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Generalized Leading-Edge Modifications for Increased $c_{l\max}$ of NACA 6-Series Airfoils

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SYMBOLS

c	chord length, m
c_d	section drag coefficient
c_l	section lift coefficient
$c_{l_{max}}$	maximum value of c_l
c_m	section pitching-moment coefficient, about 1/4-chord
c_p	pressure coefficient
$c_p{}^*$	pressure coefficient for $M = 1.0$
$c_{p_{min}}$	minimum value of c_p
ESF	exponential shape function, $ESF = MU\left(\frac{x}{c}\right)^{\frac{1}{2}}\left(1 - \frac{x}{c}\right)e^{-8\frac{x}{c}}$
K	ratio of $\mid c_{p_{min}} \mid$ to the chordwise location of $c_{p_{min}}$
M	freestream Mach number
MU	multiplier of the ESF, $ESF = MU\left(\frac{x}{c}\right)^{\frac{1}{2}}\left(1 - \frac{x}{c}\right)e^{-8\frac{x}{c}}$
Re	Reynolds number, based on chord and freestream conditions
t	maximum sectional thickness, m
x/c	nondimensional abscissa with respect to chord
y/c	nondimensional ordinate with respect to chord
α	angle of attack, deg

GENERALIZED LEADING-EDGE MODIFICATIONS FOR INCREASED $c_{l_{max}}$ OF NACA 6-SERIES AIRFOILS

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SUMMARY

A simple method has been developed for computing modifications to the forward camber and thickness of any NACA 63-, 64-, or 65-series airfoil with the objective of achieving an increase in the maximum lift coefficient. The method can be used for any design lift coefficient between 0.0 and 1.0, and for any maximum thickness/chord ratio between 0.06 and 0.18. A simple mathematical function defines an incremental thickness which is applied to the upper surface of a cambered airfoil, or is applied to both surfaces of an uncambered airfoil. The method is sufficiently compact to permit solutions to be obtained on a hand-held scientific calculator containing logarithmic functions and a power function. In applying the method, the incremental thickness is added to the thickness of the forward portion of the airfoil, thus increasing the bluntness of the leading edge and increasing the forward camber of cambered airfoils. This does not introduce any discontinuities in curvature, since the function and its derivatives are continuous over the entire chord length. An increase in the forward thickness of the NACA 6-series airfoils will reduce the adverse pressure gradient near the leading edge on the upper surface at high angles of attack, thereby postponing separation until a higher angle of attack is reached, which results in an increased $c_{l_{max}}$. An increased $c_{l_{max}}$ offers improved stalling characteristics and slower, shorter landing runs. The lower drag resulting from delayed separation at higher lift coefficients will result in an improved rate of climb.

INTRODUCTION

The maximum lift coefficient of NACA 6-series airfoils (ref. 1) can be improved by adding thickness to the forward region of the upper surface. Wind-tunnel tests have demonstrated substantial improvements in $c_{l_{max}}$ for two NACA 6-series airfoils of 12 and 15% thickness with low camber (refs. 2 and 3), and for a thin, highly-cambered 6-series section typical of compressor blades (ref. 4). In the earlier work described in references 2 and 3, the sections with low camber were modified by increasing the upper-surface thickness between the leading edge and 35% chord, thereby increasing the leading-edge bluntness and forward camber. These modifications resulted in a discontinuity in the surface curvature at the 35%-chord point which can result in a wavy pressure distribution near this point, and may cause separation to occur earlier than with smooth surface curvatures. On the other hand, the thin, highly-cambered section of reference 4 was modified by adding a smooth mathematical function, defined over the entire chord length, to the upper surface

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of the original airfoil. This latter modification resulted in a smoother contour than the method of references 2 and 3. Both modifications yielded increases in $c_{l_{max}}$ of up to 30%.

The method of this report is an extension and generalization of the smooth type of modification described in reference 4. It permits the designer to modify any 6-series airfoil in the range of 6 to 18% maximum thickness/chord ratio and with design lift coefficients between 0.0 and 1.0 by using the simple airfoil-modification graphs of this report and a hand-held calculator. The airfoil-modification graphs were obtained by examining the computed effect of incremental leading-edge thickness on $c_{l_{max}}$, $c_{p_{min}}$, c_d, and the chordwise location of $c_{p_{min}}$. Limited increases in leading-edge bluntness will reduce $\mid c_{p_{min}} \mid$ and increase its distance from the leading edge at high angles of attack, thereby delaying separation which would increase $c_{l_{max}}$. However the cruise speed may be reduced because of a reduction in the extent of laminar flow. The objective of these modifications is to increase $c_{l_{max}}$ as much as possible while minimizing any increase in cruise drag. The similarities between the modifications described herein and those of the previous work (refs. 2-4), although not experimentally verified, should give the designer confidence that the proposed modifications can be used to obtain substantial improvement in $c_{l_{max}}$.

The airfoil designer may skip to the Design Instruction section if only application of the method is desired.

METHOD

An exponential shape function (ESF) of the form $ESF = MU \cdot \frac{y}{c}$, where $\frac{y}{c} = \left(\frac{x}{c}\right)^n \left(1 - \frac{x}{c}\right) e^{-m\frac{x}{c}}$ and MU is a numerical multiplier has been used extensively to alter airfoils with round leading edges (ref. 5). A parametric study has shown that values of m larger than approximately 8.0 produce peaky pressure distributions near the leading edge at low angles of attack. Such distributions are undesirable because of greater adverse pressure gradients following the peak. Values of m less than 8.0 caused the maximum t/c ratio to become too large before the leading edge became suitably blunt. Therefore m was chosen to be 8.0. The value of n was chosen to be 0.5 because this value of n causes the maximum curvature to occur aft of the leading edge, which has been shown in reference 6 to increase the maximum lift coefficient. The resulting ESF is shown in figure 1 for the multiplier, MU, having values of 0.05, 0.1, and 0.15. The range $0.05 \leq MU \leq 0.15$ gives acceptable results for the leading-edge-thickness modifications.

Two design goals were used to establish recommended values for MU.

1. Minimize the cruise drag increase.

2. Reduce the leading-edge suction peak and shift the leading-edge suction peak toward the trailing edge.

The first goal seeks to limit the modifications to those which produce acceptable leading-edge bluntness and hence low cruise drag. The second goal is aimed at reducing the suction peak near the leading edge at high angles of attack.

Reduction of the leading-edge suction as in (2) has been shown experimentally to result in an improved maximum-lift coefficient (refs. 2-4). The second goal also involves minimizing the ratio of the absolute value of $c_{p_{min}}$ to the chordwise location of the $c_{p_{min}}$ (this ratio is designated as K). The effect of these two goals on MU will be discussed in detail in the following sections. The airfoil-modification graphs needed to use this method were developed by determining suitable multipliers for a subset of 6-series airfoils.

The values of MU which satisfy the two design goals are presented later in a set of airfoil-modification graphs which plot MU versus c_l for values of t/c between 0.06 and 0.18 for NACA 63-, 64-, and 65- series airfoils. These graphs were developed by applying the design goals to airfoils with design lift coefficients of 0.0, 0.25, 0.50, 0.75, and 1.0 and thickness/chord ratios of 0.06, 0.12, and 0.18. The multipliers for other lift coefficients and thickness/chord ratios within the stated range of values were interpolated from this subset. The ESF was added to the original NACA 6-series airfoil starting with a multiplier, MU, of 0.01, and increasing MU in increments of 0.01 until the modified airfoil had an excessively-blunt leading-edge region that produced an adverse pressure gradient and hence unacceptable cruise drag. This provided an upper bound on MU. A readily-available interactive graphics program, PROFILE (ref. 6), was used instead of a hand-held calculator to modify the ordinates of the airfoils analyzed. However, PROFILE is not required to use this method.

Minimize the Cruise Drag Increase

The shape/curvature of the leading-edge region of the modified airfoil was used in determining a suitable multiplier. The MU which optimizes $c_{l_{max}}$ may not be appropriate for cruise flight. The bluntness required for the largest $c_{l_{max}}$ may increase the adverse pressure gradient near the leading edge, resulting in greater drag at the low angles of attack typical of cruise flight. Thus the improvements in $c_{l_{max}}$ for the airfoils presented herein must be limited by cruise flight considerations. Therefore multipliers that gave excessive forward bluntness and hence drag were eliminated. For example the NACA 64(0.75)12 airfoil with multipliers of 0.04, 0.08, 0.12, and 0.15 is shown in figure 2. In this example, multipliers greater than 0.12 were considered unsuitable. Note that the multiplier of 0.15 produces excessive forward thickness.

Pressure distributions calculated by Program H (ref. 8), for the original NACA 64(0.75)12 airfoil and modified airfoil with 0.12 ESF added to the upper surface are shown in figures 3(a) and 3(b), respectively. The pressure distribution for the modified airfoil has a weak adverse-pressure gradient between x/c of 0.1 to 0.2 with a predicted drag larger than that for the unmodified airfoil because of the loss of some of the laminar flow over the upper surface. This increase in drag may not be evident in flight since the extent of laminar flow predicted for the original airfoil may not be achieved, because of poor manufacturing which may introduce surface waviness in the airfoil. Since the multiplier of 0.12 met this first goal as well as the second design goal discussed in the next sections, it was chosen for the NACA 64(0.75)12 airfoil.

Reduce the Leading Edge Suction Peak

It has been determined experimentally (refs. 2-4) that $c_{l_{max}}$ improves when the adverse pressure gradient following the suction peak near the leading edge is reduced at high angles of attack. This adverse pressure gradient was reduced by lowering the suction peak. The minimum c_p corresponding to each multiplier was calculated using Program H (ref. 8) at $M = 0.1$, and $\alpha = 10.0°$ typical of the touch-down condition. The peak pressures become smaller in magnitude with increasing multiplier of the ESF until a minimum is reached, as shown in figure 4. Similiar plots were produced for each 6-series section analyzed in this study. Multipliers greater than those which gave the minimum $\mid c_{p_{min}} \mid$ were eliminated in each case. However, in several cases, the leading edge became too blunt before the minimum value of $\mid c_{p_{min}} \mid$ was achieved. For example, multipliers in the range of 0.12 to 0.15 for the NACA 64(0.75)12 airfoil give desirable values of $c_{p_{min}}$ (fig. 4). However, multipliers greater than 0.12 produce airfoils with large forward thickness (fig. 2) that may cause an increase in drag with little change in the value of $c_{p_{min}}$. Therefore, the multiplier chosen for this section was 0.12.

