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ON THE MAXIMUM RANGE OF FLYING WINGS

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ABSTRACT

The classical equations for determining the maximum range of aircraft with propellor and jet propulsion systems are reviewed, along with previous work conducted to determine the optimal division of aircraft volume between fuselage and wing components. That the jet powered flying wing configuration produces optimal range only for limited geometries is confirmed. The optimal range of aircraft employing high bypass jet engines is explored, and found to lead to a broader range of design parameters for which the flying wing design produces maximum range than is the case when a pure jet system is used.

FIGURES

On the Maximum Range of Flying Wings

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I. INTRODUCTION

The recent unveiling of the B-2 or "stealth bomber," together with a recent communication in another journal (Ref 1), have created a renewed interest in the aerodynamic efficiency of aircraft of the flying wing configuration. Consequently, it is appropriate that some of the primary considerations, while well estblished for over fourty years, be revived as a background for the current discussions and deliberations.

The classical range equation (Coffin-Breguet, Ref 2), first presented over seventy years ago, may be used to establish that, in the case of a propellor driven aircraft, the all-wing aerodynamic configuration has desirable attributes for long-range flight. In this case, the rate of fuel consumption is taken to be proportional to the power produced, which must equal the power required, or

$$-\frac{dW}{dt}=C_p VT=C_p VD_t \tag{1}$$

where the propellor efficiency has been incorporated into the single constant, C_p, which is the rate of fuel consumption (pounds per hour per horsepower delivered to the air), V is flight velocity, T is the thrust produced and D_t is the total drag.

From this, which may be rewritten as

$$-\frac{dW}{ds}=C_p D_t \tag{2}$$

it follows that

$$-\frac{ds}{dW}=\frac{L/D_t}{C_p W} \tag{3}$$

which, upon integration, leads to the classic range equation.

Clearly, in the case of a propellor driven aircraft, the maximum range is achieved if the ratio of lift to drag is held at the maximum achievable value throughout the flight. Thus, it follows that the optimal aircraft for long range flight is that one for which all non-lift producing sources of drag have been eliminated. The attractiveness of the flying wing is evident.

Jet aircraft, however, are observed to display a different pattern of fuel consumption. The idealization described by Equation 1 is normally replaced by another, i.e., that the rate of fuel consumption depends only on the thrust produced, or

$$-\frac{dW}{dt}=C_J T \tag{4}$$

We again replace the time increment by the ratio of incremental distance to velocity, yielding:

$$-\frac{ds}{dW}=\frac{V/D_t}{C_J} \tag{5}$$

Let the coefficients of lift and drag be defined in the usual way,

$$L=\frac{C_L\rho V^2 A}{2} \tag{6}$$

and

$$D_t=\frac{C_D\rho V^2 A}{2} \tag{7}$$

where the total drag coefficient is written as

$$C_D=C_{do}+kC_L^2+\frac{C_f A_f}{A} \tag{8}$$

Here C_f is the fuselage parasite drag coefficient, A_f is the wetted ar ea of the fuselage, and A is the planform area of the wing. C_{do} is the wing parasite drag coefficient, and the coefficient k will be taken to be inversely proportional to the product of π, wing aspect ratio, and a Munk efficiency factor, e. We then arrive at an expression for the incremental range obtained per incremental fuel consumption as

$$-\frac{ds}{dW}=\frac{\frac{2^{1/2}}{C_J}}{(\rho W)^{1/2}}\frac{(C_L/A)^{1/2}}{C_D} \tag{9}$$

Where C_D is given by Equation 8.

First, it is necessary to establish and maintain the value of C_L yielding the maximum value for range. This is found by differentiating the right side of Equation 9 and setting to zero. The result is:

$$3kC_L^2=C_{do}+\frac{C_f A_f}{A} \tag{10}$$

Upon substitution of this value into Equation 9, and dropping non-essential terms from the right, it is seen that the maximum range is achieved by an aircraft for which the quantity

$$-4C_J(\rho W/2)^{1/2}3^{-3/4}\frac{ds}{dW}=\frac{(1/k)^{1/4}(1/A)^{1/2}}{[C_{do}+C_f A_f/A]^{3/4}} \tag{11}$$

is. This quantity was the subject of the earlier explorations by Ashkenas (Ref 3) and by Foa (Ref 4).

The objective is these works, and here, is to establish the overall aerodynamic design, considering the division of the total volume, v_t of the vehicle between the volume of the wing, v_w and the volume of the fuselage, v_v. Here, the wing is taken to include all lifting components, and the fuselage, all non-lifting components. Following Foa, we let

$$x=\frac{v_t}{v_w},\, x-1=\frac{v_f}{v_w} \tag{12}$$

It is useful to consider familes of generic shapes. Following Ashkenas, we relate the surface area of the wing to its volume by

$$s_w=2A=k_w v_w^{2/3} \tag{13}$$

and surface area of the fuselage to its volume by

$$s_f=k_f v_f^{2/3} \tag{14}$$

Substitution of equations 13 and 14 into the right hand side of 11 yields the following to be maximized:

$$\Phi=\frac{(AR)^{1/4}k_w^{-1/2}x^{1/3}}{[1+\frac{2(x-1)^{2/3}}{B}]^{3/4}} \tag{15}$$

where

$$B=\frac{C_{do}k_w}{C_f k_f} \tag{16}$$

A non-essential constant, $\pi e^{1/4}\ C_{do}^{3/4}$ has been moved to the left hand side of Equation 11. Foa sought the extremal values of this equation, with findings as given in Ref 4. In so doing, he regarded the first two terms of the numerator as constants. In so far as the dependence of Φ on x within a generic class (constant B) is concerned, this is appropriate. However, when comparisons are made between aircraft described by different values of B, this is not appropriate, as those changes in design parameters which affect B (aspect ratio and wing thickness) also change the afore mentioned terms in the numerator. Accordingly, they are retained in the present analysis.

