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# THE INDUCED DRAG FACTOR OF HIGHLY SWEPT DELTA WINGS AT SUBSONIC SPEEDS

BY

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SUMMARY

The induced drag factor for delta wings with approximately 60 degrees of leading edge sweepback is found to be affected greatly by the aerofoil section shape. For thin sections (e.g. 5 percent) or thick sections with sharp noses the induced drag factor varies between 0.40 and 0.50. For thick sections (e.g. 10 percent) values of 0.8 and 0.9

* The information contained in this report is taken from an unpublished report written in April, 1952.

can be obtained. The above results apply for Reynolds' Numbers of 4 million or greater.

On the basis of limited data it appears that the induced drag factor for thin sectioned wings does not change over the subsonic Mach Number range.

# TABLE OF CONTENTS

## LIST OF ILLUSTRATIONS



## List of Symbols

e Induced drag factor (or span efficiency factor)

$$= \frac{1}{\pi A} \frac{d C_L^2}{d C_D}$$

A Wing aspect ratio

# THE INDUCED DRAG FACTOR OF HIGHLY SWEPT DELTA WINGS AT SUBSONIC SPEEDS

## 1.0 INTRODUCTION

It has recently been observed that several subsonic performance calculations on aircraft fitted with delta wings having leading edge sweepback of the order of 60 degrees have been based on assumed span efficiency factors of 0.80 to 0.85.

The above values of e can be obtained with conventional straight-wing aircraft, but there are indications of an adverse sweepback effect on e (e.g. Ref. 1). However, the data collections available on this matter contain little information for delta planforms.

In view of this it was considered worthwhile to make a collection, from the available literature, of span efficiency factor data for delta wings with approximately 60 degrees of leading edge sweepback. (The values of leading edge sweepback considered herein vary between 58 and 63 degrees.)

In this survey an attempt was made to find data on the independent effects of Reynolds' Number and high subsonic Mach Numbers.

## 2.0 EXPERIMENTAL DATA

### 2.1 Low Speed Data

The experimental data collected for delta wings with leading edge sweepback angles in the range 58 to 63 degrees are plotted in Figures 1 to 3. The data were taken from References 2 to 16, and apply only to "unclipped" (i.e. zero taper ratio) delta wings.

In calculating the value of e from the various references, small scale graphs usually had to be read (as opposed to tabulated data) and therefore some error and scatter has inevitably occurred. Also, the curve of $C_L^2$ against $C_D$ for a given wing tends to lose its linearity or maintains it over a smaller range of $C_L$ as the Reynolds' Number is reduced. This becomes particularly pronounced in the Reynolds' Number region below four million.

The values of e in the present report were calculated from the drag data for the lift coefficient range from 0 to 0.4.

The data in Figure 1 correspond to aerofoil sections having a non-zero leading edge radius and a thickness chord ratio of 0.10 or greater. (The data for the DM-1 glider will be discussed in a following section.)

Above a Reynolds' Number of 4 million, e reaches values approaching 0.90, whereas below this value of Reynolds' Number e varies between 0.55 and 0.85.

It can be noticed that the wings with larger thickness chord ratios tend to have higher values of e, and this is especially true at the low Reynolds' Numbers. The value of e for the 58 degree delta wing appears much lower than a wing of the same thickness ratio. This may be due to a Reynolds' Number effect associated with the particular aerofoil section, or it might be due in part to the fact that the chordwise position of the maximum thickness is farther back than for the other wings, hence resulting in a smaller nose radius. It will be shown in a following section that nose radius has an important effect on e.

Figure 2 contains values of e for delta wings with aerofoil sections having either a very small nose radius (approaching zero) and/or a small thickness to chord ratio (around 0.05).

The majority of the data in this figure are grouped around a value of e equal to 0.47. The data for thin sections at low subsonic speeds in Figure 3 are in the range covered by the majority of the data in Figure 2. Hence there appears to be no Reynolds' Number effect on e for delta wings with thin aerofoil sections in the Reynolds' Number range of $3 \times 10^6$ to $28 \times 10^6$. This latter statement is made mainly on the basis of sharp nosed aerofoils and for plane (i.e. untwisted and uncambered) wings.