Shift the Leading-Edge Suction Peak Toward the Trailing Edge

The distance from the leading edge to the location of the suction peak increases at large angles of attack as the value of the multiplier, MU, is raised. An increase in this distance produces a more-extensive favorable pressure gradient near the leading edge, which reduces boundary-layer growth and delays flow separation. The absolute value of the ratio of $c_{p_{min}}$ to the x/c location of $c_{p_{min}}$ is denoted as K. A reduction in K indicates that the location of $c_{p_{min}}$ is moving toward the trailing edge faster than the suction peak is decreased. An example demonstrating the importance of K is illustrated for the NACA 63(1.0)18 airfoil in figure 5. The distance $\left(\frac{x}{c}\right)$ from the leading edge to the location of the $c_{p_{min}}$ and the values of K are shown as a function of MU. The value of K decreases for all multipliers shown, whereas the lowest pressure peak occurred for a multiplier of 0.08. The code CLMAX (ref. 9) was used to obtain a theoretical $c_{l_{max}}$ for the modified sections. The interrelations between the theoretical $c_{l_{max}}$, the reduction of the pressure peak, and the movement of the pressure peak aft are demonstrated for the NACA 64-518 airfoil in figures 6(a)-6(c). Different multipliers are indicated by the three figures. The theoretical $c_{l_{max}}$ continues to improve with increasing multiplier values indicating that a value of at least 0.16 is required (fig. 6(a)). However, the optimum value of $c_{p_{min}}$ was obtained with a multiplier of 0.10 (fig. 6(b)), although the minimum K indicates a multiplier of 0.23 is required (fig. 6(c)). The multiplier actually chosen for this section was 0.12 because multipliers greater than 0.12 produced excessive bluntness and would thus cause larger drag at cruise flight.

If a multiplier of 0.23 was selected as dictated by K, the airfoil shown by the dashed line is produced.

The drag at cruise of this airfoil would be too large.

COMPARISON WITH EARLIER MODIFICATIONS

Three different upper-surface modifications to NACA 6-series airfoils were previously developed (refs. 2-4). All three modified airfoils were tested in wind tunnels to assess the improvements in $c_{l_{max}}$. The modifications described in this report are compared with the previously-developed modifications.

The modified NACA 63-215 (MOD B) airfoil from reference 2 is compared with the NACA 63-215 airfoil with 0.12 ESF added to the upper surface in figure 7(a). Note the similarity between the two airfoils. Although not apparent in the figure, the MOD B airfoil exhibits a slight discontinuity in curvature at approximately 35% chord because of the method used to modify the upper surface. The exponential function that was used here did not impose any discontinuity in curvature. Experimental lift curves (from ref. 2) for the modified and original NACA 63-215 airfoil are shown in figure 7(b).

The NACA 64-212 airfoil was also modified to achieve an increase in $c_{l_{max}}$ (ref. 3). This modified airfoil is denoted NACA 64-212 MOD B and is compared with the NACA 64-212 airfoil with 0.09 ESF added to the upper surface shown in figure 8(a). The present and earlier modification methods give similar shapes near the leading edge. However, the predicted pressure distributions near the leading edge for the two modified airfoils are different at an angle of attack of zero (compare figs. 8(b) and 8(c)). The predicted suction peaks for the two modified airfoils are identical at an angle of attack of 20° which indicates that the improvement in $c_{l_{max}}$ should be comparable to the value of 0.3 shown in figure 8(d) (from ref. 3).

An NACA 65(0.82)(9.9) airfoil was modified using the same exponential shape function presented here (ref. 4). The multiplier used for the previous modification was 0.09, with the result shown in figure 9(a). The experimental increase in $c_{l_{max}}$ of approximately 0.3 shown in figure 9(b) is taken from reference 4.

The improvements in $c_{l_{max}}$ shown for the three airfoils tested should be realized by the modified airfoils which can be obtained by the method described in this report.

DESIGN INSTRUCTIONS

The method presented in this report can be applied to any NACA 63-,64-, or 65-series airfoil. The coordinates of any NACA 6 or 6A-series airfoil can be generated by using the computer program of reference 10. These coordinates or an existing set of coordinates can be used with the ESF modification described here.

The airfoil coordinates must be made nondimensional by dividing by the chord length, if different from 1.0. The graphs of figures 10(a)-10(c) show the multiplier to be used with the ESF for any lift coefficient and thickness/chord ratio in the ranges of $0.0 \leq c_l \leq 1.0$, and $0.06 \leq (t/c)_{max} \leq 0.18$. If the design lift coefficient is positive, indicating a nonsymmetrical section, the ESF with the multipliers shown in these figures is added to the upper-surface ordinates of the airfoil. If the airfoil is symmetrical, the shape function with the multiplier shown is added to the upper surface and subtracted from the lower surface. This will result in a thickness increase on both surfaces of the airfoil. The approximate incremental increase in maximum thickness for the NACA 63-, 64-, and 65-series airfoils resulting from the modifications suggested here is shown in figures 11(a)-11(c). These values are approximate for two reasons: (1) the point of maximum camber occurs at different $\frac{x}{c}$ values for each design lift coefficient, and (2) the graphs are straight-line interpolations of the three thickness/chord ratios analyzed, $(t/c)_{max} = 0.06$, 0.12, and 0.18. The procedure for modifying NACA 6-series airfoils is summarized as follows.

1. Divide the (x, y) coordinates of the airfoil section by the length of the chord.
2. Find the multiplier, MU, by using the charts shown in figures 10(a)-10(c), and compute ESF.
3. Add ESF to nondimensional upper-surface ordinates of a nonsymmetrical airfoil. Also subtract ESF from the normalized lower surface ordinates if the airfoil is symmetrical.
4. Calculate the increase in thickness, if desired, from figures 11(a)-11(c).

A subset of the NACA 64-series airfoils modified by the multiplier of the ESF determined from figure 10(b) is shown in figure 12.

For example, the NACA 65-415 airfoil, which is widely used, is found on the Piper-Cherokee-series airplanes. This airfoil will be taken as an example to demonstrate the method described in this report. The coordinates of the NACA 65-415 airfoil are shown in table 1. These coordinates must first be made nondimensional by dividing by the chord length, which in this case is 63 in. The nondimensional coordinates are listed in table 2. The NACA 65-series modification chart is reproduced in figure 13. A line for a maximum thickness/chord ratio of 0.15 is not shown on the chart; therefore the correct value of the multiplier must be interpolated between the two closest thickness/chord ratios (0.14 and 0.16). Interpolation gives MU a value of 0.117. Add the function $ESF = 0.117\left(\frac{x}{c}\right)^{\frac{1}{2}}\left(1 - \frac{x}{c}\right)e^{-8\frac{x}{c}}$ to the nondimensional upper-surface ordinates to obtain the new ordinates for the modified airfoil. The coordinates for the modified airfoil are listed in table 3. Multiply the coordinates by the length of the chord to obtain the dimen-

sional coordinates of the new airfoil listed in table 4. The original and modified airfoils are illustrated in figure 14. The approximate increase in thickness for this example found from interpolation is 0.182 (fig. 11(c)).

CONCLUSION

A method for modifying any NACA 6- and 6A-series airfoils with maximum thickness/chord ratio between 0.06 and 0.18 and design lift coefficients between 0.0 and 1.0 has been developed. The modifications should produce substantial improvements in $c_{l_{max}}$. With only a hand-held calculator and the graphs given here, the designer can add an analytic function to the ordinates of the airfoil so as to achieve the desired improvement in $c_{l_{max}}$. The modifications produce slight increases in airfoil thickness which vary with thickness/chord ratio and design lift coefficient. The modification proposed can be retrofitted to existing wings or selected for new design applications.

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TABLE 1.- DIMENSIONAL COORDINATES FOR THE NACA 65-415 AIRFOIL

UPPER SURFACE		LOWER SURFACE	
X/C	Y/C	X/C	Y/C
0.00000	0.00000	0.00000	0.00000
0.06470	0.31872	0.06470	-0.34795
0.25861	0.65822	0.25861	-0.63674
0.58086	0.97385	0.58086	-0.89914
1.03024	1.29251	1.03024	-1.12909
1.60474	1.62445	1.60474	-1.33781
2.30215	1.97209	2.30215	-1.54382
3.11951	2.32804	3.11951	-1.75008
4.05348	2.68449	4.05348	-1.95155
5.10023	3.03950	5.10023	-2.14729
6.25546	3.38934	6.25546	-2.33780
7.51445	3.72884	7.51445	-2.51994
8.87197	4.05468	8.87197	-2.69174
10.32255	4.36395	10.32255	-2.85226
11.86007	4.65368	11.86007	-3.00031
13.47835	4.92087	13.47835	-3.13425
15.17065	5.16260	15.17065	-3.25237
16.93005	5.37610	16.93005	-3.35305
18.74930	5.55868	18.74930	-3.43476
20.62103	5.70692	20.62103	-3.49474
22.53737	5.81937	22.53737	-3.53209
24.49062	5.89201	24.49062	-3.54394
26.47260	5.91683	26.47260	-3.52246
28.47531	5.88798	28.47531	-3.46172
30.49036	5.80318	30.49036	-3.35960
32.50964	5.66093	32.50964	-3.21464
34.52469	5.46405	34.52469	-3.03024
36.52740	5.22068	36.52740	-2.81497
38.50938	4.93989	38.50938	-2.57827
40.46263	4.62943	40.46263	-2.32760
42.37903	4.29408	42.37903	-2.06785
44.25069	3.94178	44.25069	-1.80671
46.06995	3.58004	46.06995	-1.55068
47.82935	3.21382	47.82935	-1.30372
49.52166	2.84791	49.52166	-1.06987
51.13993	2.48844	51.13993	-0.85384
52.67745	2.14049	52.67745	-0.65911
54.12803	1.80823	54.12803	-0.48825
55.48555	1.49549	55.48555	-0.34304
56.74454	1.20607	56.74454	-0.22548
57.89977	0.94223	57.89977	-0.13532
58.94652	0.70623	58.94652	-0.07239
59.88049	0.49915	59.88049	-0.03522
60.69785	0.32180	60.69785	-0.02180
61.39526	0.17382	61.39526	-0.02885
61.96976	0.05456	61.96976	-0.05065
62.41914	-0.03875	62.41914	-0.08404
62.74139	-0.10811	62.74139	-0.12310
62.93530	-0.15158	62.93530	-0.15441
63.00000	-0.16638	63.00000	-0.16638

TABLE 2.- NORMALIZED COORDINATES FOR THE NACA 65-415 AIRFOIL

UPPER SURFACE		LOWER SURFACE	
X/C	Y/C	X/C	Y/C
0.00000	0.00000	0.00000	0.00000
0.00103	0.00506	0.00103	-0.00552
0.00410	0.01045	0.00410	-0.01011
0.00922	0.01546	0.00922	-0.01427
0.01635	0.02052	0.01635	-0.01792
0.02547	0.02578	0.02547	-0.02124
0.03654	0.03130	0.03654	-0.02451
0.04952	0.03695	0.04952	-0.02778
0.06434	0.04261	0.06434	-0.03098
0.08096	0.04825	0.08096	-0.03408
0.09929	0.05380	0.09929	-0.03711
0.11928	0.05919	0.11928	-0.04000
0.14083	0.06436	0.14083	-0.04273
0.16385	0.06927	0.16385	-0.04527
0.18825	0.07387	0.18825	-0.04762
0.21394	0.07811	0.21394	-0.04975
0.24080	0.08195	0.24080	-0.05162
0.26873	0.08534	0.26873	-0.05322
0.29761	0.08823	0.29761	-0.05452
0.32732	0.09059	0.32732	-0.05547
0.35774	0.09237	0.35774	-0.05607
0.38874	0.09352	0.38874	-0.05625
0.42020	0.09392	0.42020	-0.05591
0.45199	0.09346	0.45199	-0.05495
0.48397	0.09211	0.48397	-0.05333
0.51603	0.08986	0.51603	-0.05103
0.54801	0.08673	0.54801	-0.04810
0.57980	0.08287	0.57980	-0.04468
0.61126	0.07841	0.61126	-0.04092
0.64226	0.07348	0.64226	-0.03695
0.67268	0.06816	0.67268	-0.03282
0.70239	0.06257	0.70239	-0.02868
0.73127	0.05683	0.73127	-0.02461
0.75920	0.05101	0.75920	-0.02069
0.78606	0.04521	0.78606	-0.01698
0.81174	0.03950	0.81174	-0.01355
0.83615	0.03398	0.83615	-0.01046
0.85918	0.02870	0.85918	-0.00775
0.88072	0.02374	0.88072	-0.00544
0.90071	0.01914	0.90071	-0.00358
0.91904	0.01496	0.91904	-0.00215
0.93566	0.01121	0.93566	-0.00115
0.95048	0.00792	0.95048	-0.00056
0.96346	0.00511	0.96346	-0.00035
0.97453	0.00276	0.97453	-0.00046
0.98365	0.00087	0.98365	-0.00080
0.99078	-0.00062	0.99078	-0.00133
0.99590	-0.00172	0.99590	-0.00195
0.99897	-0.00241	0.99897	-0.00245
1.00000	-0.00264	1.00000	-0.00264

