due to pitching acceleration and normal acceleration add.

(d) Checked push-down - Not critical for design.

(e) Rolling pull-out - Consists of rolling the aircraft at the maximum allowable rate while pulling 2/3 of the maximum load factor. On old fashioned airplanes this manoeuvre was used to design the vertical tail. However, on aircraft with low aspect ratio wings and long high density fuselages the fin side load so induced is prohibitively large.

The problem is met by use of automatic stability augmentation which precludes the possibility of attaining large fin loads.

The rolling pull-out is still of importance due to wing loads. The combination of loads due to 2/3 the normal load factor with those due to aileron and damping in roll produce design torque shear and B.M. for the outer wing.

(f) Rudder damper failure - The use of automatic artificial stability solves one problem but brings another in its wake. Depending on electronic and other devices rather than on unfailing air flow about air foils raises the possibility of a failure in the system. The worst possible thing which could occur would be a runaway of the control surface to either hinge moment max. or to the control stop. To guard against this possibility, safety devices are built into the system which switch the runaway system off and a duplicated emergency system on, which then brings the aircraft back under control. However, signifigant loads may occur during the manoeuvre.

The most signifigant loads on the aircraft occur during a rudder runaway due to lack of stability about the yaw axis. These cases have been investigated in detail on the electronic analogue (flight simulator) in 5 degrees of freedom and the worst loads used for fin strength checking.

(g) Gust loads - loads due to gusts are much smaller than manoeuvre loads.

Table 1 gives a summary of the flight conditions for which loads were issued.

Centre of Gravity Positions and Arbitrary Increments:

For all cases but Yaw Damper Failure, loads have been found for three c.g. positions
$.28$, $.30$, and $.32\ \bar{c}$

Following the practice of AP 970 , arbitrary increments of pitching moment and aerodynamic centre shift have been included in each case.

Max. nose up combination $\Delta C_{m_o} = .0075$ $\Delta a.c. = -.025\bar{c}$

Max. nose down combination $\Delta C_{m_o} = -.0075$ $\Delta a.c. = .025\bar{c}$

A consequence of taking the two arbitrary increments acting together has been to make elevator loads to balance the aircraft larger than those available from the jacks. However, to retain the conservatism (and prepare for unforeseen problems) elevator loads were shifted forward arbitrarily to reduce the hinge moments to ~~possible~~ available limits while still balancing the aircraft.

Methods of Airloads Analysis:

From the outset an attempt has been made to keep the determination of overall aircraft airloads on an analytic basis. The powerful linearized supersonic flow techniques of J.C. Evvard have been extended by F. Woodward and Prof. B. Etkin to the solution of loads over wings of arbitrary planform and camber. These methods have been used to compute the rigid wing and control surface loads for the three supersonic Mach Numbers.

An interesting extension to account for the effects of elasticity has been made by F. Woodward. In this technique the wing is assumed to be composed of a number at control surface shaped panels free to take up any incidence to the wind. The deflection of any one panel causes a load on itself and loads on other panels in its Mach cone. These loads form a column of an aerodynamic matrix. Written in algebraic form the loads on the wing for a certain incidence distribution would be

$$L_1 = a_{11}\alpha_1 + a_{12}\alpha_2 + a_{13}\alpha_3 ----$$

$$L_2 = a_{21}\alpha_1 + a_{22}\alpha_2 + a_{23}a_3 -----$$

or in matrix notation

$$|L| = [A]|\alpha|$$

where $|\alpha_0|$ is the rigid camber and incidence distribution of the wing, panel rigid loads are

$$|L| = [A]|\alpha_1|$$

to find elastic loads use is made of the structural slope matrix

$$|\Delta\alpha| = [S]|L|$$

Elastic loads are

$$|L| = [A]\Big||\alpha_0| + |\Delta\alpha|\Big| = [A]|\alpha_0| + [A][S]|L|$$

$$|L| - [A][S]|L| = [A]|\alpha_0|$$

$$|L| = [I - [A][S]]^{-1}[A]|\alpha_0|$$

The implementation of this simple method of course requires the use of computing facilities capable of rapidly inverting matrices of the order of 50 x 50.