Two of the points in Figure 2 do not follow the general trend. One of these is for a flat plate aerofoil which has an e value of 0.372. This is probably due to the reduced thickness ratio and nose radius. The other point is for a wing-body-fin combination with a modified N.A.C.A. 0005 section which has an e value of 0.575. This higher value might be due to a larger leading edge radius (a radius of 0.275 percent chord) than that associated with the other sections.

However, the data for a similar section in Figure 3 fall in line with the majority of the data in Figure 2. The data in Figure 3 are at a lower Reynolds' Number and hence there might conceivably be a Reynolds' Number effect.

It is not too clear how the data in the Ames full scale tunnel (where the majority of the data in Figure 2 and for the point in question were obtained), were corrected for strut tare, interference and alignment; different procedures could cause variations in e.

No mention has been made of the fact that some of the data are for wings alone and some for wing-body combinations because, as can be seen from the data in Figures 2 and 3, the addition of a fuselage does not change e from the wing alone value.

## 2.2 Data at High Subsonic Speeds

Figures 3 and 4 contain the available data on e at high subsonic speeds. The data in Figure 3 are deduced from wind tunnel tests and those in Figure 4 from a free-flight test (Ref. 17).

For plane wings, Figure 3 shows the improvement in e caused by an increased thickness chord ratio (or nose radius); however for the 8 percent thick section e is reduced as the Mach Number increases. Mach Number has little effect on the thin sections (3 and 5 percent thick) and, if anything, e increases with Mach Number (the same trend is seen in Fig. 4). The e values in Figure 3 for "twisted and cambered" wings are substantially better. These wings are designed to have a uniform loading at a Mach Number of 1.5.

The free-flight data in Figure 4 agree well with the data of Figure 2 at the lower Mach Numbers, although the aerofoil thickness ratio is higher (6.5 percent) than that for the majority of the wings in Figure 2.

## 3.0 DISCUSSION

Some observed characteristics of the flow about delta wings help to interpret the collection of induced drag data presented herein.

During the tests described in References 6 and 18 a "separation vortex" was observed to spring from the leading edge when the wing was tested at incidence. A schematic diagram illustrating this flow is given in Figure 5. This type of flow has been found to be associated with thin aerofoils. Also during wind tunnel tests on a 1/15-scale model of the DM-1 glider (15 percent section) this type of flow was observed. However, it was not observed during tests on the full-scale glider (Ref. 4). Placing a metal strip over a portion of the centre span produced this type of flow on the full-scale glider. This resulted in a reduction of e (see Fig. 1) and also increased considerably the maximum lift coefficient. (A correlation between $C_{L_{max}}$ and section shape is made in an unpublished report and it is found in general that wings with aerofoil sections producing a separation vortex have higher values of $C_{L_{max}}$.)

Assuming that a delta wing of the planform discussed herein can be treated by slender wing theory (e.g. Jones' theory) then the separation vortex is a function of the transverse section of the wing (see Fig. 5). In view of this it can be clearly seen that a sharp nosed section will cause a separation of the transverse flow which, combined with the longitudinal component, produces the separation vortex. Rounding the leading edge of the wing will tend to reduce the severity of the adverse pressure gradient through which the transverse flow must travel.

Alternatively it can be stated that the separation of the boundary layer in the region of the nose of the wing will depend on the pressure gradient along the streamline in this region. The pressure distribution in this region is a function of all the geometrical section parameters; however, for a given thickness ratio the tendency for leading edge separation at incidence will increase as the nose radius is reduced and for a given finite nose radius the tendency for separation will increase as thickness ratio is reduced.

The value of e calculated from the potential flow spanwise loading is close to unity as the loading for such wings is approaching elliptic loading. Therefore the low values of e found for thin or sharp nosed delta wings are associated with viscous phenomena and, keeping in mind the above discussion of the separation vortex, a study of Figures 1 and 2 leads to the conclusion that this is responsible for the variation of e with section shape.

It is found with conventional aerofoil sections that laminar separations near the leading edge tend to occur at incidence when the Reynolds' Number is low. It was no doubt such reasons that caused the separation vortex to form for the 1/15-scale model of the DM-1 glider and is also responsible for the reduced values of e at low Reynolds' Numbers for delta wings with conventional aerofoil sections (Fig. 1).

For aircraft performance estimation the e for the complete aircraft must be estimated. The data herein indicate that the e for a wing-body combination is approximately the same as the e for the wing alone. The effect of a horizontal tail surface on e depends on the downwash field at the tailplane and the aircraft centre of gravity position.