TABLE 3.- NORMALIZED COORDINATES FOR THE MODIFIED NACA 65-415 AIRFOIL

UPPER SURFACE		LOWER SURFACE	
X/C	Y/C	X/C	Y/C
0.00000	0.00000	0.00000	0.00000
0.00103	0.00878	0.00103	-0.00552
0.00410	0.01769	0.00410	-0.01011
0.00922	0.02582	0.00922	-0.01427
0.01635	0.03346	0.01635	-0.01792
0.02547	0.04066	0.02547	-0.02124
0.03654	0.04742	0.03654	-0.02451
0.04952	0.05364	0.04952	-0.02778
0.06434	0.05924	0.06434	-0.03098
0.08096	0.06429	0.08096	-0.03408
0.09929	0.06884	0.09929	-0.03711
0.11928	0.07292	0.11928	-0.04000
0.14083	0.07661	0.14083	-0.04273
0.16385	0.07997	0.16385	-0.04527
0.18825	0.08303	0.18825	-0.04762
0.21394	0.08581	0.21394	-0.04975
0.24080	0.08831	0.24080	-0.05162
0.26873	0.09051	0.26873	-0.05322
0.29761	0.09239	0.29761	-0.05452
0.32732	0.09388	0.32732	-0.05547
0.35774	0.09495	0.35774	-0.05607
0.38874	0.09552	0.38874	-0.05625
0.42020	0.09545	0.42020	-0.05591
0.45199	0.09462	0.45199	-0.05495
0.48397	0.09299	0.48397	-0.05333
0.51603	0.09051	0.51603	-0.05103
0.54801	0.08722	0.54801	-0.04810
0.57980	0.08323	0.57980	-0.04468
0.61126	0.07868	0.61126	-0.04092
0.64226	0.07368	0.64226	-0.03695
0.67268	0.06831	0.67268	-0.03282
0.70239	0.06267	0.70239	-0.02868
0.73127	0.05690	0.73127	-0.02461
0.75920	0.05107	0.75920	-0.02069
0.78606	0.04525	0.78606	-0.01698
0.81174	0.03953	0.81174	-0.01355
0.83615	0.03400	0.83615	-0.01046
0.85918	0.02872	0.85918	-0.00775
0.88072	0.02375	0.88072	-0.00544
0.90071	0.01915	0.90071	-0.00358
0.91904	0.01496	0.91904	-0.00215
0.93566	0.01121	0.93566	-0.00115
0.95048	0.00793	0.95048	-0.00056
0.96346	0.00511	0.96346	-0.00035
0.97453	0.00276	0.97453	-0.00046
0.98365	0.00087	0.98365	-0.00080
0.99078	-0.00062	0.99078	-0.00133
0.99590	-0.00172	0.99590	-0.00195
0.99897	-0.00241	0.99897	-0.00245
1.00000	-0.00264	1.00000	-0.00264

TABLE 4.- DIMENSIONAL COORDINATES FOR THE MODIFIED NACA 65-415 AIRFOIL

UPPER SURFACE		LOWER SURFACE	
X/C	Y/C	X/C	Y/C
0.00000	0.00000	0.00000	0.00000
0.06470	0.55327	0.06470	-0.34795
0.25861	1.11434	0.25861	-0.63674
0.58086	1.62660	0.58086	-0.89914
1.03024	2.10773	1.03024	-1.12909
1.60474	2.56152	1.60474	-1.33781
2.30215	2.98771	2.30215	-1.54382
3.11951	3.37938	3.11951	-1.75008
4.05348	3.73225	4.05348	-1.95155
5.10023	4.05027	5.10023	-2.14729
6.25546	4.33673	6.25546	-2.33780
7.51445	4.59415	7.51445	-2.51994
8.87197	4.82662	8.87197	-2.69174
10.32255	5.03798	10.32255	-2.85226
11.86007	5.23070	11.86007	-3.00031
13.47835	5.40584	13.47835	-3.13425
15.17065	5.56347	15.17065	-3.25237
16.93005	5.70232	16.93005	-3.35305
18.74930	5.82044	18.74930	-3.43476
20.62103	5.91419	20.62103	-3.49474
22.53737	5.98160	22.53737	-3.53209
24.49062	6.01757	24.49062	-3.54394
26.47260	6.01310	26.47260	-3.52246
28.47531	5.96119	28.47531	-3.46172
30.49036	5.85837	30.49036	-3.35960
32.50964	5.70232	32.50964	-3.21464
34.52469	5.49486	34.52469	-3.03024
36.52740	5.24355	36.52740	-2.81497
38.50938	4.95678	38.50938	-2.57827
40.46263	4.64184	40.46263	-2.32760
42.37903	4.30322	42.37903	-2.06785
44.25069	3.94846	44.25069	-1.80671
46.06995	3.58495	46.06995	-1.55068
47.82935	3.21741	47.82935	-1.30372
49.52166	2.85050	49.52166	-1.06987
51.13993	2.49033	51.13993	-0.85384
52.67745	2.14187	52.67745	-0.65911
54.12803	1.80923	54.12803	-0.48825
55.48555	1.49619	55.48555	-0.34304
56.74454	1.20658	56.74454	-0.22548
57.89977	0.94261	57.89977	-0.13532
58.94652	0.70648	58.94652	-0.07239
59.88049	0.49934	59.88049	-0.03522
60.69785	0.32193	60.69785	-0.02180
61.39526	0.17388	61.39526	-0.02885
61.96976	0.05462	61.96976	-0.05065
62.41914	-0.03875	62.41914	-0.08404
62.74139	-0.10811	62.74139	-0.12310
62.93530	-0.15158	62.93530	-0.15441
63.00000	-0.16638	63.00000	-0.16638

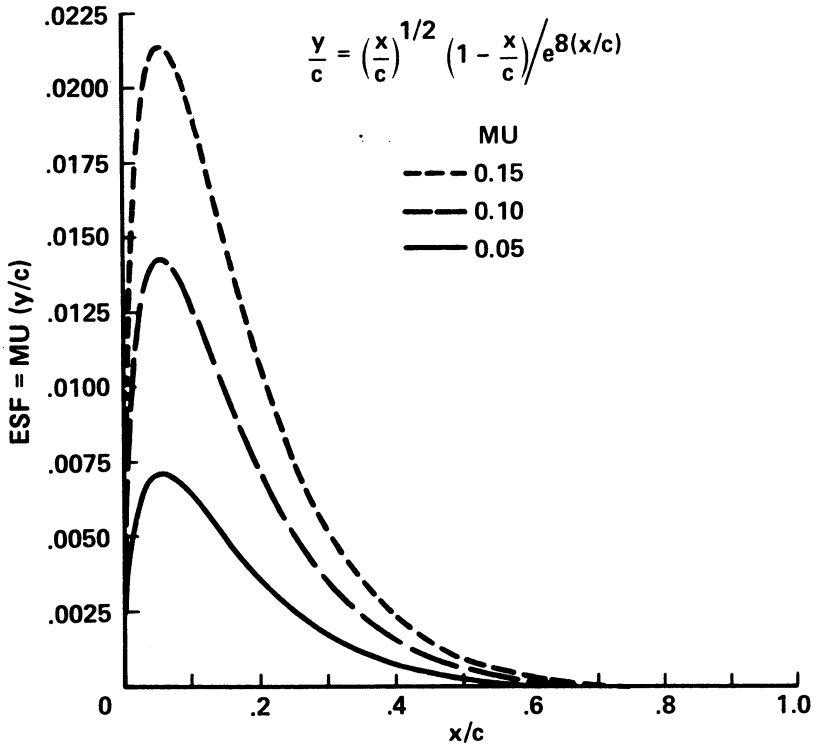

Figure 1.- Exponential shape function used to modify NACA 6-series airfoils.

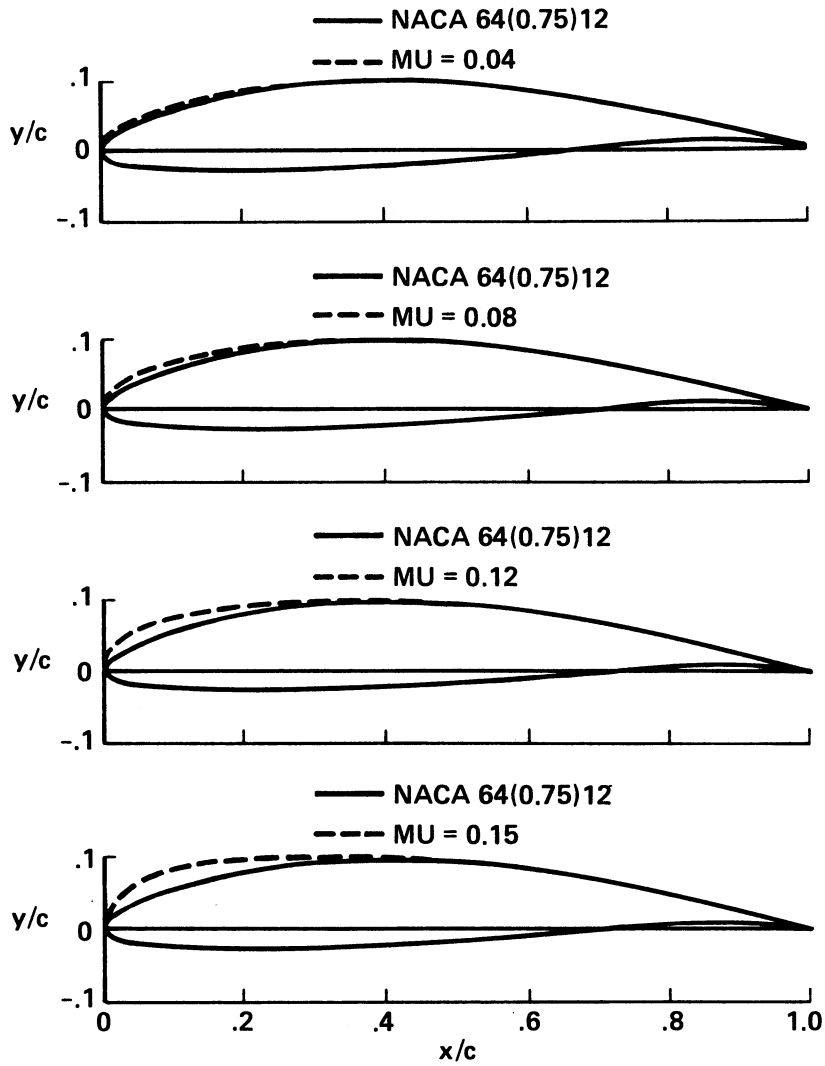

Figure 2.- Effect of multiplier on the forward camber of the NACA 64(0.75)12 airfoil.