We must next establish the dependence of the parameter B on significant aircraft design variables. Following Ashkenas (Ref 3), three critical parameters are identified:

L/D, the ratio of fuselage length to diameter,

t/c, the ratio of wing thickness to chord, and

AR, the wing aspect ratio.

Generic fuselage and wing geometries are then defined in terms of these parameters, and from these, values of k_w and k_f are computed. If the wing parasite drag, per unit area, is taken equal to the fuselage parasite drag, per unit area, B may be written as:

$$B=\frac{1.39}{[(L/D)/AR]^{1/3}[t/c]^{2/3}} \tag{17}$$

The term, k_w, in the numerator of Equation 15 is also found to depend on these parameters, as follows:

$$k_w=\frac{3.04(AR)^{1/3}}{(t/c)^{2/3}} \tag{18}$$

II. RESULTS

Two figures follow. In Figure 1, a fuselage fineness (L/d) of 12 has been used to develop a dimensionless range measure for wings of relatively high aspect ratio (AR = 16, 12, and 8) with two values of wing thickness, t/c = 0.1 and 0.05. In these figures, x' = 1/x has been used as the abcissa. The flying-wing design then corresponds to x' = 1.0, while x' = .05 corresponds to an aircraft with 5% of total volume contained in the wing. For each family a maximum and a minimum, as predicted by Foa, are observed. As observed by Foa, the minimum value is for a configuration which is nearly all wing. In such a case the flying wing would indeed be the worst possible design.

In Figure 2, a directly comparable dimensionless range is given for wings of lower aspect ratio (AR = 8, 5 and 3) and greater thickness (t/c = 0.1 and 0.2). Here, a fuselage fineness ratio of 6 has been used. Note that results for one possible wing geometry (aspect ratio 8, thickness ratio .1) are given on both figures as a means of establishing (at least for one case) the relative influence of the fuselage fineness ratio, L/d. We now see very different trends from those of the previous figure. For thick wings of low aspect ratio, the range increases continuously as the design moves towards the all-wing limit.

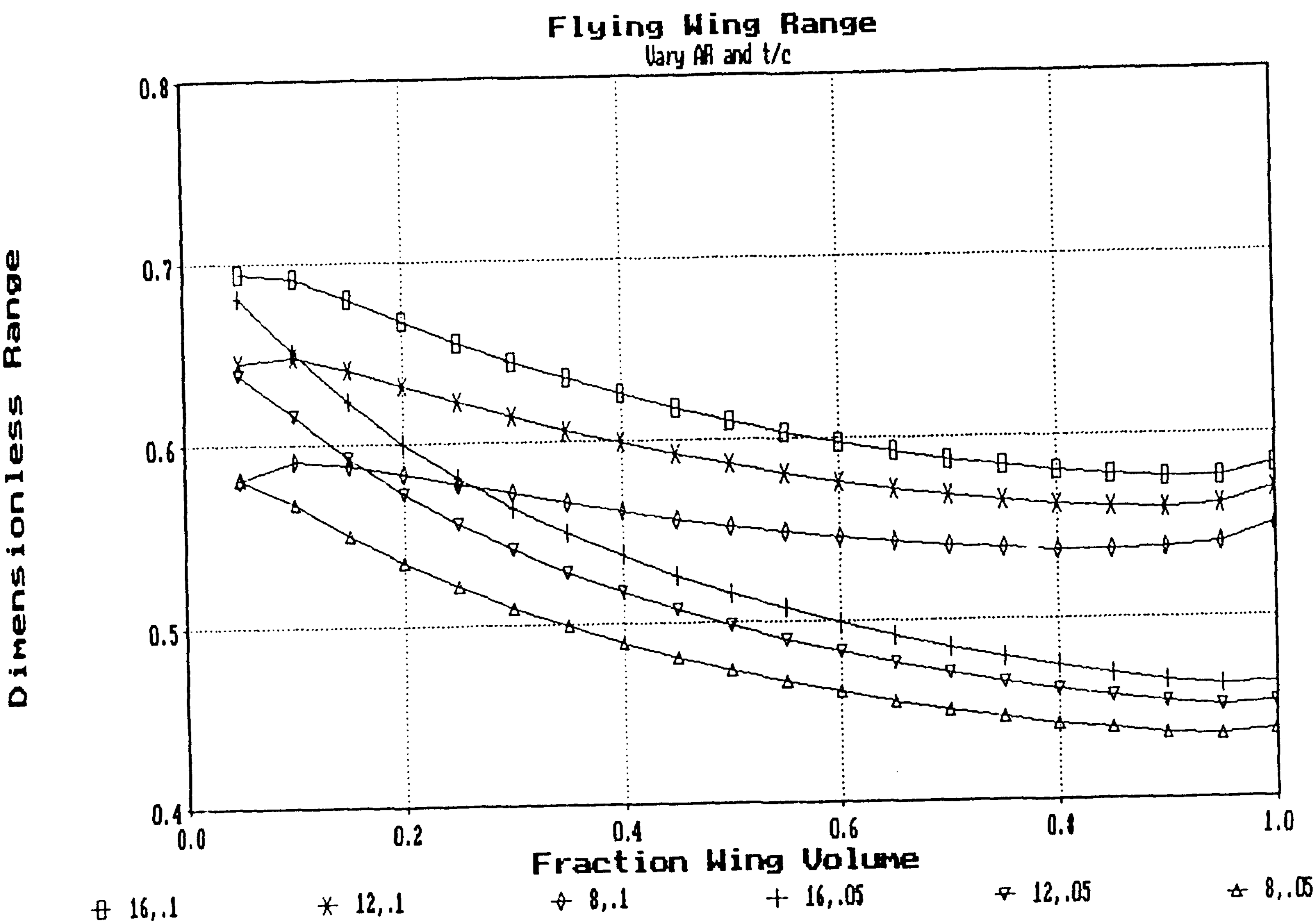

Figure 1 Maximum Range for Aircraft with Fuselage Fineness of 12 and Various Wings (AR,t/c)

