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A.R.C. Technical Report

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A Family of Camber Lines for Subsonic Applications (A Modified Form of the N.A.C.A. Family with Uniform Loading over the Forward Part and Linear Loading over the Rear Part)

by

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A Family of Camber Lines for Subsonic Applications

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N. B. Squire, M.A .

17th April, 1958

1. Introduction

The N.A.C.A. camber lines[1] designed to have loading uniform from the leading edge $(x = 0)$ to a station $x = X$, and with a linear variation from this station to the trailing edge $(x = 1)$, at which the loading is zero, are widely used. They have however appreciable curvature over the rear part, which is of some disadvantage in practical applications. This can be removed by modifying the initial design so that the design loading is taken to be a small positive quantity at the trailing edge. The family of camber lines described in the present note are derived in this way from the N.A.C.A. camber lines and are straight over the trailing edge region.

The present family are convenient for use with the RAE 100-104 symmetrical aerofoil sections[2] and the camber lines which correspond to these sections have been given the additional designations C100-C104 to bring out the connection.

2. Derivation

The camber lines have a first order load and pressure distribution which consists of a uniform part from the leading edge $x = 0$ to a point $x = X$, where x is the distance along the chord line, and a linear variation from $x = X$ to the trailing edge $x = 1$, but the theoretical loading does not vanish at the trailing edge. They have been calculated mainly from the data of Table 2 of Ref. 3, with some assistance from the data of Ref. 1.

The theoretical trailing-edge loading is chosen to give a close approximation to a straight line aerofoil shape near the trailing edge and this slightly curved line is replaced by an actual straight line (see Appendix and Ref. 2).

Let C_{p_1}, C_{p_2} be the pressure coefficients for the upper and lower surfaces at a point on the camber line and u_1, u_2 be the corresponding velocity increments. Then on linearised theory, for a stream velocity U,

$$C_{p_1} = - C_{p_2}, \qquad u_1 = - u_2$$

$$C_{p_1} = - 2u_1/U, \; C_{p_2} = - 2u_2/U$$

$$\text{Loading} = C_{p_2} - C_{p_1} = - 2C_{p_1} = 4u_1/U.$$

3./

3. Co-ordinates

The co-ordinates of the camber lines are given in Table II and some of the shapes are shown in Fig. 1.

4. Pressure and Velocity Distribution

The theoretical pressure coefficient and velocity increment on the upper surface for a nominal ideal lift coefficient $C_{Li} = 1.0$ are given in Table I and Fig. 2.

Acknowledgement

The calculations were made by Miss M. Lawson.

References

No.	Author(s)	Title, etc.
1	I. H. Abbott and A. E. von Doenhoff	Theory of wing sections (including a summary of airfoil data). McGraw-Hill Book Co. 1st edition, 1949.
2	R. C. Pankhurst and H. B. Squire	Calculated pressure distributions for the RAE 100-104 aerofoil sections. (With addendum). A.R.C. C.P. No. 80. March, 1950.
3	S. Goldstein	Approximate two-dimensional aerofoil theory Part IV. The design of centrelines. A.R.C. C.P. No. 71. March, 1945.

APPENDIX

Method of Calculation

Let the N.A.C.A. camber line with a uniform load distribution from $x = 0$ to $x = X$ and a linear variation from $x = X$ to $x = 1$ with zero loading at the trailing edge and designed for $C_{Li} = 1.0$ have the equation

$$y = f(x, X).$$

The camber line with uniform loading along the whole chord with $C_{Li} = 1.0$ has the equation

$$y = f(x, 1).$$

A camber line with uniform loading from $x = 0$ to $x = X$ and a linear loading from $x = X$ to $x = 1$ with non-zero loading at $x = 1$ and designed for $C_{Li} = 1.0$ has the equation

$$y = Af(x, X) + (1 - A)f(x, 1)$$

where A is a constant. We now choose A by trial and error to be such that the aerofoil has a point of inflection towards the trailing edge such that the tangent at this point passes through the trailing edge. Behind this point of inflection the aerofoil is replaced by its tangent.