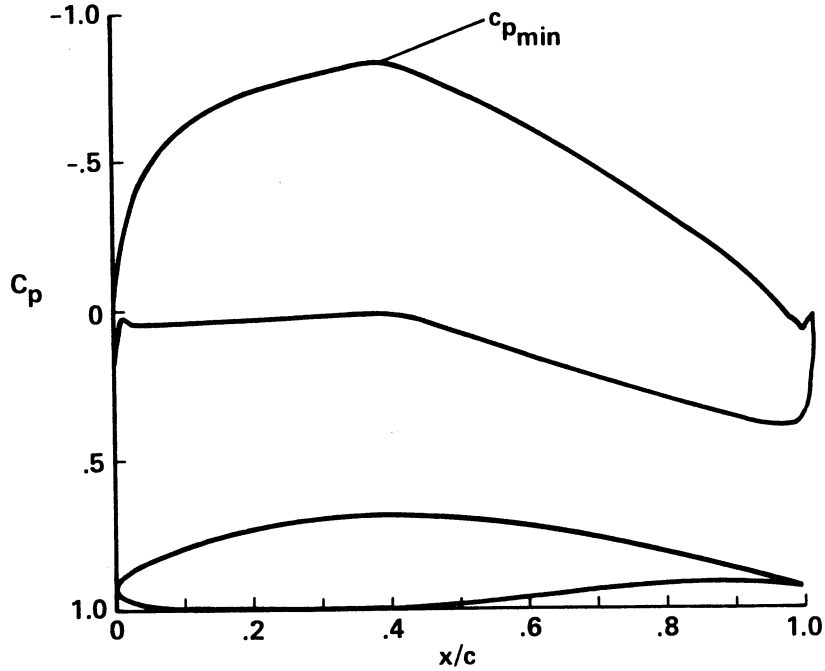

Figure 3(a).- Original NACA 64(0.75)12 airfoil; $\alpha = 0.0°$, Re = 6 million, $c_l = 0.67$, $c_m = -0.155$, $c_d(form) = 0.0057$.

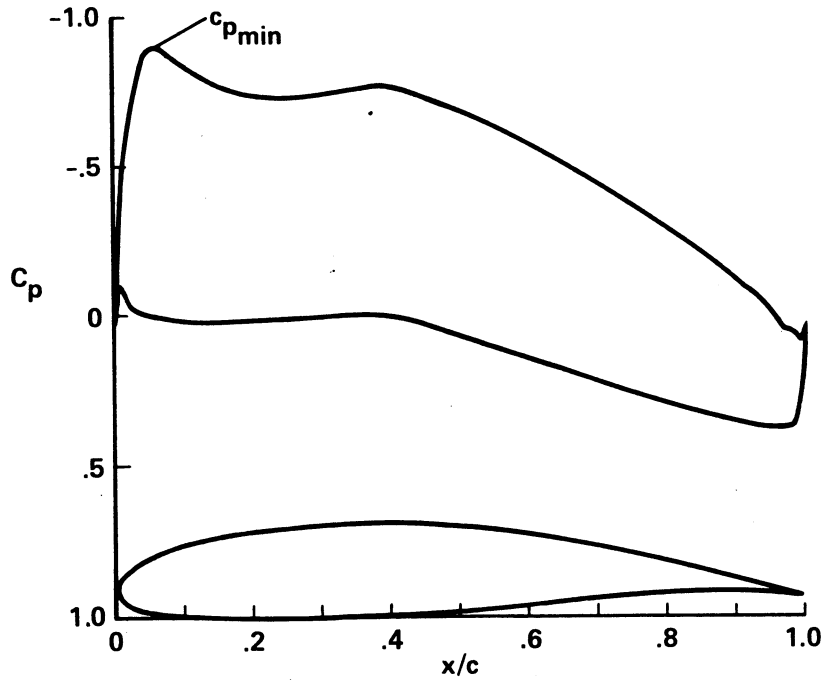

Figure 3(b).- Modified NACA 64(0.75)12 airfoil; $\alpha = 0.0°$, Re = 6 million, $c_l = 0.66$, $c_m = -0.139$, $c_d(form) = 0.0079$.

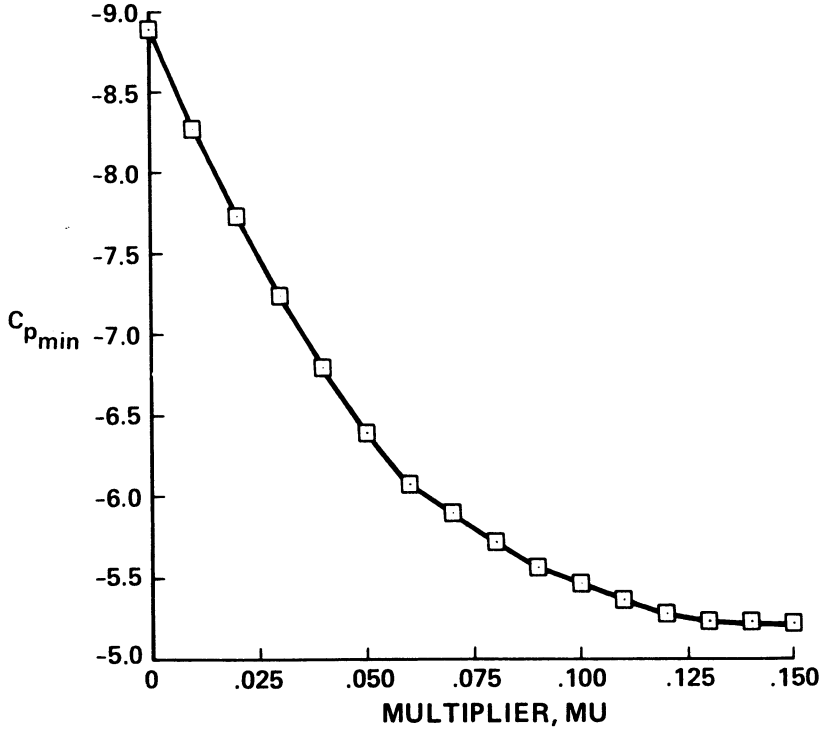

Figure 4.- Effect of multiplier, MU, on $c_{p_{min}}$ for the NACA 64(0.75)12 airfoil; $\alpha = 10°$, $M = 0.1$.

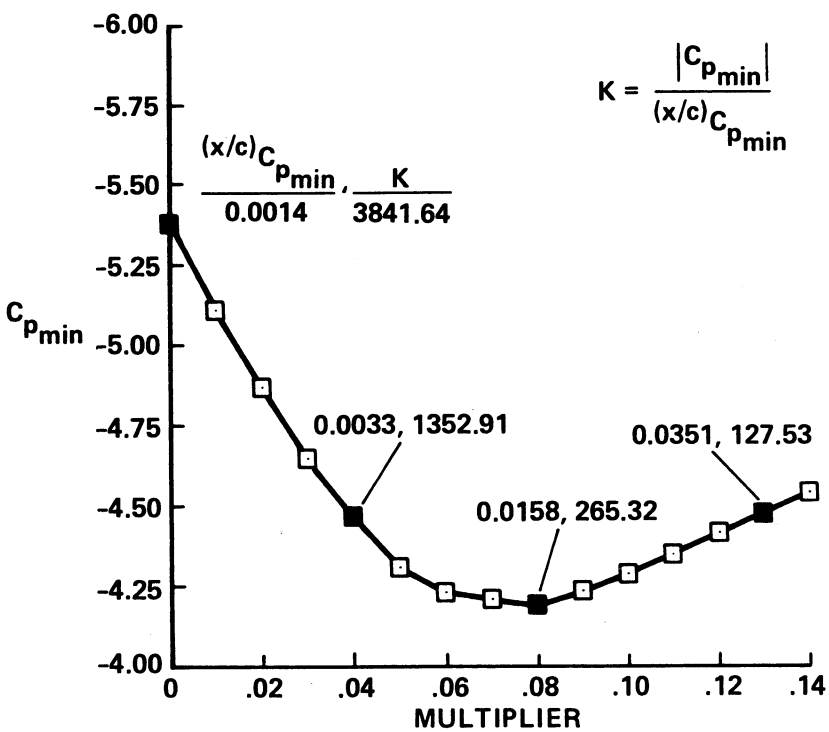

Figure 5.- Effect of multiplier on $c_{p_{min}}$ with K-values, and location of $c_{p_{min}}$ for the NACA 63(1.0)18 airfoil, $\alpha = 10°$, $M = 0.1$.

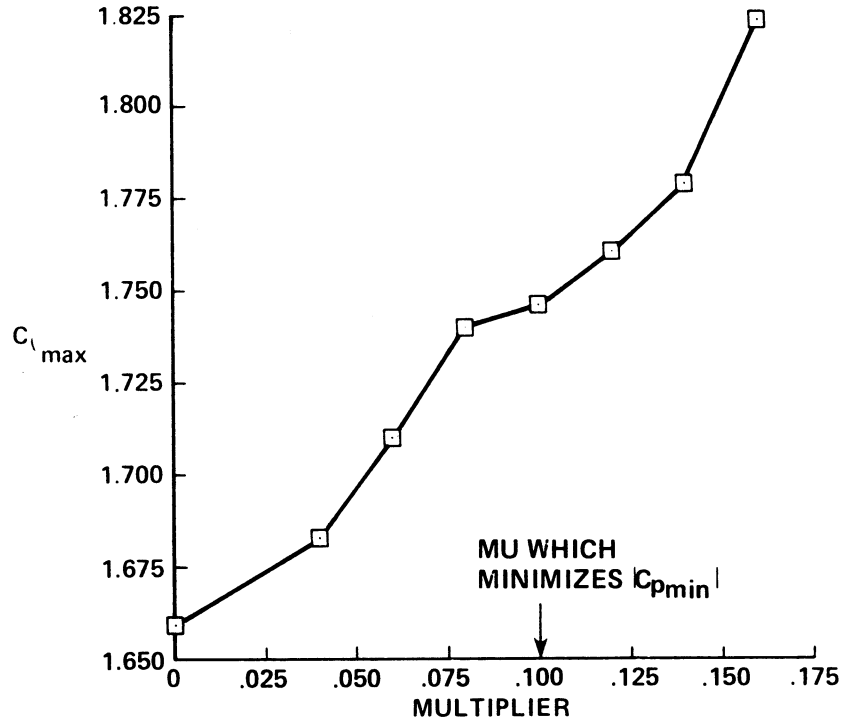

Figure 6(a).- Effect of multiplier on $c_{l_{max}}$ as predicted by program CLMAX for the NACA 64-518 airfoil, $M = 0.1$, Re = 6 million.