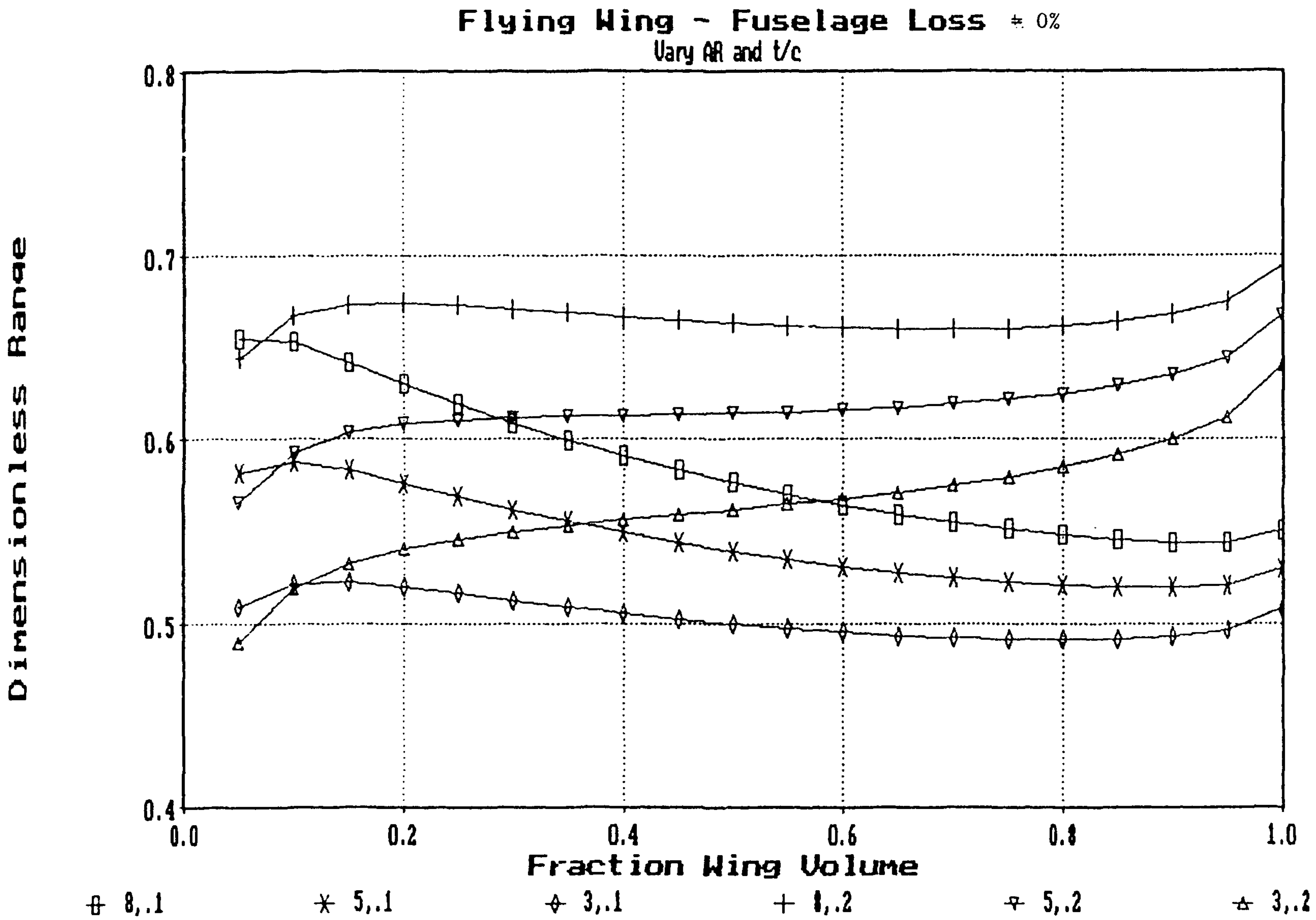

Figure 2 Maximum Range for Aircraft with Fuselage Fineness of 6 and Various Wings (AR, t/c)

The results serve to confirm the general conclusion reached by Ashkenas (Ref 3): that aircraft with thin wings are most efficient in the wing-body configuration, while aircraft with thick wings are most efficient in the all-wing configuration. It is especially interesting to note that (for the extreme parameters used here and within the scope of this simplified analysis) that comparable ranges can be achieved by either type. The results of Reference 3 are presented in a somewhat different manner and, although equivalent, may not make the point as clearly as do the presentations of Figures 1 and 2.

Foa properly identified the critical role of his parameter, B, which is directly related to the parameter used by Ashkenas (Ref 3).

$$B=2\frac{k_w}{k_n} \tag{19}$$

For a fuselage fineness ratio (L/d) of 8, values of the parameter B, as defined and used by Foa, are given in Table I, along with the parameters used by Ashkenas.

Table I
Summary of Wing Design Parameters

Aspect Ratio -AR:	16	12	8	16	12	8	5	3	8	5	3
Thickness -t/c:	.05	.05	.05	.1	.1	.1	.1	.1	.2	.2	.2
B (Eq. 17):	12.9	11.7	10.2	8.1	7.4	6.4	5.5	4.6	4.1	3.5	2.9
k_n/k_w Ref 3:	.16	.17	.20	.25	.27	.31	.36	.43	.49	.58	.68

The observations made by Foa (Ref 3) concerning the location of maxima and minima of the equation for range are confirmed by the present results. However, a focus on the maxima, and minima, may cause one to loose sight of the fact that there are design classes for which range is nearly independent of the volume ratio, and that, in those cases where there are no minima, the extremal value appears to be always at the all-wing configuration. Foa has noted that for aircraft designed to have low B, the all-wing design maximizes range.

The retention of all of the appropriate terms in the numerator of Equation 15 leads to values of Φ for low-B all-wing designs which are not only greater than wing-body designs for the same B, but which are also comparable to that for the best wing-body designs obtained at much higher values of B. Thus, based on a closer examination of the actual values of the dimensionless measure of range, we conclude that there is a significant potential for the use of all-wing configurations in the design of long-range aircraft.