From unit elastic loads found for angle of attack, control surface deflection, roll rate etc. complete elastic cases have been synthesized.

The elastic loads study showed the effects of elasticity to be insignifigant up to all but the highest speed. At M=2.0 at 30,000 ft. the symmetric pull-up loads were of the order of 10% more severe than the rigid loads. Investigation of the cause of this revealed that if it had not been for the arbitrarily high elevator hinge moment the elastic load would have been signifigantly below the rigid one. Since it was deemed not reasonable to double penalize the struture for the high elevator load, rigid loads were used throughout for symmetric manoeuvres.

For the rolling pull-out case at M=2.0 Alt. = 30,000 ft. full effects of elasticity have been considered in loads evaluation and used for wing stressing.

Subsonic airloads have been calculated using the methods of Lawrence, Multhopp and Faulkner. Comparison with wind tunnel tests on the Arrow for aerodynamic centre position and lift curve slope showed the method of Faulkner to be most representative and it was used for all design cases. Elevator load distributions were obtained from work done by N.A.C.A. on an almost identical wing. The distributions were altered slightly to agree with pitching moment, hinge moment and lift due to elevator deflection obtained from wind tunnel tests on the Arrow.

Elastic effects were neglegible for the subsonic cases.

STRUCTURAL FLIGHT ENVELOPE — AVRO ARROW
A.U.W. = 47,000 LB.