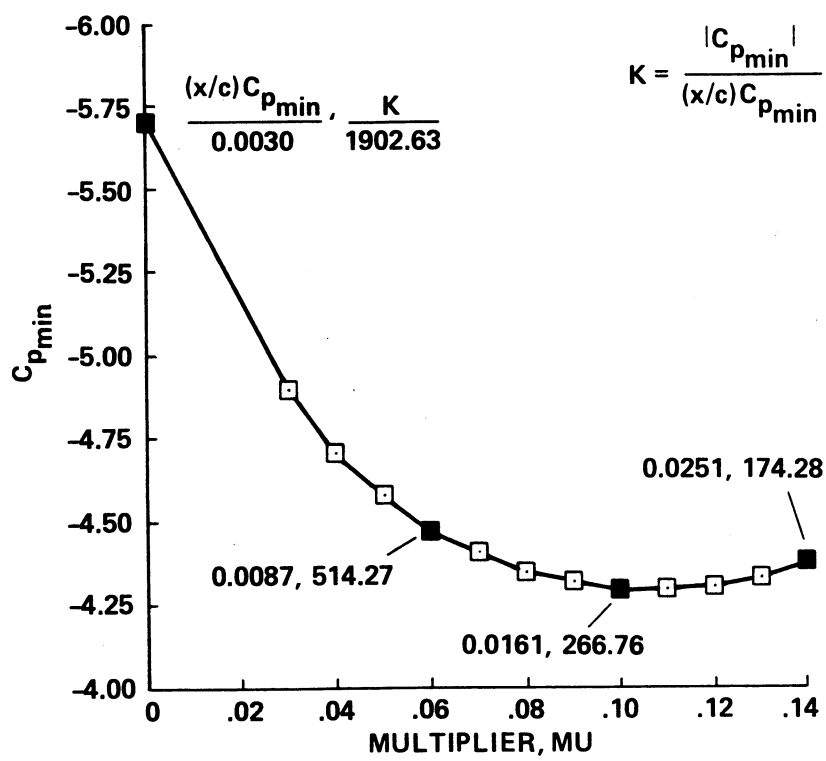

Figure 6(b).- Effect of multiplier on $c_{p_{min}}$ with K-values and location of $c_{p_{min}}$ for the NACA 64-518 airfoil, $\alpha = 10°$, $M = 0.1$.

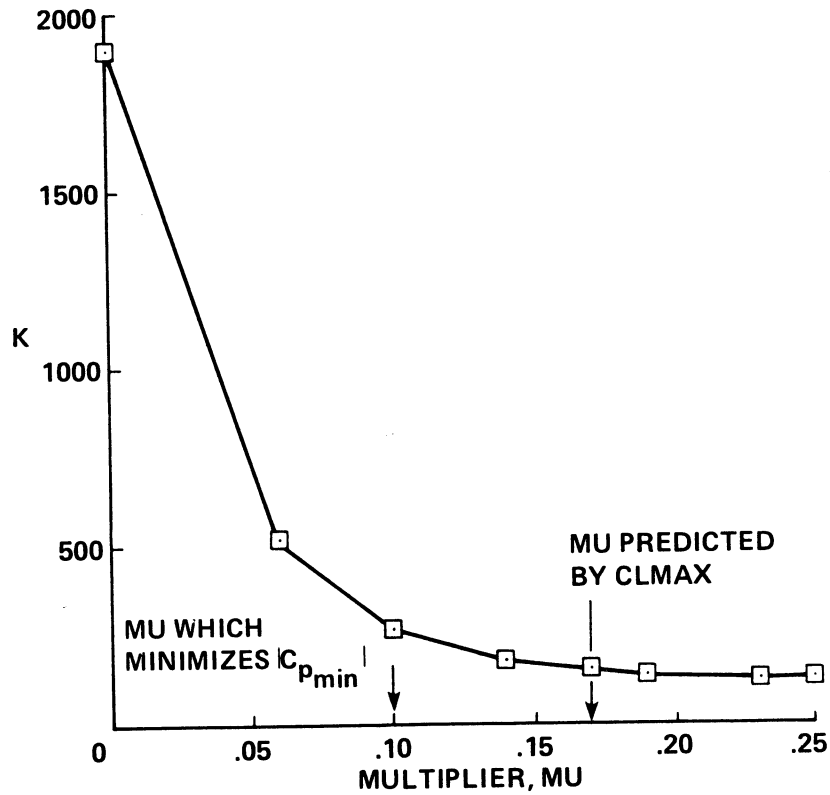

Figure 6(c).- Effect of multiplier on K-values.

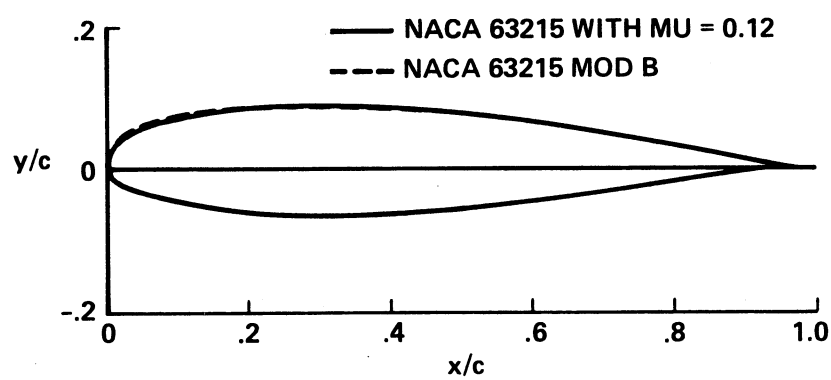

Figure 7(a).- Profile of the NACA 63-215 airfoil with 0.12 ESF added to the upper surface, compared with the NACA 63-215 MOD B airfoil of reference 3.

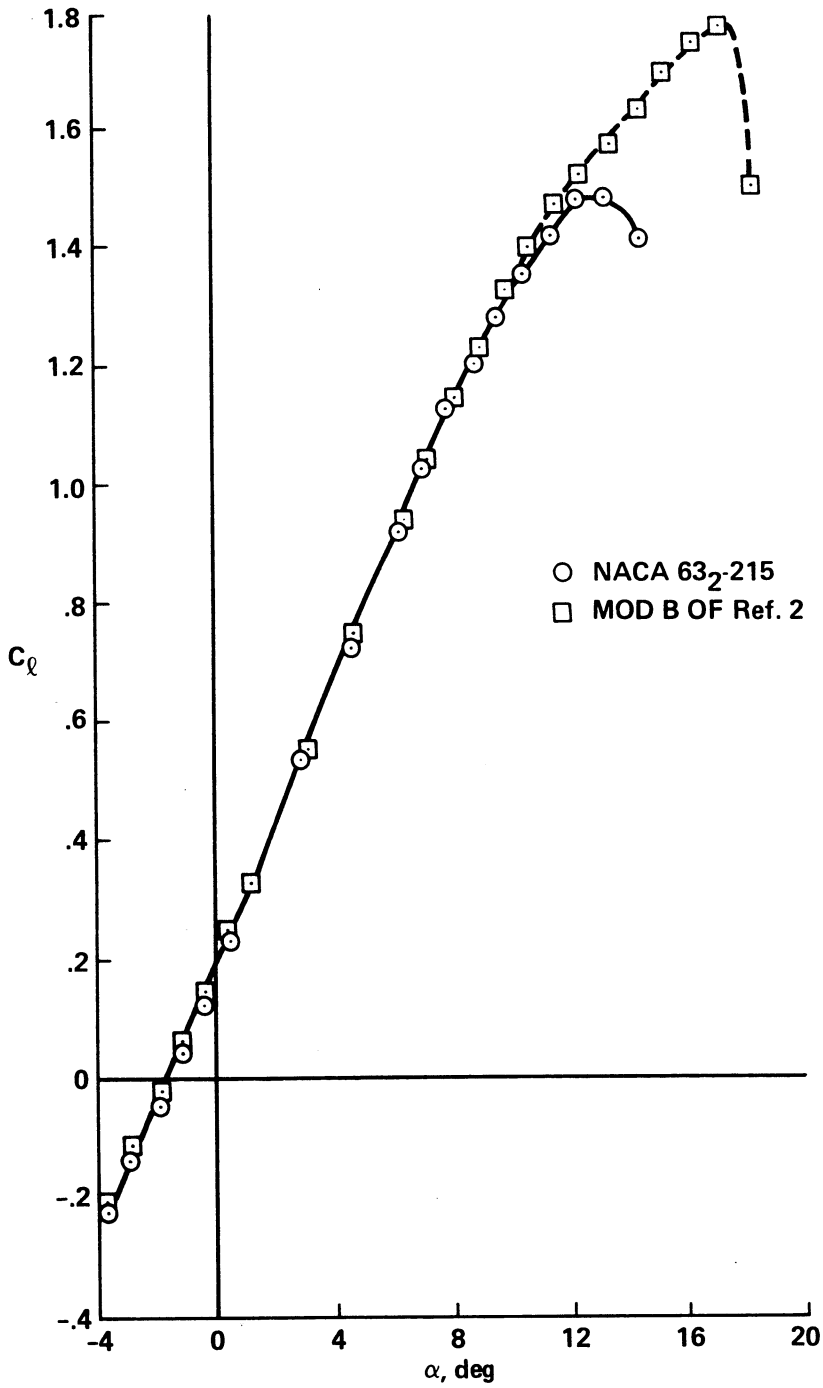

Figure 7(b).- Experimental data showing the effect of adding forward camber to the NACA 63-215 airfoil, $M = 0.20$, Re $= 2.5 \times 10^6$.

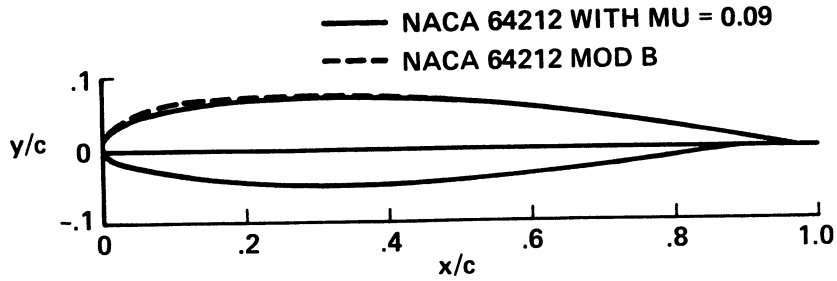

Figure 8(a).- NACA 64-212 airfoil with 0.09 ESF added to the upper surface compared with the NACA 64-212 MOD B of reference 2.