Unorthodox aircraft configurations such as the DM-1 glider can have values of e much less than that for the wing alone. In this case adding a large vertical tail (housing the pilot at the root) caused a considerable disturbance to the flow over the wing (as observed by wool tufts) and not only reduced e from 0.91 to 0.65 but reduced the maximum lift by an appreciable amount.

## 4.0 CONCLUSIONS

A literature survey pertaining to the span efficiency factor, e, of delta wings with leading edge sweepback of 60 degrees or thereabouts and with no camber and twist, has led to the following conclusions:

(1) At Reynolds' Numbers below about $4 \times 10^6$, a leading edge separation vortex (Fig. 5) forms at incidence regardless of the aerofoil section. This separation vortex is probably of stronger intensity for thin and/or sharp nosed sections.

(2) At Reynolds' Numbers above $4 \times 10^6$ a separation vortex is found only for thin (e.g. 5 percent) and/or sharp nosed sections.

(3) Wings that exhibit a separation vortex have lower values of e than wings without such a flow, indicating that leading edge separations are responsible for the low values of e.

(4) At Reynolds' Numbers from 4 to 9 million (the limits of the available data) the values of e for thick sectioned wings (e.g. 10 percent) range between 0.8 and 0.9.

(5) At Reynolds' Numbers of 3 to 28 million (the limits of the available data) the values of e for thin sectioned and/or sharp nosed wings range between 0.4 and 0.5.

(6) Increasing the Mach Number up to the drag rise, does not appreciably change the value of e for thin sectioned wings (i.e. 3 to 5 percent). Tests on a wing with an 8 percent section showed a decrease of e with an increase in Mach Number.

(7) For thin sectioned wings, at any rate, adding a fuselage (and vertical tail) does not appreciably change e from the wing alone value. However, unorthodox arrangements (Fig. 2) may cause large changes in e.

The above conclusions refer to plane wings; however, some tests on twisted and cambered wings with 3 and 5 percent sections (Fig. 3) show that an appreciable gain in e can be obtained by this means.