This geometrical modification changes the actual load distribution over the camber line near the trailing edge in such a way that the load actually falls to zero at the trailing edge. This effect has been roughly calculated by thin aerofoil theory and it was found that the actual design lift coefficient was 0.99 instead of 1.0.

The values obtained for A and the positions of the points of inflection, behind which the aerofoils are replaced by straight lines, are as follows:-

X	0	0.1	0.2	0.3	0.4	0.5	0.6
A	0.8595	0.863	0.860	0.855	0.850	0.848	0.859
Point of Inflection	0.8684	0.92	0.92	0.8699	0.8772	0.8721	0.8535

Table I/

Table I

$C_{Li} = 1.0$

X	0	0.1	0.2	0.3	0.4	0.5	0.6
Designation	C100	-	-	C101	C102	C103	C104
Upper surface pressure coefficient from $x = 0$ to $x = X$	0.930	0.854	0.788	0.730	0.682	0.641	0.607
Nominal upper surface pressure coefficient at $x = 1$	0.070	0.068	0.070	0.072	0.075	0.076	0.071
Ideal angle of incidence (deg.)	3.92	3.83	3.59	3.28	2.94	2.58	2.22
Pitching moment $C_{m_{c/4}}$	-0.105	-0.108	-0.116	-0.125	-0.137	-0.153	-0.172