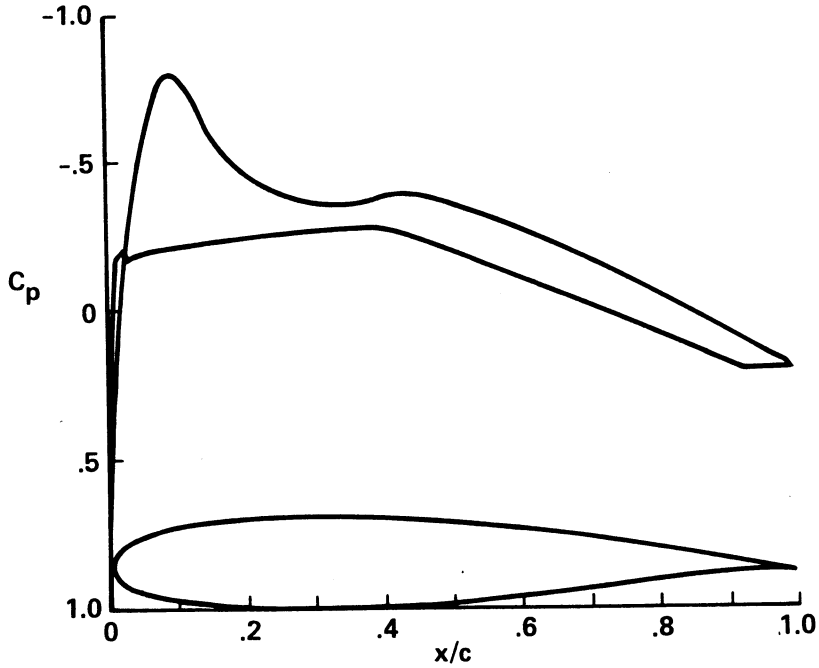

Figure 8(b).- Pressure distribution for the NACA 64-212 MOD B airfoil predicted by Program H, $\alpha = 0.0°$, $M = 0.1$, Re $= 6.0 \times 10^6$, $c_l = 0.165$, $c_m = -0.0247$, $c_d(form) = 0.0062$.

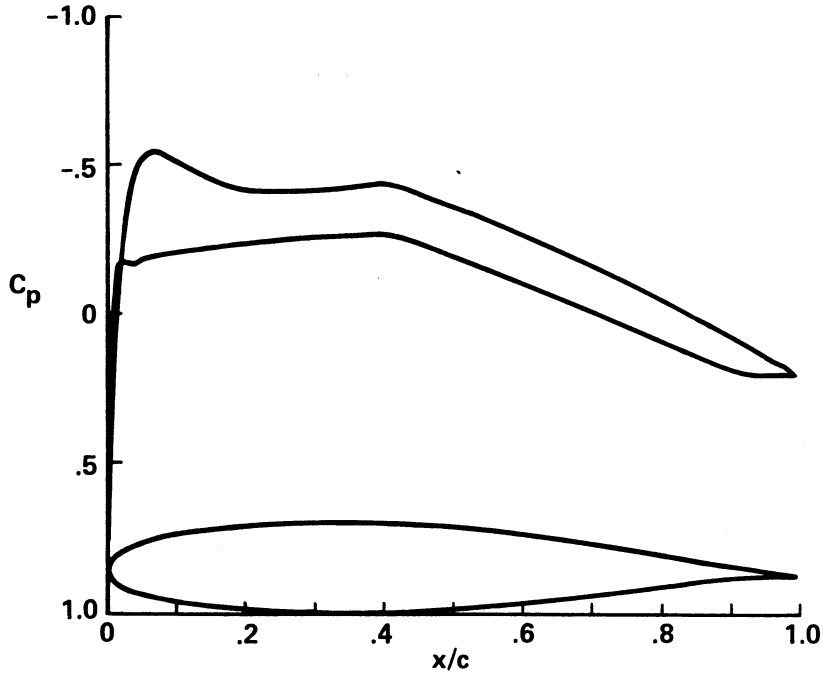

Figure 8(c).- Pressure distribution predicted by Program H for the NACA 64-212 airfoil with 0.09 ESF added to the upper surface, $\alpha = 0.0°$, $M = 0.1$, Re $= 6.0 \times 10^6$, $c_l = 0.162$, $c_m = -0.0283$, $c_d(form) = 0.0062$.

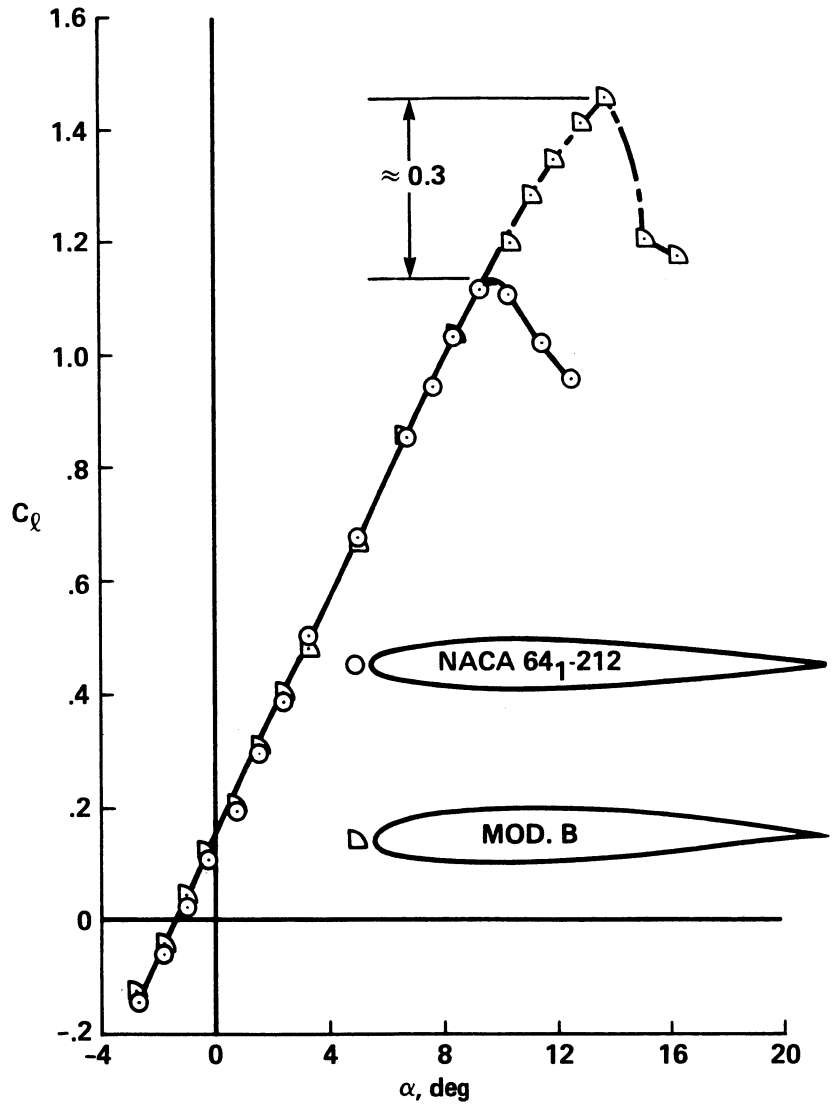

Figure 8(d).- Lift curves for the NACA 64-212 and the NACA 64-212 MOD B airfoils showing improvement in $c_{l_{max}}$, $M = 0.20$, Re $= 1.9 \times 10^6$.

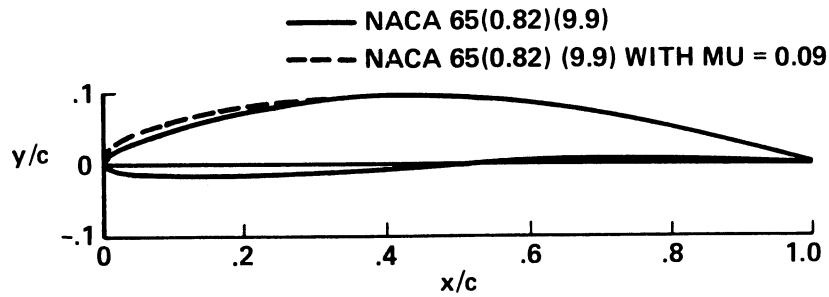

Figure 9(a).- NACA 65(0.82)(9.9) airfoil with 0.09 ESF added to the upper surface.

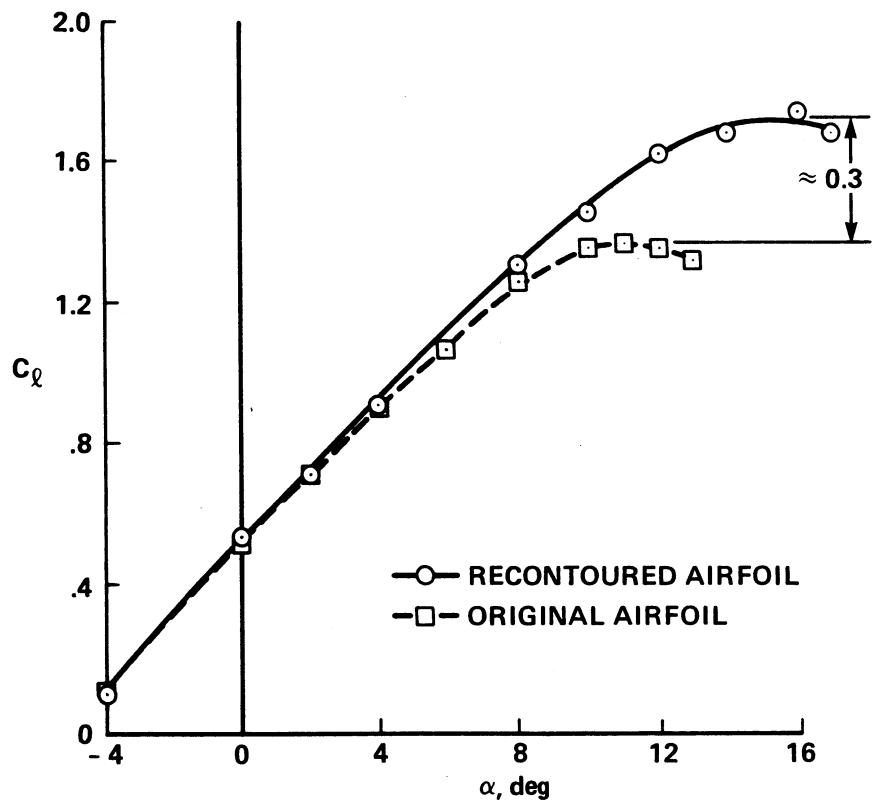

Figure 9(b).- Experimental lift curves for the NACA 65(0.82)(9.9) airfoil and the NACA 65(0.82)(9.9) airfoil with 0.09 ESF added to the upper surface, $M = 0.3$, $\mathrm{Re} = 4.3 \times 10^6$.