Although the wing thickness parameter, t/c, does appear in these results, it should be noted that a second, and possibly significant dependence on this parameter has not been included: that being, the dependence of parasite drag on wing thickness. One evaluation of this effect (Ref 5) suggests that

$$C_{do}=a'+.0056+0.01\frac{t}{c}+0.1\frac{t^2}{c^2} \tag{20}$$

which, for the largest value of wing thickness used here, could double the wing parasitic drag. As this would serve to double the value of B, it would make all-wing configurations less desirable. In a similar vein, however, it must be noted that the assumption concerning equality of wing and fuselage parasitic drag may favor the wing-body configurations over the all-wing classes of design, as has been noted by Ashkenas (Ref. 6). Indeed, it must be noted many other highly significant effects, both aerodynamic and otherwise, have not been included in this analysis. Nor have such operational considerations as desired speeds. Clearly, for the same payload (taken as equivalent to total volume in these analyses), an all wing design will have lower wing loading, lower landing speed, and lower cruise speed.

Only aerodynamic considerations have been considered, as was the case in the analyses of Foa (Ref 4) and Ashkenas (Ref 3). Foa has also raised other issues, adverse to the all-wing design, which merit attention. (Ref. 7) But there are other considerations which may be expected to favor the all-wing families of design. One of these is structural efficiency, of which it has been suggested that the distribution of both lift and weight in a generally comparable manner over the entire vehicle should lead to a more efficient, or lighter, structure. Sears (Ref. 8) has discussed this, along with some issues of stability and control.

III. ACTUAL ENGINE CHARACTERISTICS

Another issue is that the use of engines of the high by-pass type may be expected to show fuel consumption trends intermediate to those assumed in Equations 1 and 4. As a consequence, one should expect that optimum designs for aircraft so powered would be driven somewhat towards the optimal aerodynamic shape for a propellor driven aircraft, that is, the flying-wing.

Shown in Figure 3 are fuel consumption characteristics for several turbojet engines. In all cases, sea-level fuel consumptions in lbm/hr/lbf are given as a function of flight Mach number. The J79 and J57 are low-bypass engines, the by-pass ratios of the TF41 and TF30 are .73, and that for the TF34 is 4 to 6. Data are from Ref. 9. The data is quite well described by linear relaltionships, with non-zero intercepts, or alternatively, by a family of curves:

$$C_M=C_{Mo}V^{\alpha} \tag{21}$$

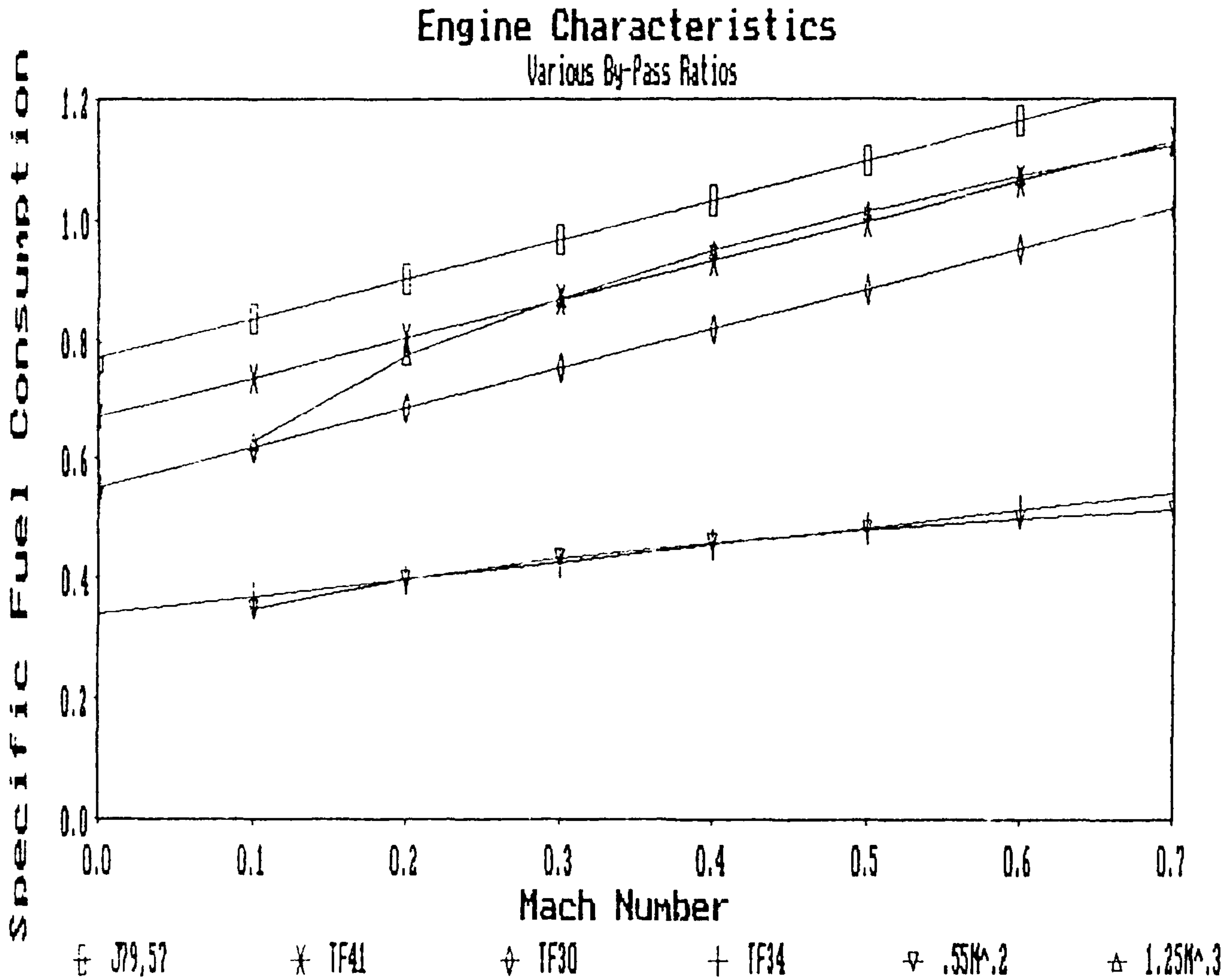

Figure 3 Specific Fuel Consumption for Various Engines

Two such relationships are shown in Figure 3. For values of flight Mach number of posible interest in a long range aircraft, it is interesting to note that a fractional-power dependence on flight velocity captures quite well the actual behavior of a range of engines. Consequently, it is of interest to revise the classical analysis, to propulsion systems with characteristics hypothesized in Equation 21.