Table II/

Table II

Co-ordinates of Camber Lines

Values of 100y C_{Li} = 1.0

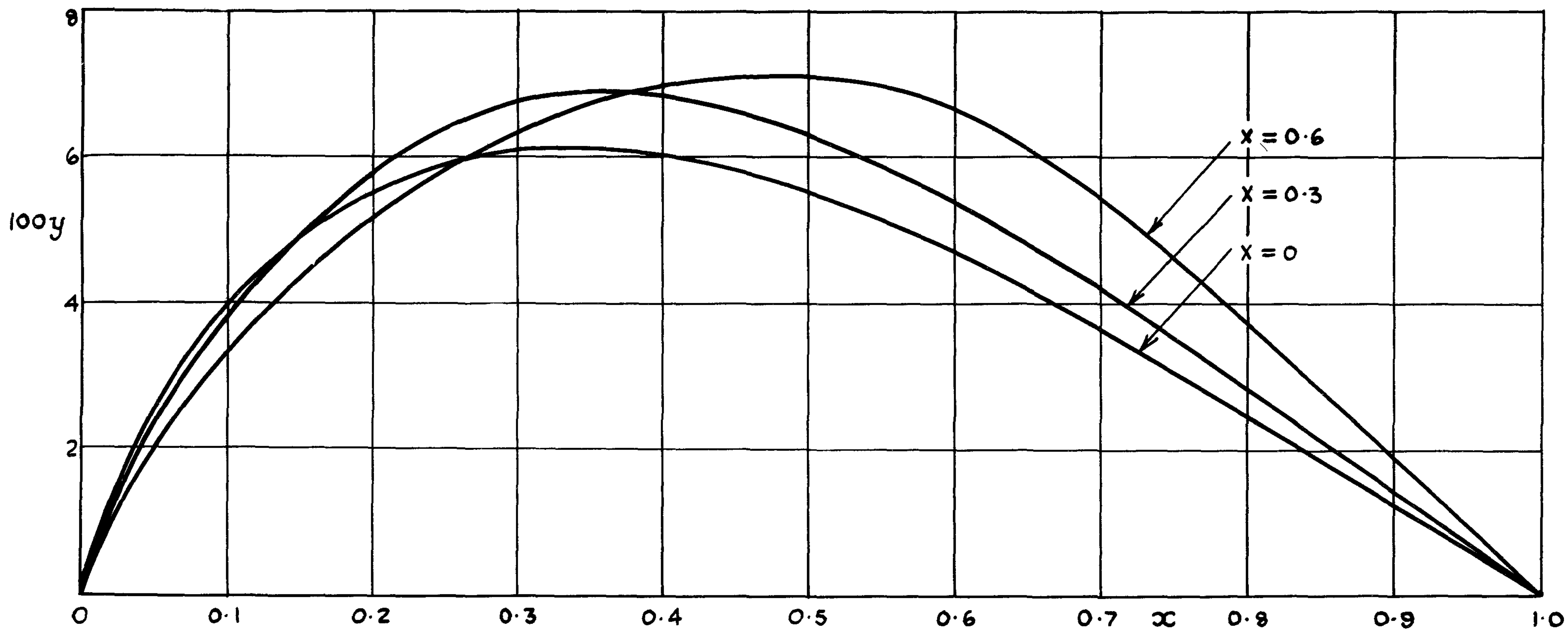
X	0	0.1	0.2	0.3	0.4	0.5	0.6
Desig-nation	C100	-	-	C101	C102	C103	C104
x							
0.001	0.1101			0.0926	0.0872	0.0824	0.0783
0.002	0.1996			0.1690	0.1593	0.1508	0.1432
0.003	0.2813			0.2393	0.2258	0.2136	0.2030
0.004	0.3578			0.3057	0.2886	0.2731	0.2627
0.005	0.4307	0.414	0.391	0.3691	0.3486	0.3300	0.3136
0.006	0.5003			0.4301	0.4063	0.3848	0.3658
0.007	0.5673			0.4891	0.4623	0.4378	0.4163
0.0075	0.6002	0.580	0.549	0.5181	0.4896	0.4638	0.4410
0.008	0.6324			0.5466	0.5167	0.4894	0.4654
0.009	0.6955			0.6024	0.5697	0.5397	0.5132
0.010	0.7568			0.6570	0.6215	0.5888	0.5600
0.012	0.8749			0.7621	0.7218	0.6841	0.6507
0.0125	0.9036	0.878	0.834	0.7886	0.7463	0.7074	0.6728
0.014	0.9879			0.8646	0.8184	0.7759	0.7382
0.016	1.0964			0.9630	0.9119	0.8647	0.8227
0.018	1.2009			1.0584	1.0026	0.9509	0.9048
0.020	1.3019			1.1511	1.0908	1.0348	0.9847
0.025	1.5411	1.515	1.446	1.3728	1.3018	1.2356	1.1762
0.030	1.7641			1.5823	1.5016	1.4258	1.3577
0.035	1.9735			1.7816	1.6919	1.6072	1.5308
0.040	2.1712			1.9720	1.8740	1.7809	1.6966
0.05	2.5363	2.541	2.442	2.3302	2.2171	2.1086	2.0097
0.06	2.8676			2.6630	2.5367	2.4144	2.3022
0.07	3.1701			2.9741	2.8363	2.7015	2.5771
0.075	3.3119	3.355	3.258	3.1225	2.9795	2.8389	2.7087
0.08	3.4478			3.2664	3.1186	2.9725	2.8368
0.09	3.7035			3.5417	3.3854	3.2292	3.0831
0.10	3.9395	4.024	3.947	3.8019	3.6383	3.4729	3.3172
0.12	4.3602			4.2811	4.1067	3.9258	3.7532
0.14	4.7175			4.7115	4.5308	4.3380	4.1512
0.15	4.8765	5.001	5.044	4.9099	4.7278	4.5302	4.3373
0.16	5.0231			5.0978	4.9154	4.7140	4.5155
0.18	5.2821			5.4438	5.2643	5.0574	4.8496
0.20	5.4992	5.641	5.818	5.7519	5.5799	5.3706	5.1558
0.22	5.6785			6.0236	5.8643	5.6558	5.4365
0.24	5.8234			6.2599	6.1187	5.9143	5.6924
0.25	5.8838	6.034	6.279	6.3648	6.2352	6.0341	5.8117
0.26	5.9367			6.4610	6.3444	6.1476	5.9253
0.28	6.0208			6.6259	6.5420	6.3563	6.1362
0.30	6.0781	6.231	6.508	6.7515	6.7119	6.5412	6.3257
0.32	6.1103			6.8307	6.8541	6.7027	6.4944
0.34	6.1193			6.8728	6.9681	6.8412	6.6428
0.35	6.1154	6.266	6.560	6.8620	7.0143	6.9019	6.7094
0.36	6.1064			6.8838	7.0531	6.9568	6.7711
0.38	6.0733			6.8668	7.1073	7.0493	6.8796
0.40	6.0212	6.167	6.466	6.8247	7.1265	7.1184	6.9683
0.42	5.9514			6.7594	7.1024	7.1636	7.0372