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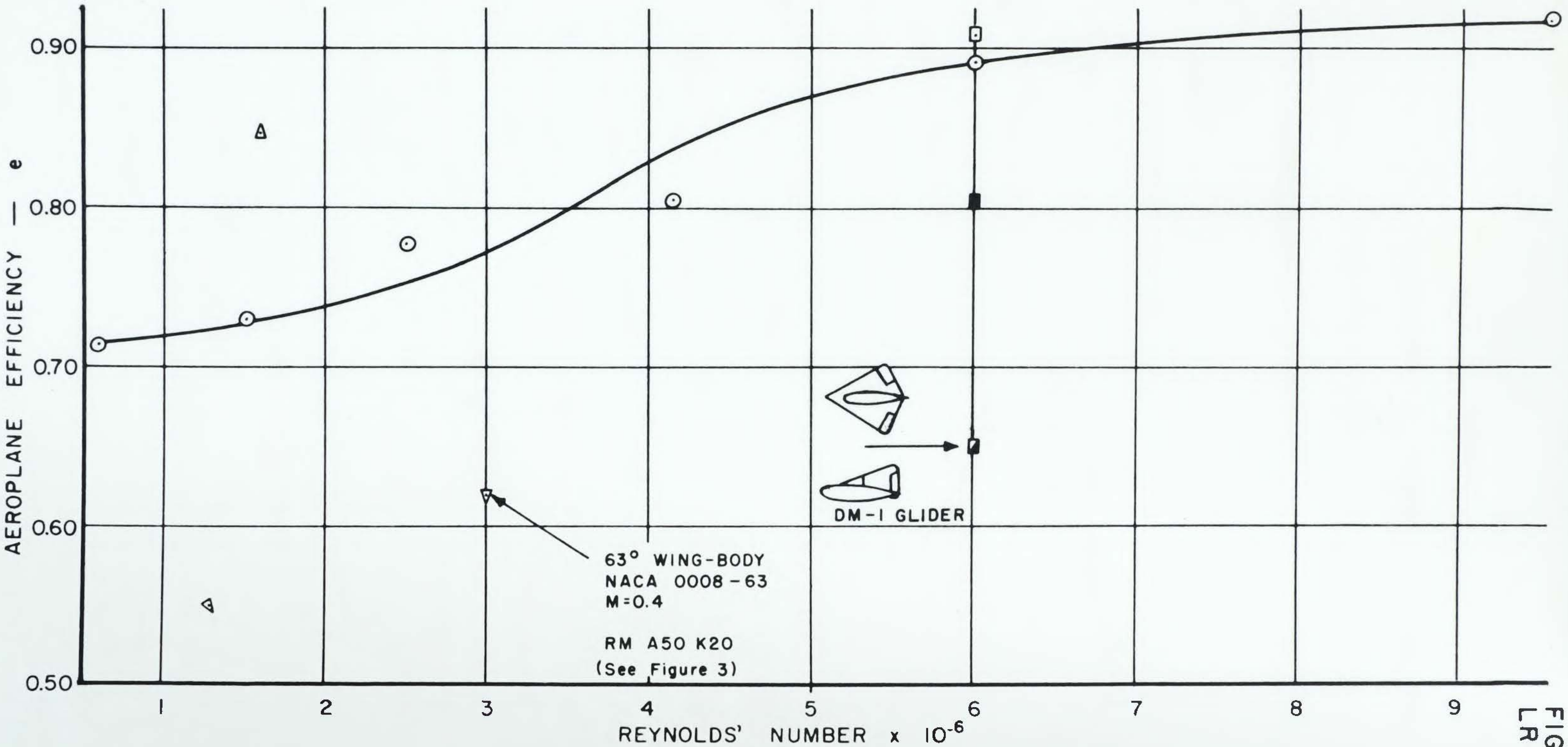
| SYMBOL | REPORT | ASPECT RATIO | SWEEPBACK | AEROFOIL |
|---|---|---|---|---|
| ○ | ACR 11,354 | 2.31 | 60° | t/c = 0.10 |
| □ | NACA RM L6K20 | 1.8 | 60° | NACA 0015-64 |
| ■ | SAME AS □ EXCEPT FOR SHARP LEADING EDGE | | | |
| ◪ | SAME AS □ EXCEPT FOR VERTICAL FIN | | | |
| Δ | NACA TM 1176 | 2.0 | 63° | NACA 0012 |
| ◁ | KTH AERO NO.4 | 2.50 | 58° | FFA 104-5106 (t/c = 0.10) |