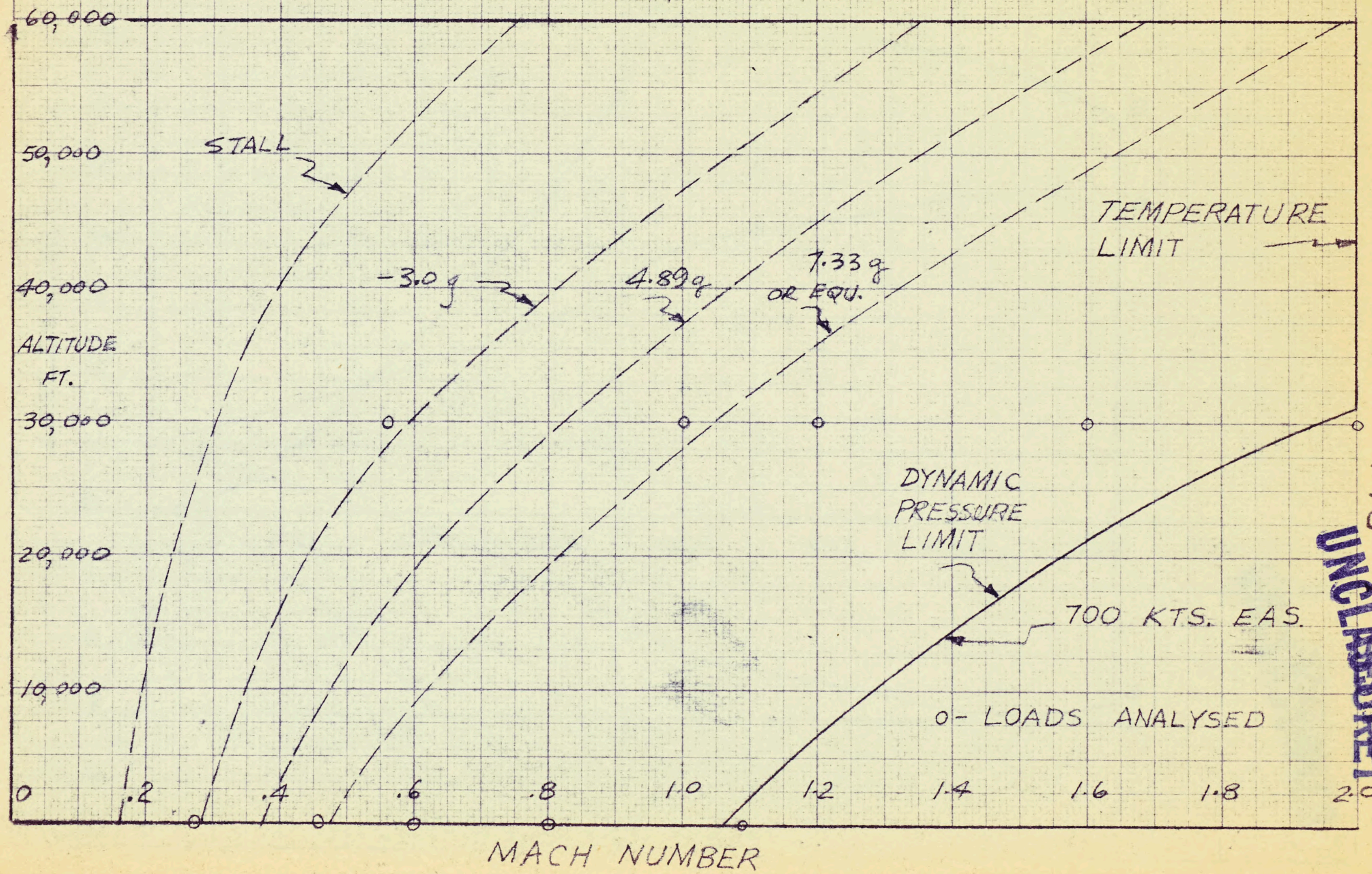


FIG 1

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SHEET 4 Y/LOADS/120 APPENDIX 1