(contd.)/

Table II (contd.)

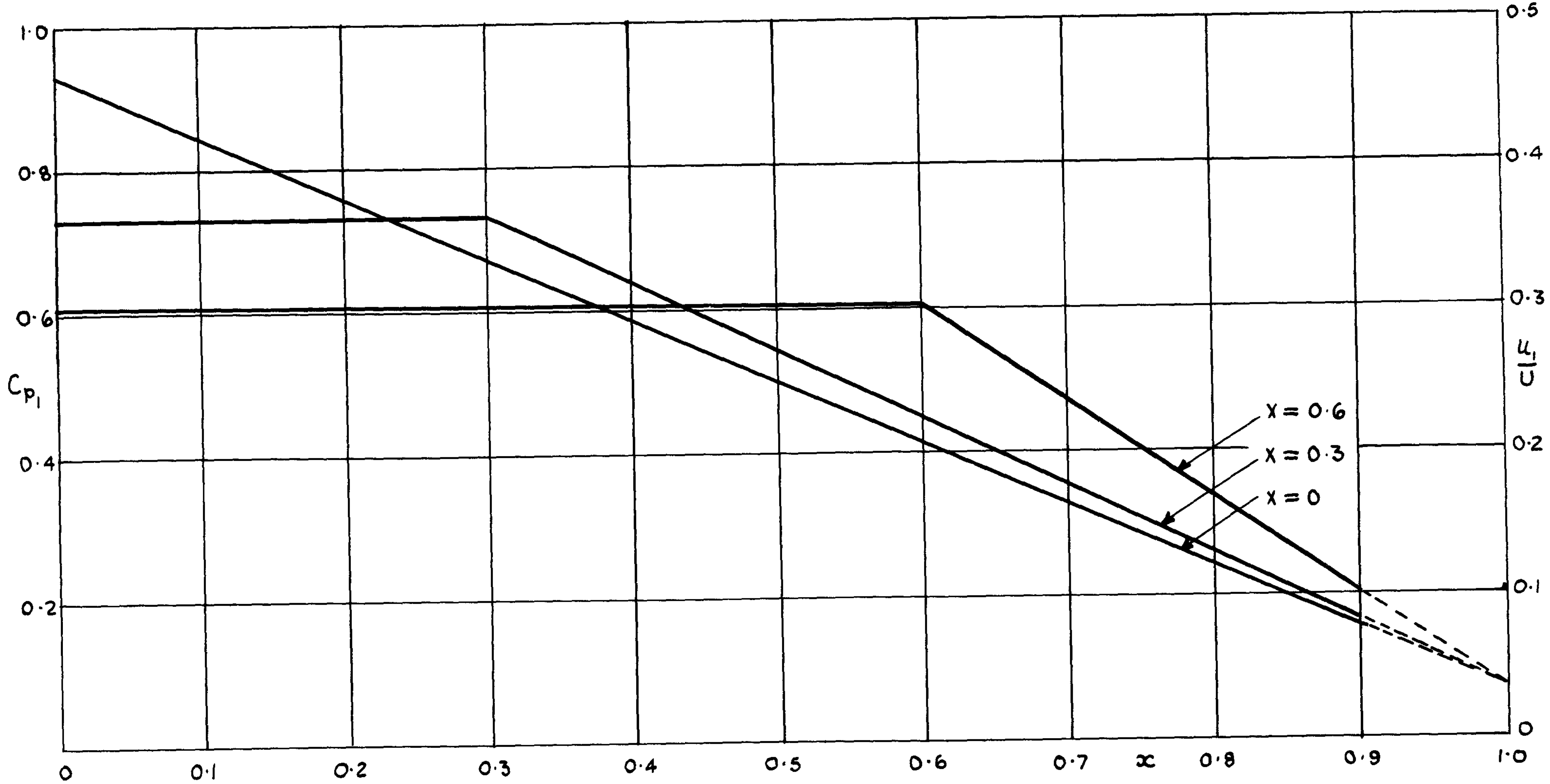
X	0	0.1	0.2	0.3	0.4	0.5	0.6
Designation	C100	-	-	C101	C102	C103	C104
x							
0.44	5.8649			6.6727	7.0448	7.1839	7.0861
0.45	5.8158	5.953	6.248	6.6219	7.0052	7.1844	7.1030
0.46	5.7628			6.5664	6.9592	7.1780	7.1148
0.48	5.6462			6.4418	6.8491	7.1433	7.1226
0.50	5.5159	5.644	5.927	6.3003	6.7170	7.0750	7.1090
0.52	5.3728			6.1431	6.5648	6.9637	7.0730
0.54	5.2180			5.9713	6.3944	6.8196	7.0132
0.55	5.1363	5.253	5.520	5.8802	6.3027	6.7375	6.9738
0.56	5.0520			5.7861	6.2071	6.6493	6.9278
0.58	4.8758			5.5882	6.0045	6.4561	6.8139
0.60	4.6901	4.795	5.041	5.3789	5.7878	6.2429	6.6651
0.62	4.4956			5.1589	5.5583	6.0116	6.4704
0.64	4.2930			4.9293	5.3169	5.7640	6.2422
0.65	4.1890	4.281	4.501	4.8110	5.1922	5.6347	6.1179
0.66	4.0831			4.6906	5.0648	5.5020	5.9877
0.68	3.8666			4.4440	4.8030	5.2270	5.7111
0.70	3.6441	3.722	3.915	4.1901	4.5325	4.9402	5.4156
0.72	3.4163			3.9297	4.2542	4.6431	5.1036
0.74	3.1838			3.6636	3.9689	4.3358	4.7774