Thus, in place of Equation 4, we use

$$-\frac{dW}{dt}=C_{Mo}TV^{\alpha} \tag{22}$$

Equations 6, 7 and 8 are unchanged, but Equation 9 becomes

$$-\frac{ds}{dW}=\frac{\left(\frac{2}{\rho A}\right)^{\frac{1-\alpha}{2}}\left(\frac{C_L}{W}\right)^{\frac{1+\alpha}{2}}}{C_{Mo}C_D} \tag{23}$$

The value of C_L which provides maximum range is found by substituting Equation 8 into Equation 23 and setting to zero the derivative. The result is

$$kC_L^2=(1+\alpha)\frac{C_{Do}+\frac{C_f A_f}{A}}{3-\alpha} \tag{24}$$

and leads to the following expression from which the range may be determined.

$$-C_{Mo}\frac{4}{3-\alpha}\left(\frac{\rho}{2}\right)^{\frac{1-\alpha}{2}}\left(\frac{3-\alpha}{1+\alpha}\right)^{\frac{1+\alpha}{4}}W^{\frac{1+\alpha}{2}}\frac{ds}{dW}$$
$$=\frac{A^{\frac{\alpha-1}{2}}(1/k)^{\frac{1+\alpha}{4}}}{\left(C_{Do}+\frac{C_f A_f}{A}\right)^{\frac{3-\alpha}{4}}} \tag{25}$$

Maximizing the right hand side will now identify the design for maximum range by establishing the division of aircraft volume between the wing fraction, 1/x, and fuselage fraction, 1 - 1/x. After substitution of Equations 13 and 14 and moving non-essential terms to the left leaves the function to be maximized as:

$$\Phi=\frac{k_w^{\frac{\alpha-1}{2}}(AR)^{\frac{1+\alpha}{4}}x^{\frac{1-\alpha}{3}}}{\left[1+\frac{2}{B}(x-1)^{\frac{2}{3}}\right]^{\frac{3-\alpha}{4}}} \tag{26}$$

with B as defined by Equation 16. For α = 0, this reduces to Equation 15.

It is appropriate to make one additional modification to the original analysis. In the determination of the shape parameter, k_w, for the wing, it was assumed that the wing was of rectangular planform. It is easy to show that a swept wing of constant chord yields the same volume for the same span and thickness distribution. However, for a wing with varying chord, Equation 18 must be adjusted. For example, it is found for a wing of triangular planform with t/c constant along the span that the value of k_w as computed above must be reduced by a factor

$$s=(3/4)^{\frac{2}{3}}=0.8255 \tag{27}$$

Accordingly, for a triangular planform of the same aspect ratio (AR) and the same value of t/c, the parameter B as appearing in Equation 15, or in Equation 26, must be reduced by this factor.

In a similar manner, it is possible to develop a correction factor appropriate to wings of trapezoidal planform. The result is:

$$s=\frac{(\frac{3}{4})^{\frac{2}{3}}[1+\frac{C_t}{C_r}]^{\frac{4}{3}}}{[1+\frac{C_t}{C_r}+(\frac{C_t}{C_r})^2]^{\frac{2}{3}}} \tag{28}$$

where C_r and C_t are root and tip chord, respectively. Note that Equation 28 reduces to Equation 27 when $c_t = 0$, and to unity when tip and root chords are the same. For $c_t/c_r = .5$, $s = (27/28)^{2/3}$.

Given in Figures 4 and 5 are some results obtained from Equation 26 for various aircraft with propulsion systems characterized by $\alpha = .50$ and wings of triangular planform. All results are for a fuselage fineness (L/D) = 8. It is noted that the range of design variables AR and t/c for which all-wing aircraft produce the greatest range has been expanded significantly over the results given in Figure 1 and 2.

Values of x which lead to extremal values of Equation 26 may again be identified through the procedure employed by Foa on Equation 15. Let