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FIG 2

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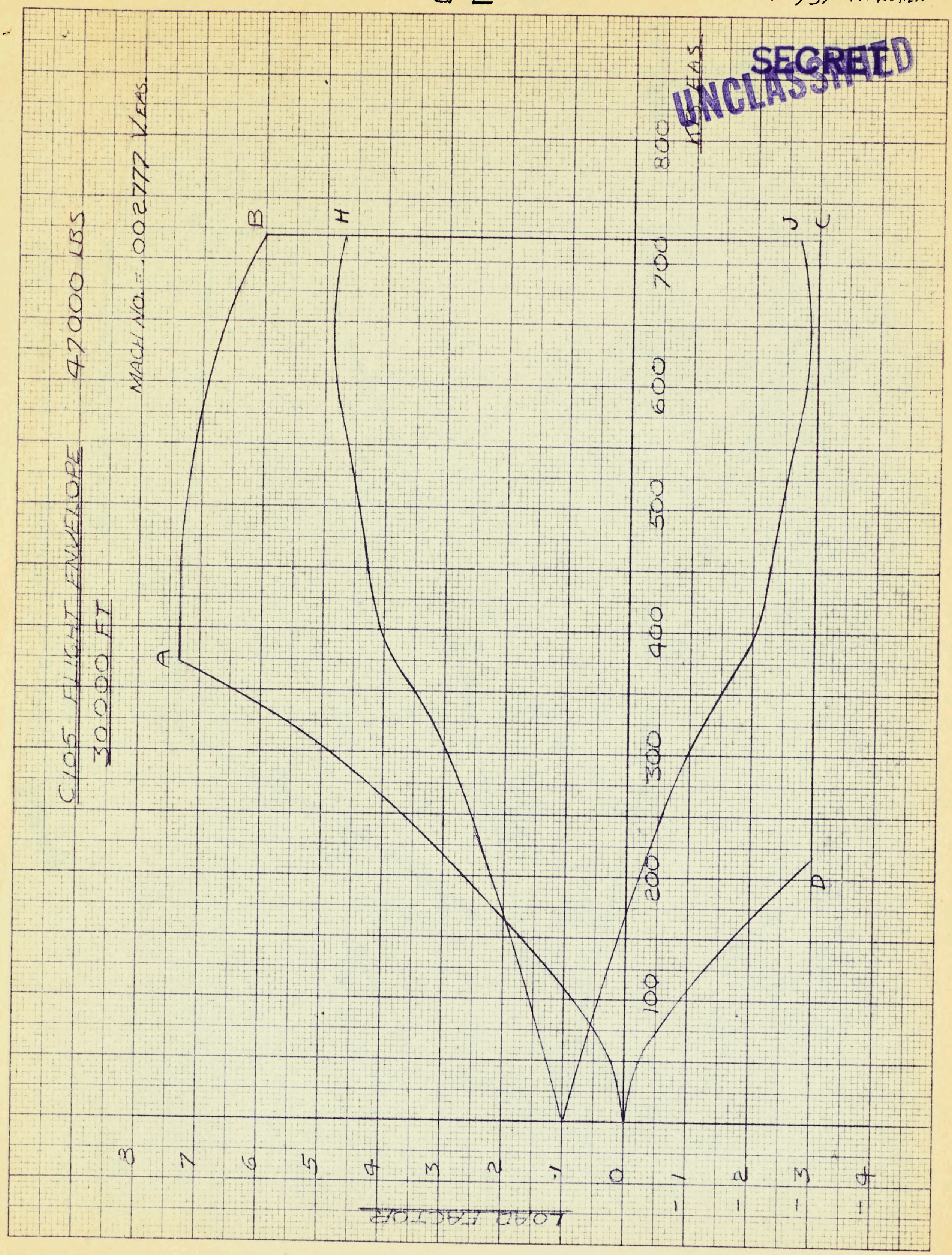