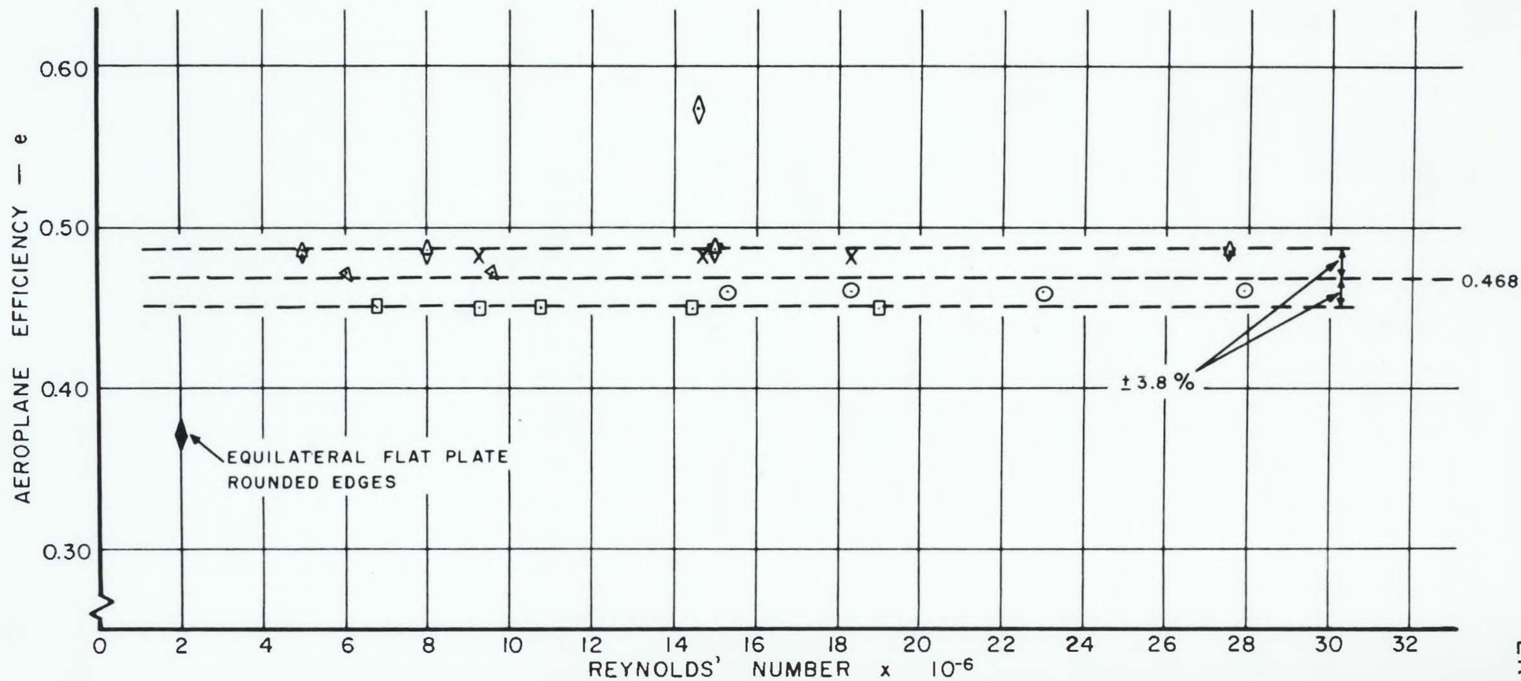



AEROPLANE EFFICIENCY AGAINST REYNOLDS' NUMBER ($M \leq 0.13$)

FIG. 1
LR-105

Re BASED ON MAC, MACH NO. $\leq$ 0.18, SWEEPBACK (L.E.) = 63°, AR = 2.0

| SYMBOL | NACA REPORT | MODEL |
|---|---|---|
| ○ | RM A7F06 | SYM., SHARP-EDGED, DOUBLE WEDGE AEROFOIL SECTION, t/c = 0.05 (NO BODY) |
| □ | RM A7H28 | SAME WING AS ABOVE EXCEPT MAXIMUM THICKNESS AND L.E. ROUNDED (NO BODY) |
| △ | RM A7K05 | DOUBLE WEDGE AEROFOIL, t/c = 0.05 |
| ▽ | RM A7K05 | DOUBLE WEDGE AEROFOIL, t/c = 0.05 (PLUS BODY) |
| X | RM A7H28 | SAME WING AS □ EXCEPT 0.011C RAD. NORMAL TO L.E. INSTEAD OF 0.0025C CHORDWISE |
| ◊ | RM A51B21 | MOD. NACA 0005 WING PLUS BODY AND VERTICAL FIN |
| ◁ | RM L8G05 | BICONVEX AEROFOIL, t/c = 0.10 |