0.75	3.0660	3.130	3.295	3.5288	3.8240	4.1806	4.6095
0.76	2.9473			3.3927	3.6777	4.0226	4.4389
0.78	2.7075			3.1176	3.3814	3.7017	4.0900
0.80	2.4650	2.514	2.648	2.8391	3.0809	3.3749	3.7324
0.82	2.2204			2.5580	2.7770	3.0436	3.3677
0.84	1.9745			2.2750	2.4707	2.7087	2.9975
0.85	1.8512	1.887	1.988	2.1330	2.3169	2.5403	2.8108
0.86	1.7277			1.9908	2.1627	2.3715	2.6233
0.88	1.4813			1.7062	1.8537	2.0332	2.2490
0.90	1.2344	1.256	1.324	1.4218	1.5448	1.6944	1.8742
0.92	0.9875			1.1375	1.2358	1.3555	1.4993
0.925	0.9258			1.0664	1.1586	1.2708	1.4056
0.94	0.7406			0.8531	0.9268	1.0166	1.1245
0.95	0.6172	0.630	0.664	0.7109	0.7724	0.8472	0.9371
0.96	0.4938			0.5687	0.6179	0.6777	0.7497
0.975	0.3086			0.3557	0.3862	0.4236	0.4685
0.98	0.2469			0.2844	0.3090	0.3389	0.3748
0.9875	0.1543			0.1777	0.1931	0.2118	0.2343
1.0	0	0	0	0	0	0	0

FIG. 1.



Shape of camber lines. $C_{L_i} = 1{\cdot}0$.

FIG. 2



Pressure coefficient and velocity increment on upper surface. $C_{L_i} = 1·0$.

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