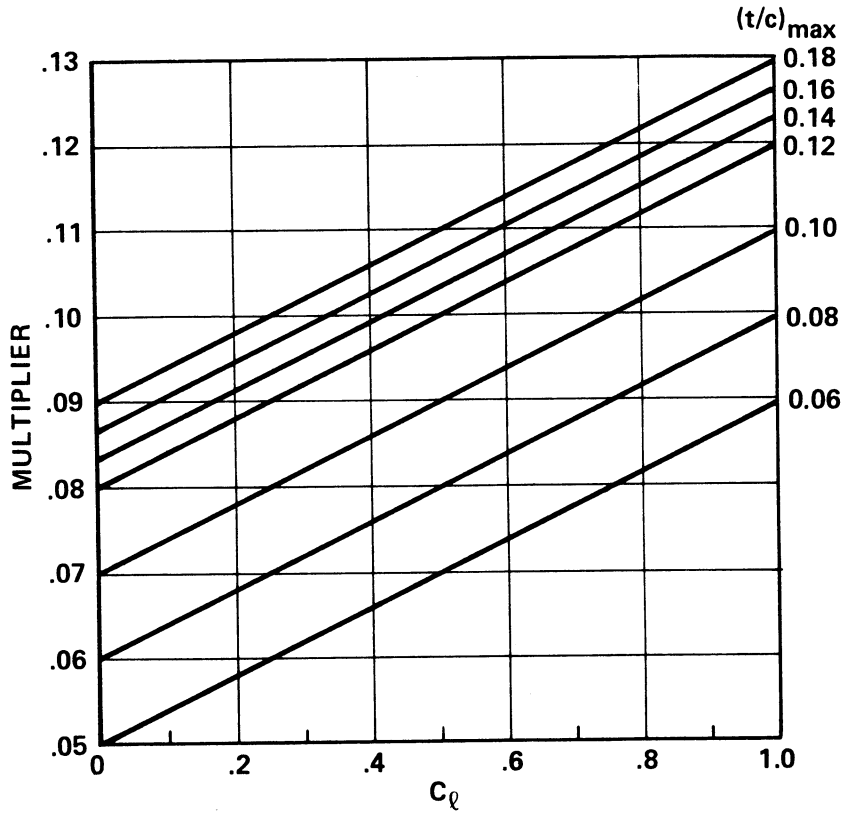

Figure 10(a).- Exponential function multiplier required for $c_{l_{max}}$ improvement for the NACA 63-series airfoils.

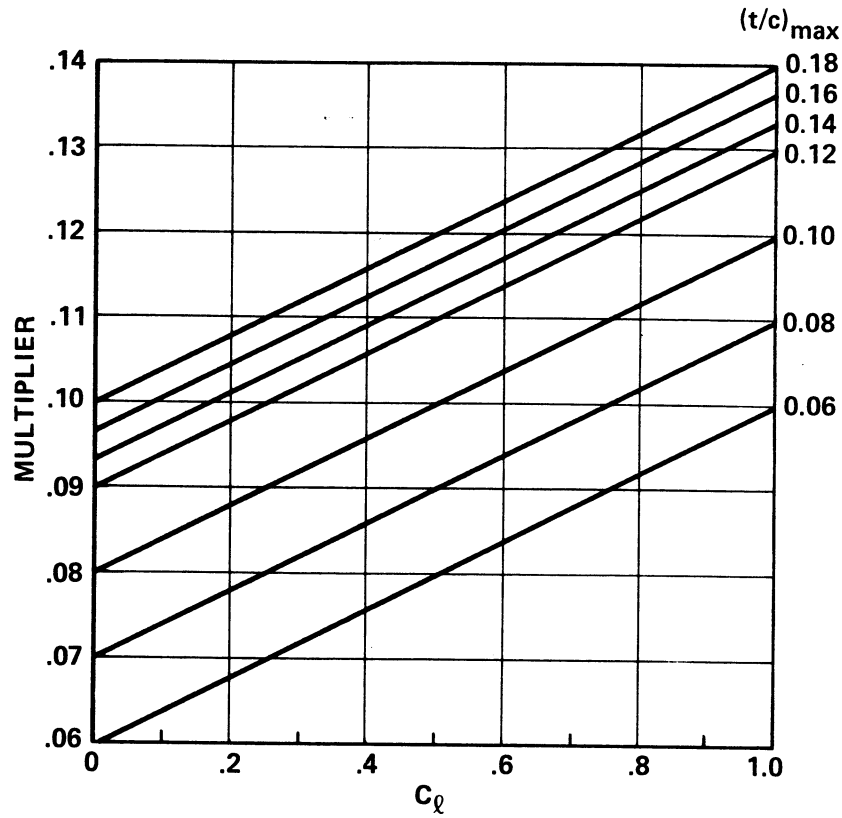

Figure 10(b).- Exponential function multiplier required for $c_{l_{max}}$ improvement for the NACA 64-series airfoils.

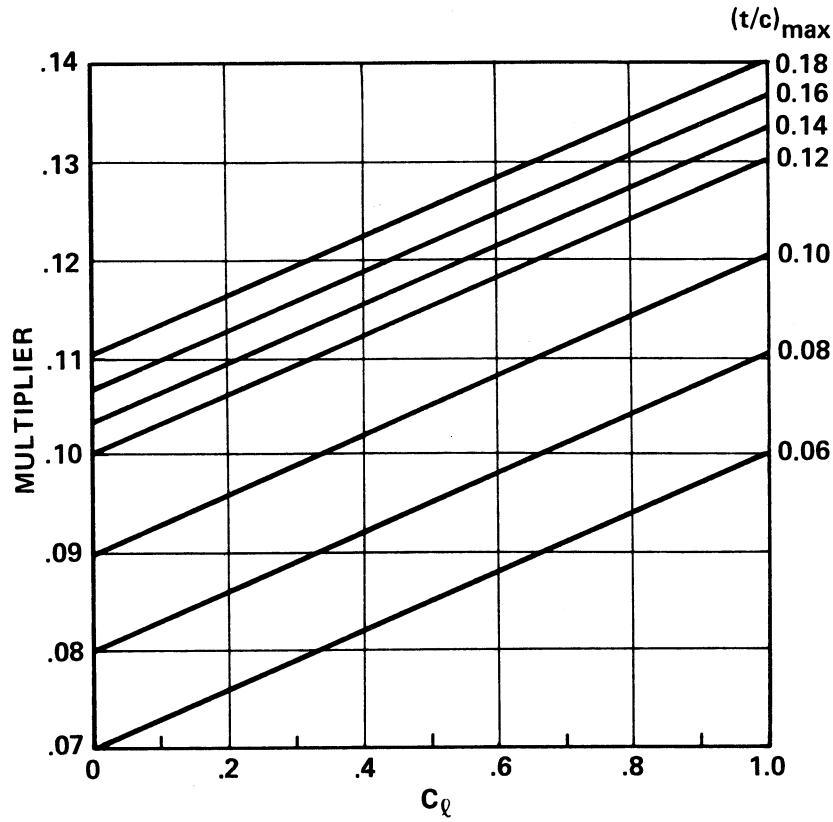

Figure 10(c).- Exponential function multiplier required for $c_{l_{max}}$ improvement for the NACA 65-series airfoils.

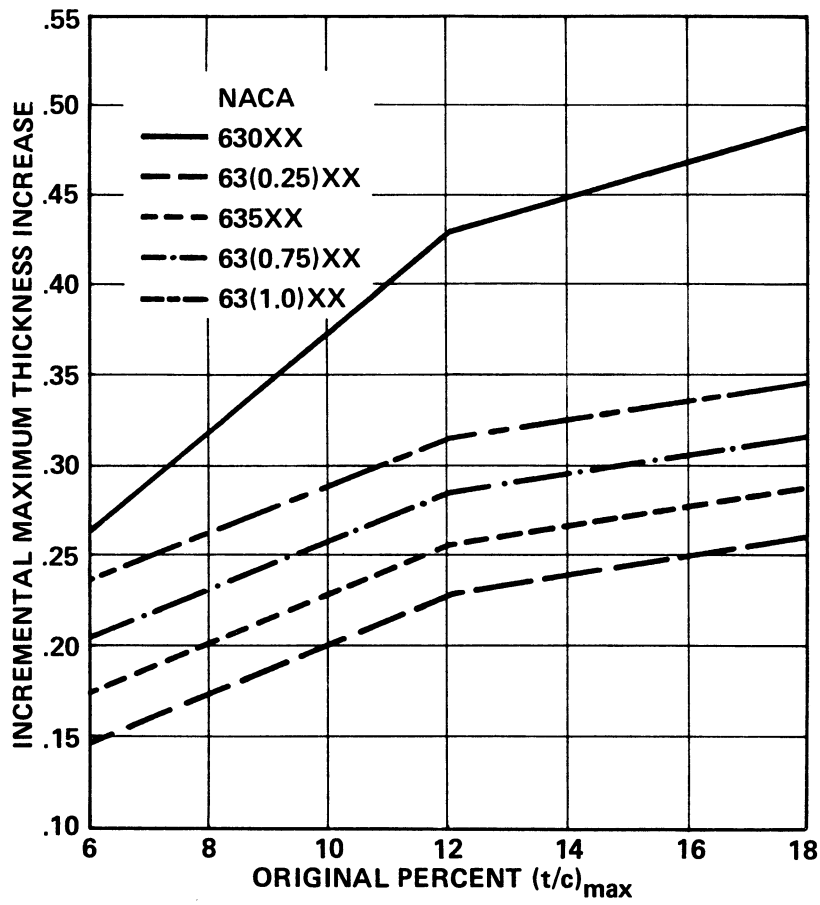

Figure 11(a).- Increase in $(t/c)_{max}$ resulting from the addition of the ESF to the NACA 63-series airfoils.

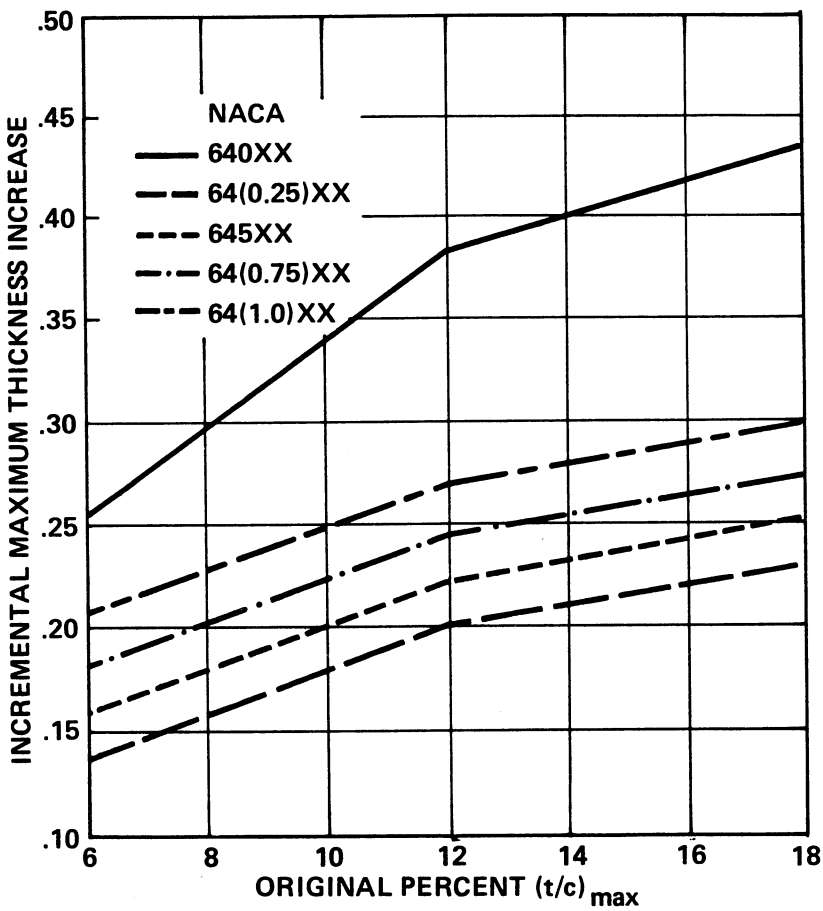

Figure 11(b).- Increase in $(t/c)_{max}$ resulting from the addition of the ESF to the NACA 64-series airfoils.