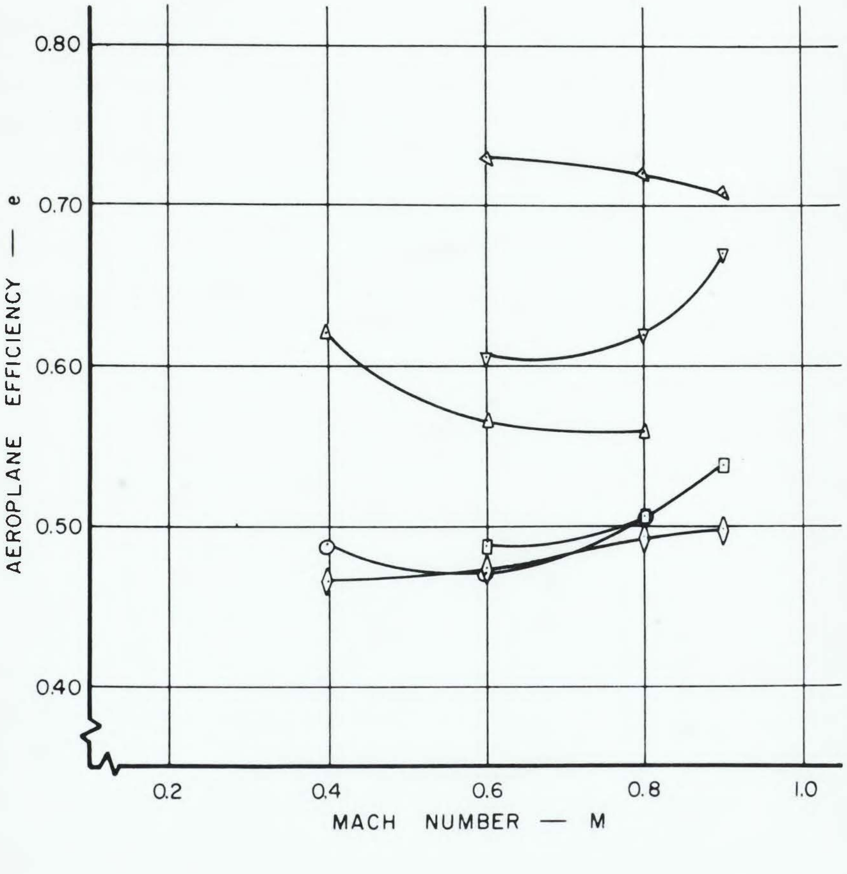



AEROPLANE EFFICIENCY AGAINST REYNOLDS' NUMBER

FIG. 2
LR-105

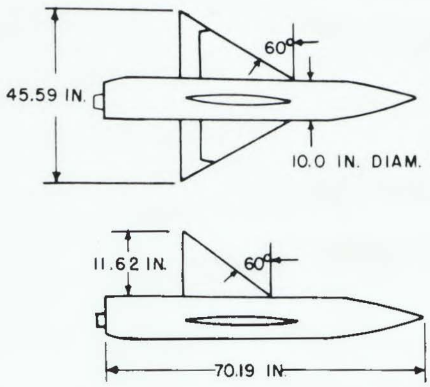
FIG. 3
LR-105

| SYMBOL | Re x $10^{-6}$ | AEROFOIL |
|---|---|---|
| □ | 5.0 | NACA 0003-63 |
| ○ | 3.0 | NACA 0005-63 |
| △ | 3.0 | NACA 0008-63 |
| ▽ | 3.0 | NACA 0003-63 CAMBERED AND TWISTED |
| ◁ | 3.0 | NACA 0005-63 CAMBERED AND TWISTED |
| ◇ | 5.3 | DOUBLE-WEDGE (MAX. t/c = 0.05 AT 0.20 C) |