SHEET I P/LOADS/120 APPENDIX I
JAN/57 R. LOKEN

FIG 3

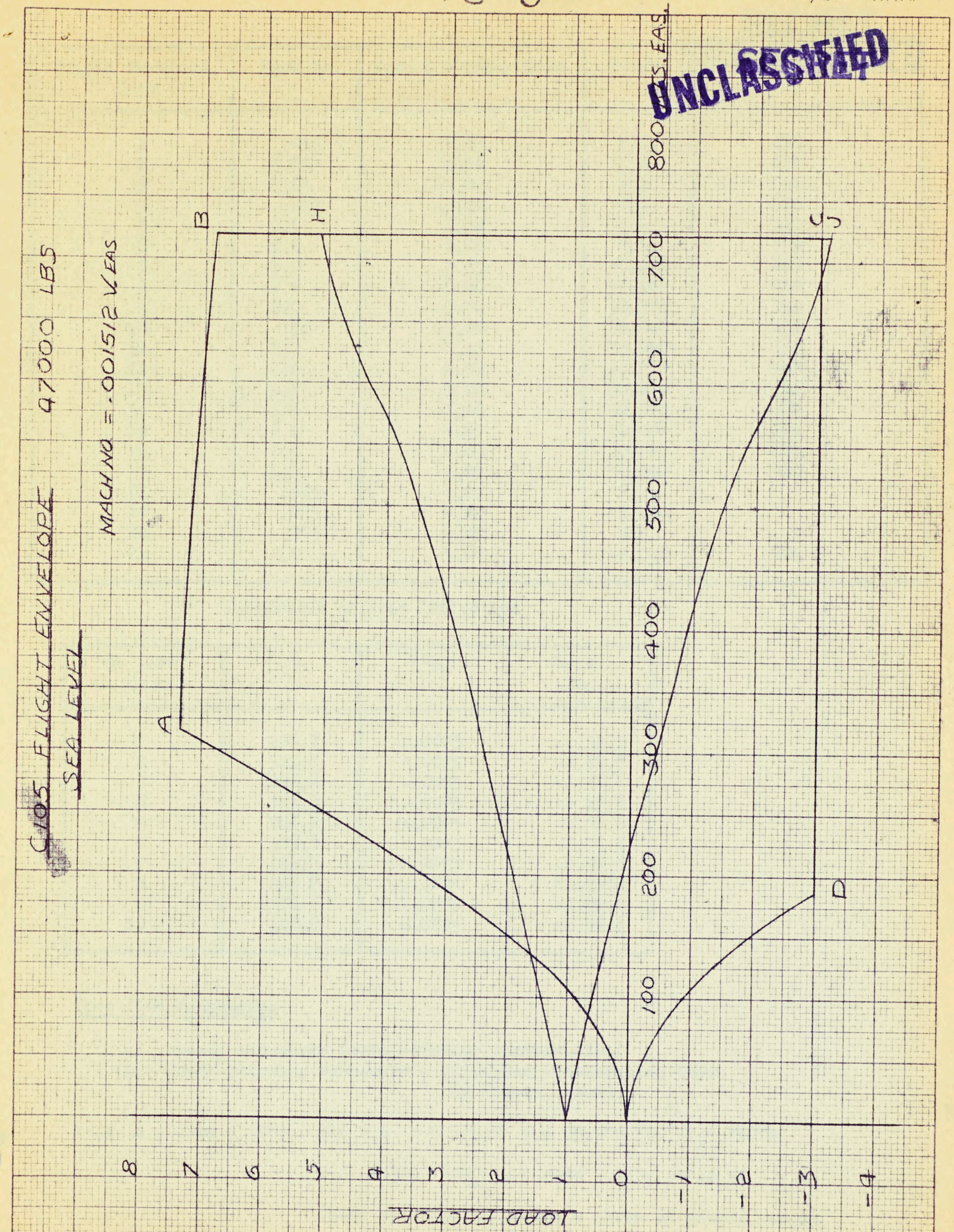

TABLE 1

March 18th, 1957.

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Flight Envelope Case	Mach No.	Normal Acceleration	Height	Surface Temp. (°R)
1	.453	7.33	S.L.	538
2	1.09	6.80	S.L.	631
3	1.09	-3.00	S.L.	631
4	.272	-3.00	S.L.	526
5	0.6	7.28	S.L.	553
6	0.8	7.12	S.L.	579
7	1.0	7.33	30,000	487
8	2.0	6.00	30,000	710
9	2.0	-3.00	30,000	710
10	.56	-3.00	30,000	435
11	1.2	7.33	30,000	520
12	1.6	7.02	30,000	603
13	1.2	-3.00	30,000	520
14	1.6	-3.00	30,000	603
15	2.0	4.89	30,000	710
16	1.6	4.89	30,000	603
17	1.2	4.89	30,000	520
18	.825	4.89	S.L.	
19	.825	4.89	30,000	
20	.375	4.89	S.L.	
21 *	2.0	6.40	45,000	710

* Symetrical Aileron Deflection

Sub Case	C.G.	ΔC_{M_0}	Δa.c.
1	27%.c̄	+.0075	-2.5%.c̄
2	31%.c̄	+.0075	-2.5%.c̄
3	27%.c̄	-.0075	+2.5%.c̄
4	31%.c̄	-.0075	+2.5%.c̄
5	28%.c̄	+.0075	-2.5%.c̄
6	30%.c̄	+.0075	-2.5%.c̄
7	32%.c̄	+.0075	-2.5%.c̄
8	28%.c̄	-.0075	+2.5%.c̄
9	30%.c̄	-.0075	+2.5%.c̄
10	32%.c̄	-.0075	+2.5%.c̄

Specified Manoeuvres

(a) Steady manoeuvre (zero pitching acceleration)
(b) Check pitching manoeuvre - (pitching acceleration shown on sheets issued).

Example - Case 9.2(b): M = 2.0 n = -3.00 h = 30,000 ft. T = 710°R

C.G. = 31% ΔC_{M_0} = +.0075 Δa.c. = -2.5%

Check pitching manoeuvre $\ddot{\theta}$ shown on sheets issued.