$$(x-1)=Z^3 \tag{29}$$

Extremal values are found to occur for the roots of

$$(1+\alpha)Z^3-(1-\alpha)BZ+(3-\alpha)=0 \tag{30}$$

or

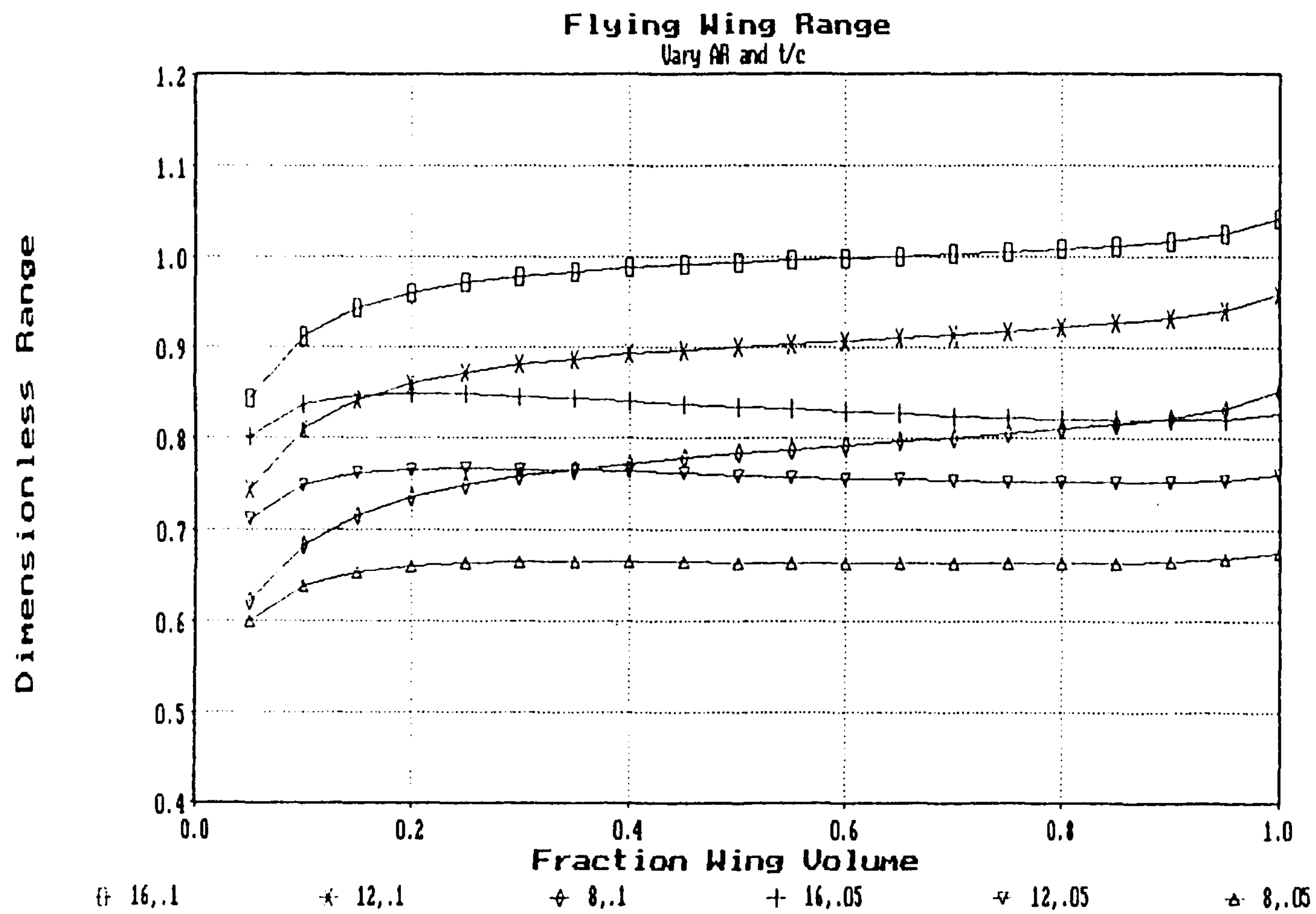

Figure 4. Dimensionless Range for Aircraft with Triangular Wings. s=.5; L/D = 8

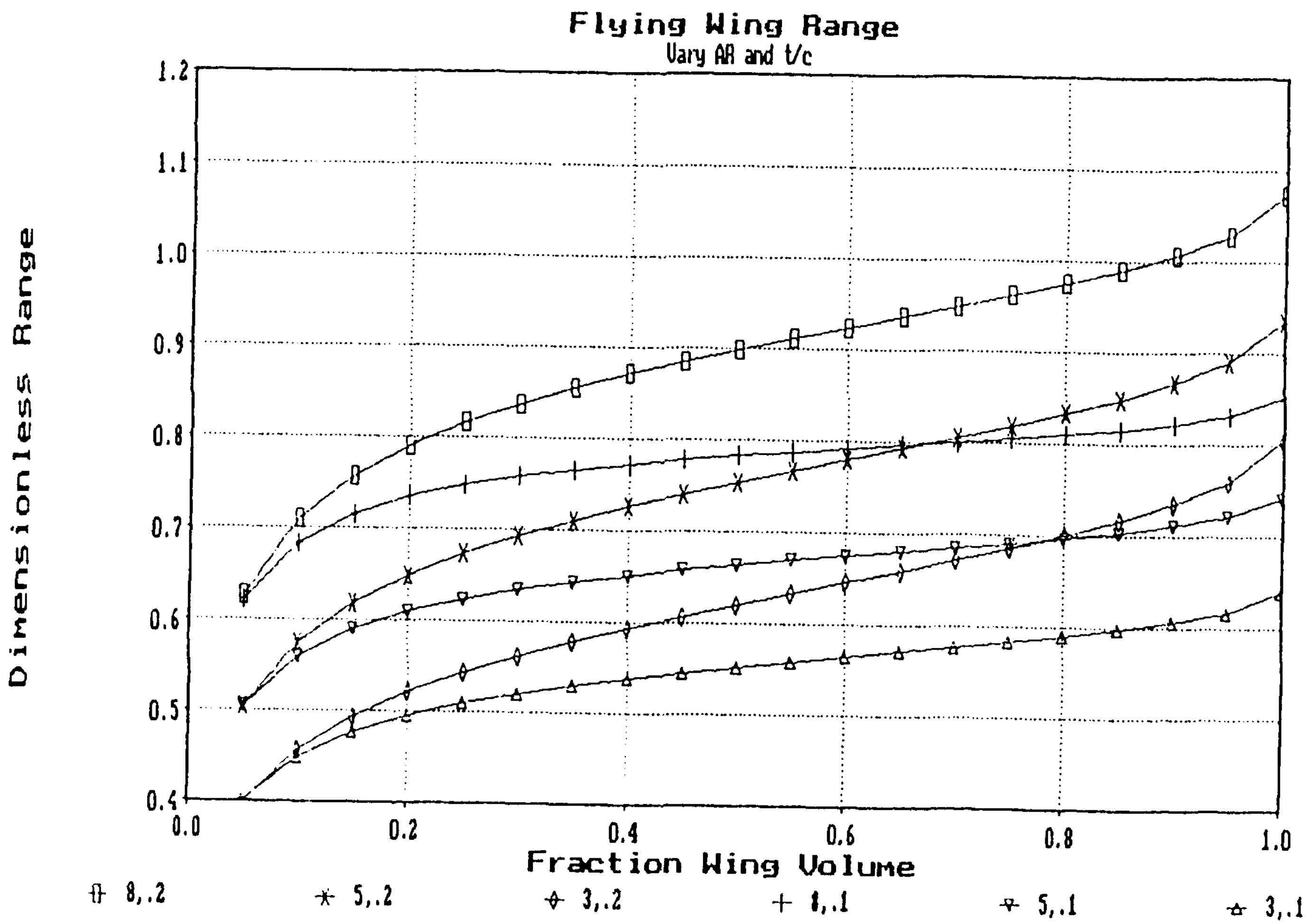

Figure 5 Dimensionless Range for Aircraft with Triangular Wings. s=.5; L/D = 8

$$Z_i=2\left[\frac{B(1-\alpha)}{3(1+\alpha)}\right]^{\frac{1}{2}}\cos\left(\frac{p+2\pi i}{3}\right) \tag{31}$$

where

$$\cos p=-0.5\frac{(3-\alpha)(1+\alpha)^{\frac{1}{2}}}{\left[\frac{B(1-\alpha)}{3}\right]^{\frac{3}{2}}} \tag{32}$$

The two positive roots coalesce into one for

$$B_c=3\frac{1+\alpha}{1-\alpha}\left[0.5\frac{3-\alpha}{1+\alpha}\right]^{\frac{2}{3}} \tag{33}$$

and the minimum and maximum of Φ with respect to x disappear for lesser values of B. Thus, the all wing-configuration is optimal for values of B below B_c (and for some range above), where the critical value depends on the propulsion parameter α.

For $B_c < B < B_{mw}$, the value $\Phi(x=1)$ is greater than the stationary value of Φ, and the all-wing design shows superior range. For values of B above B_{mw} wing-body designs have superior range. Values of B_c and B_{mw} are depicted graphically as Figure 6 for a range of values of the propulsion system parameter, α. This figure also demonstrates clearly the role of the propulsion system parameter, α, in expanding the range of designs, as characterized by the parameter B, for which the all-wing designs are optimal. In interpreting these results, it is to be noted that, for non-rectangular planforms, the value of B found from Equation 17 must be reduced by the appropriate factor s.

Comparison of the values of B as tabulated in Table I with the values given on the figure show that a substantial range of vehicles exists for which the flying wing is optimal, especially, in the case of triangular planforms. Further examples are given in Figure 7. Here the fuselage fineness ratio is held at 6, a propulsion system characterized by α = 0.25 is assumed, and wings of aspect ratios of 7 and 4 are considered for thicknesses of 0.07, 0.1, and 0.15. The superiority of the all wing configuration, in all but the case of the thinnest wing combined with the larger aspect ratio, is self-evident. Included here are two cases where extremal values occur, producing a worst-case which is near to the all-wing configuration, but for which the all-wing design remains optimal.

IV. SUMMARY AND CONCLUSIONS

In summary, the findings of Foa concerning the non-optimality of the all-wing configuration for long range flight have been reviewed. The results confirm Foa's findings for the case of an ideal turbojet