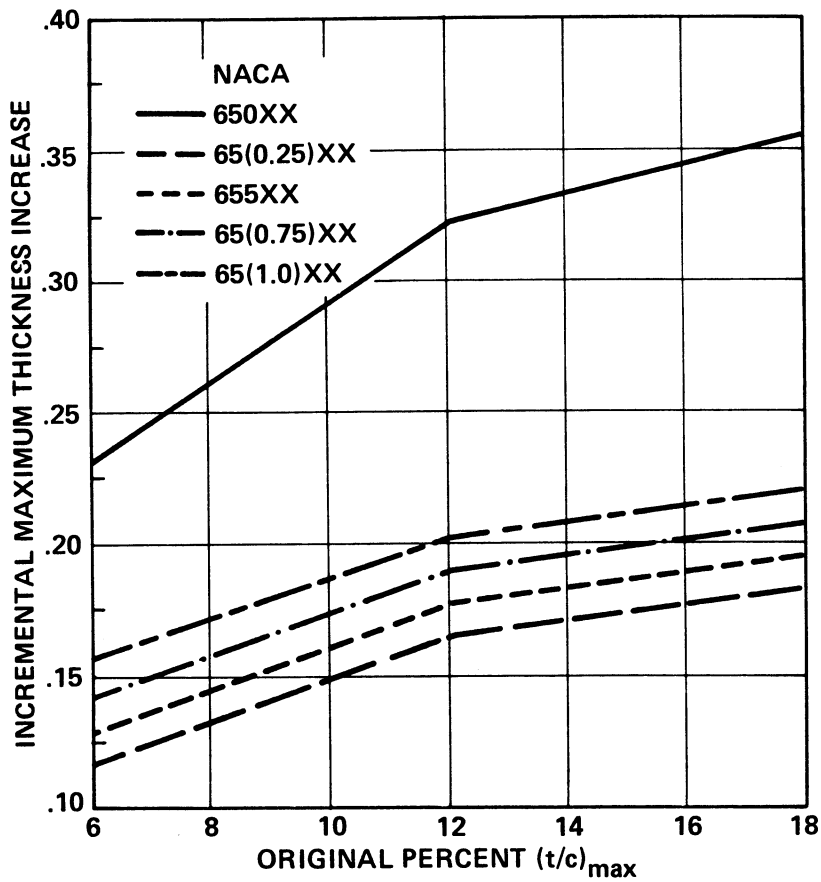

Figure 11(c).- Increase in $(t/c)_{max}$ resulting from the addition of the ESF to the NACA 65-series airfoils.

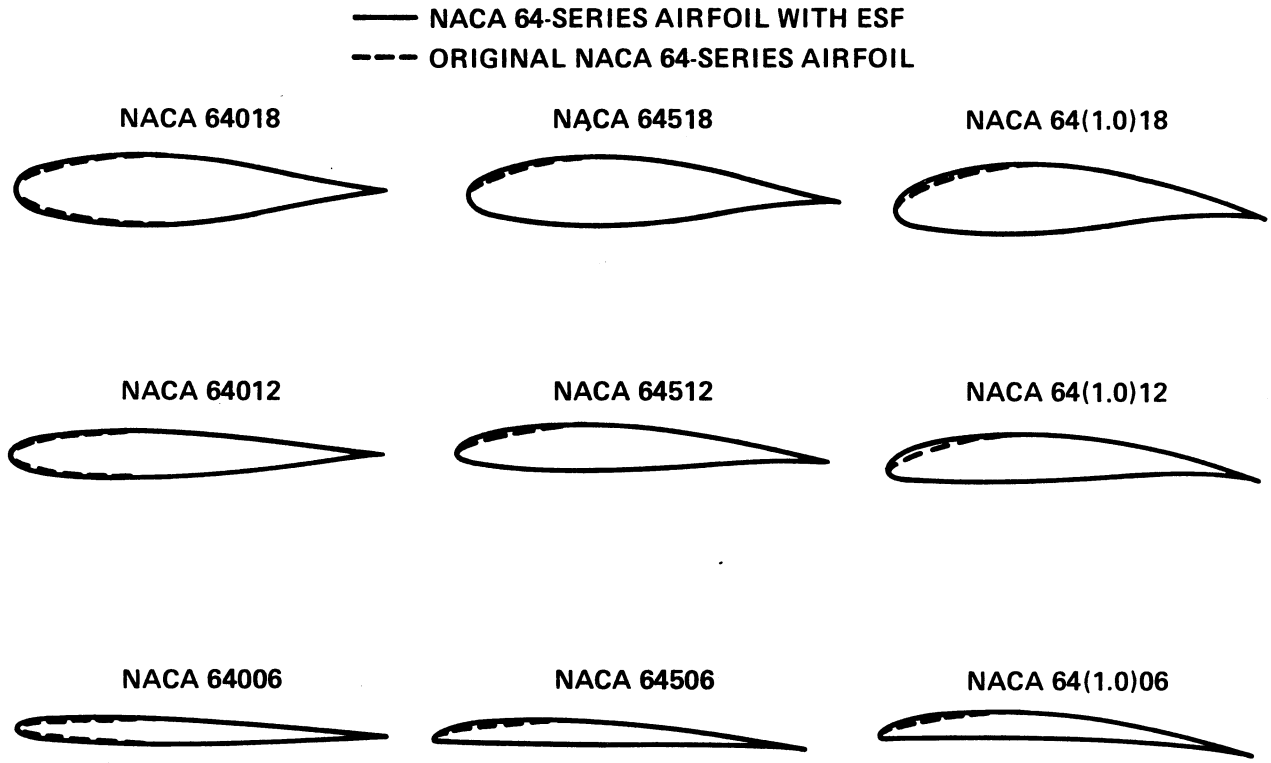

Figure 12.- Recommended modifications for selected NACA 64-series airfoils.

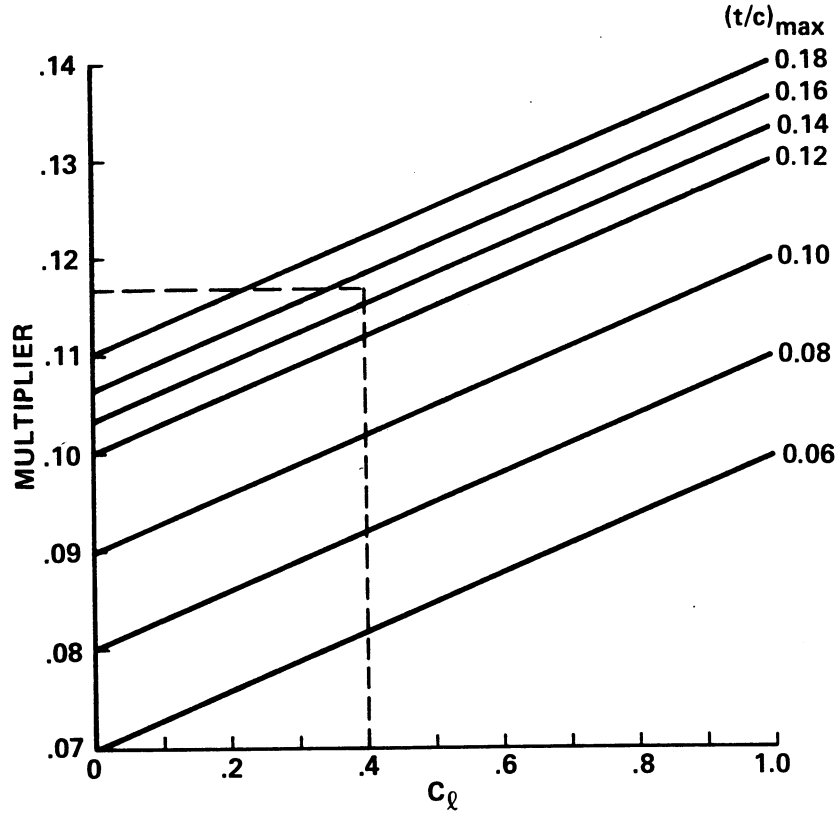

Figure 13.- Determination of the multiplier for the NACA 65-415 airfoil.

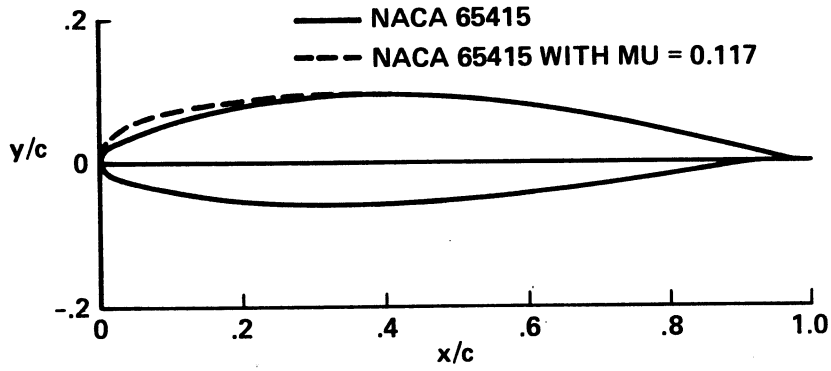

Figure 14.- Modified airfoil resulting from the addition of 0.117 ESF to the NACA 65-415 airfoil with the multiplier obtained from figure 18.

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Report Documentation Page

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15. Supplementary Notes

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16. Abstract

A simple method has been developed for computing modifications to the forward camber and thickness of any NACA 63-, 64-, or 65-series airfoil with the objective of achieving an increase in the maximum lift coefficient. The method can be used for any design lift coefficient between 0.0 and 1.0, and for any maximum thickness/chord ratio between 0.06 and 0.18. A simple mathematical function defines an incremental thickness which is applied to the upper surface of a cambered airfoil, or is applied to both surfaces of an uncambered airfoil. The method is sufficiently compact to permit solutions to be obtained on a hand-held scientific calculator containing logarithmic functions and a power function. In applying the method, the incremental thickness is added to the thickness of the forward portion of the airfoil, thus increasing the bluntness of the leading edge and increasing the forward camber of cambered airfoils. This does not introduce any discontinuities in curvature, since the function and its derivatives are continuous over the entire chord length. An increase in the forward thickness of the NACA 6-series airfoils will reduce the adverse pressure gradient near the leading edge on the upper surface at high angles of attack, thereby postponing separation until a higher angle of attack is reached, which results in an increased $c_{l_{max}}$. An increased $c_{l_{max}}$ offers improved stalling characteristics and slower, shorter landing runs. The lower drag resulting from delayed separation at higher lift coefficients will result in an improved rate of climb.

17. Key Words (Suggested by Author(s)): Airflow design, Computational fluid dynamics, General aviation		18. Distribution Statement: Unclassified—Unlimited; Subject Category: 02	
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