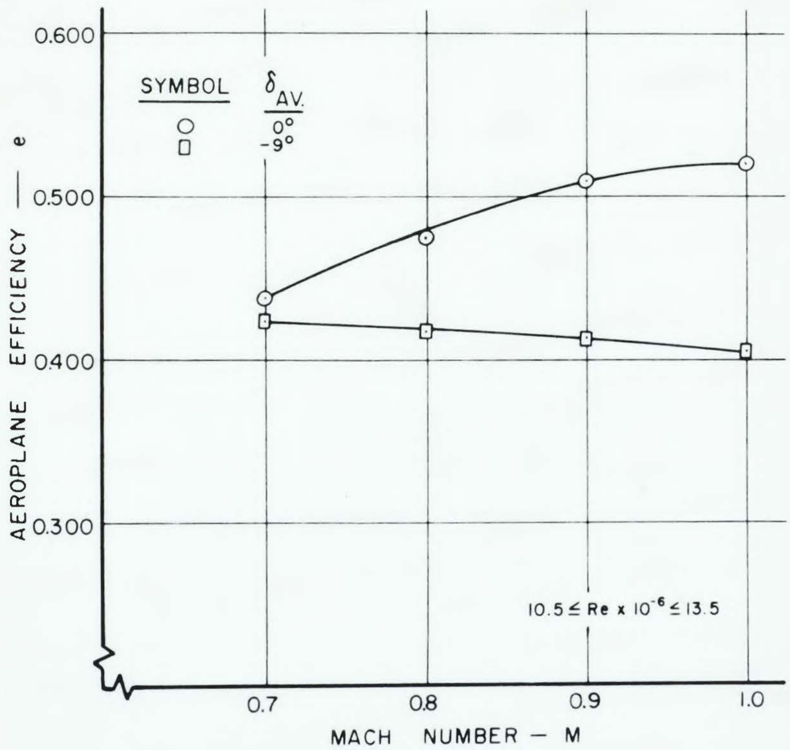



AEROPLANE EFFICIENCY AGAINST MACH NUMBER

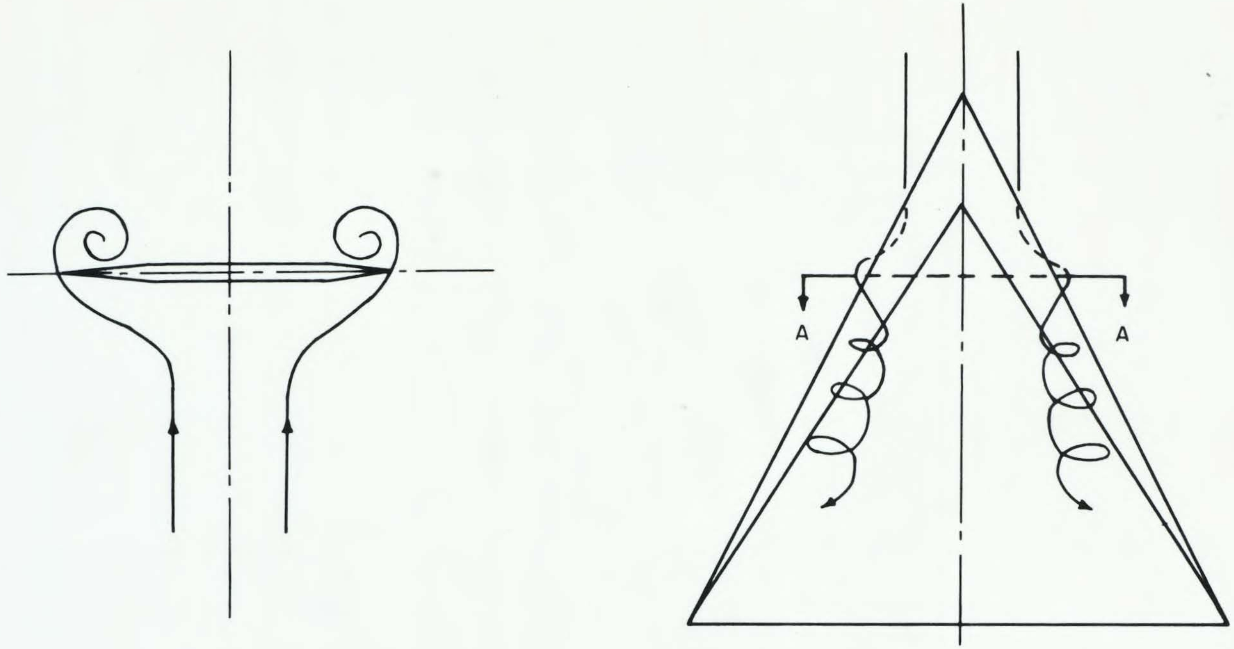
ALL MODELS CONSIST OF A DELTA WING PLUS BODY
ASPECT RATIO = 2.0, i.e. SWEEPBACK = 63.3°




AEROFOIL SECTION: NACA $65_{(06)}$A006.5




VARIATION OF AEROPLANE EFFICIENCY WITH MACH NUMBER




(a)
FLOW ABOUT THE TRANSVERSE SECTION A-A

(b)
RESULTANT FLOW OVER DELTA WING

SCHEMATIC DIAGRAM OF THE FLOW OVER A SHARP NOSED DELTA WING

FIG. 5
LR-105

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THE INDUCED DRAG FACTOR OF HIGHLY SWEPT DELTA WINGS AT SUBSONIC SPEEDS.
P.J. Pocock and J.R. Westell. July 1954.
13 p. + 5 figs. (Lab. Rept. LR-105)

The induced drag factor for delta wings with approximately 60 degrees of leading edge sweepback is found to be affected greatly by the aerofoil section shape. For thin sections (e.g. 5 percent) or thick sections with sharp noses the induced drag factor varies between 0.40 and 0.50. For thick sections (e.g. 10 percent) values of 0.8 and 0.9 can be obtained. The above results apply for Reynolds' Numbers of 4 million or greater.

On the basis of limited data it appears that the induced drag factor for thin sectioned wings does not change over the subsonic Mach Number range.

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1. Wings, Delta - Drag
2. Wings, Delta - Aerodynamic characteristics

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