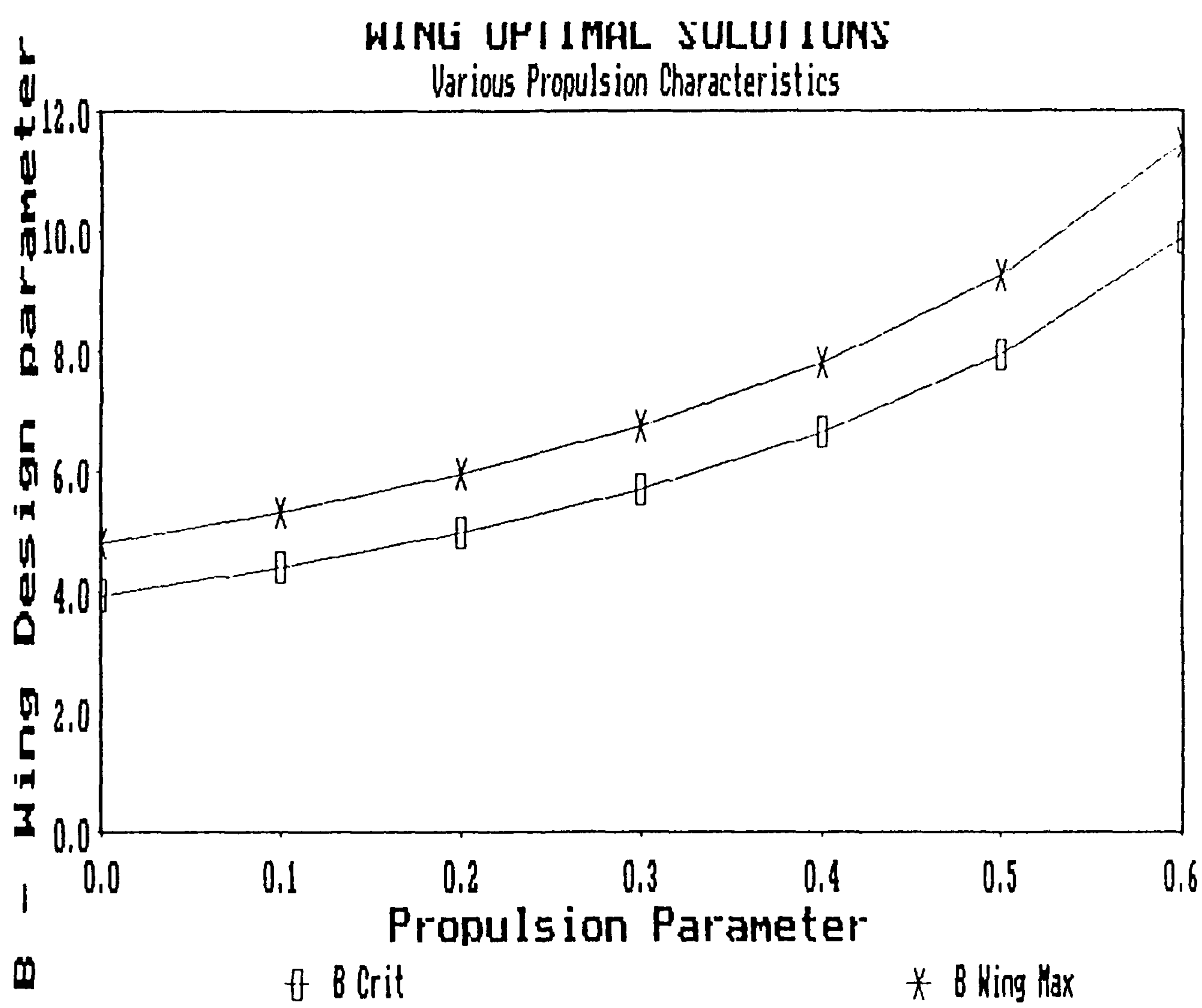

Figure 6 Influence of Propulsion Parameter on Wing Design Parameter

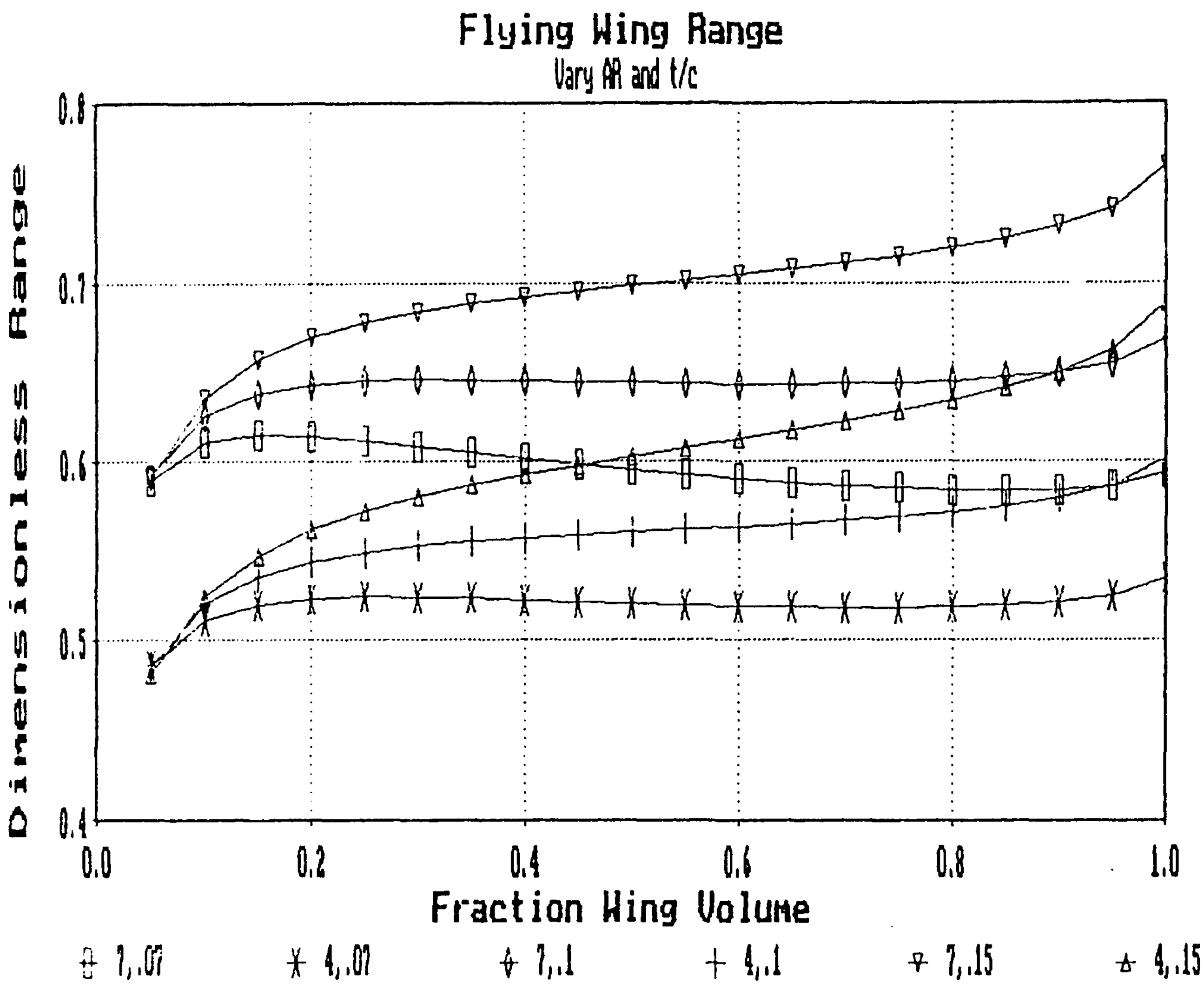

Figure 7 Range for Aircraft with Triangular Wings, s=.25 and L/D = 8

propulsion system coupled with thin, high-aspect ratio wings, especially of rectangular plan-form. However, the inclusion of two more realistic assumptions: (1) the dependence of specific fuel consumption on flight velocity, as is actually observed for jet engines, especially of the high-bypass type, and (2) the adjustment for wings of a triangular planform, leads to a remarkable expansion of the range of design parameters for which the all-wing configuration leads to the optimal design for long-range flight.

It must be noted that this study includes only aerodynamic considerations, and that these are far from complete. Further, the selection of the optimum design has not been constrained by any consideration of the resulting cruising speed, stall speed, stability characteristics, or structural design problems. Needless to say, each and all of these could serve to make totally inappropriate a design selected totally on the basis of an incomplete consideration of a highly idealized aerodynamic model.

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