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FEBRUARY 1964 AIAA JOURNAL VOL 2, NO 2

Mach 8 to 22 Studies of Flow Separations Due to Deflected Control Surfaces

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This paper describes measurements of separation phenomena obtained during a hypersonic aerodynamic test program conducted in the Boeing Company 44-in Hotshot wind tunnel and in the Arnold Engineering Development Center tunnel "B" Test results establish the influence on separation parameters exerted by systematic variations of model geometry and Mach and Reynolds numbers Pressure and heat-transfer measurements are correlated with Schlieren photography, temperature-sensitive paint, and oil-streak flow visualization data Applications of theories are described

Nomenclature

C_f = skin-friction coefficient
C_P = pressure coefficient based on P_0, $(P - P_0)/q_0$
$f_3(1)$ = 1 47, Ref 16
K_3 = correction function to heating rate due to pressure gradient, Ref 10
l_i = interaction length, Ref 16
M = Mach number
n = exponent in expression $P \sim X^n$
P = pressure
S_t = Stanton number based on stagnation enthalpy
T = temperature
u = streamwise velocity
X = streamwise coordinate
y = normal coordinate
δ = turning angle from freestream direction
ξ = correction factor for linear flow, Ref 16

Subscripts

BL = boundary layer
F = flap
FP = undisturbed flat plate
n = normal to spoiler
0 = beginning of interaction
p = plateau
s = separation point
W = wall
∞ = freestream

Presented as Preprint 63 173 at the AIAA Summer Meeting Los Angeles, Calif, June 17–20, 1963; revision received October 23, 1963 The pressure and heat-transfer tests being conducted in the Hotshot wind tunnel are part of an internal Boeing aero dynamic research program The Mach 8 results presented were obtained during the course of a joint Air Force Aeronautical Systems Division and Boeing investigation of aerodynamic interference effects Boeing personnel who have contributed significantly to this program are the Boeing wind tunnel design and operations group E J Sears, and Richard McKenzie The assistance of G Alexander, W Hankey, and J Ondrejka of the Aeronautical Systems Division and the unknown Air Force personnel who expedited the clearance for publication of the Mach 8 data is gratefully acknowledged

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Introduction

FLOW separation effects are not completely understood at low Mach numbers, but the body of empirical knowledge and the theoretical understanding that have been developed can generally prevent, or quickly solve, flight problems resulting from separation This knowledge of separation phenomena from subsonic through low supersonic speeds was acquired progressively over a period of approximately 40 to 50 years The extension of wind tunnel data to flight regimes involved conventional wind tunnel scaling techniques because wind tunnels simulated flight conditions reasonably well

Approximately only 10 years have elapsed from the advent of routine supersonic to orbital flight Thus, in huge steps, flight progress has advanced far beyond present capability to predict confidently the effects of the high Mach numbers and low densities of present flight environments The effects of separation on a vehicle in this flight environment are possibly of even greater importance than at lower speeds The review by Kaufman et al [1] furnishes considerable information on many aspects of these problems

To the very critical problem of hypersonic control effectiveness might be added substantial lift and drag changes and highly nonlinear hinge moments Very high heating is encountered in the reattachment region, and the high temperature of aircraft surfaces can be expected to affect the extent of the separation region

This situation puts a high premium on acquisition of hypersonic test data The limited availability of facilities, the limited simulation capability, and the high cost of hypersonic testing emphasize the requirements for improvement of theoretical methods To date, the degree of applicability of lower-speed theoretical methods is untested because of the scarcity of high-speed data

The purpose of this paper is to present a portion of the results of a current Boeing research program on hypersonic aerodynamics A considerable body of hypersonic pressure, heat-transfer, and flow visualization data is presented for flow separations at Mach numbers ranging from 8 to 22 and unit Reynolds numbers of as low as $0\ 9 \times 10^5$ to 30×10^5 The results of systematic variations of model geometry, such as flap angle and aspect ratio, are also described

Some insight into more complicated model geometries is possible with the data of this report For example, because of viscous interaction effects, the Mach 16 flat-plate pressure distributions are similar in form to those of practical airfoil sections at large Reynolds numbers Also, the effects of low angles of attack, not covered herein, are reasonably accounted for by using edge rather than freestream conditions in appropriate analyses All known transitional data have been excluded so that pure laminar results can be described

The data are evaluated in a manner that shows the influence of the significant variables It is hoped that this can provide a basis for the understanding of other wind tunnel tests, especially hypersonic control effectiveness studies using conventional force balances Correlations of separation measurements with causative flow phenomena are presented, and the requirements for further testing are discussed

Separation Test Program

As an adjunct to studies of hypersonic inlet design and shock-boundary-layer interference, the Boeing Company is conducting a systematic series of wind tunnel investigations to establish basic separation criteria in order to define the efficiencies, effectiveness, and power requirements of various control surfaces Data applicable to trailing edge controls were obtained with the Hotshot wind tunnel models shown in Fig 1 These models permitted aspect ratio variations from 0 57 to infinity (exclusive of flap) and leading-edge-to-hinge-line chord lengths of 8 to 17 in The trailing edge flap had a chord of 8 in and could be moved through a wide range of angles

Model instrumentation consisted of variable reluctance differential pressure transducers and thermistor heat-transfer gages Flow visualization was obtained with temperature-sensitive paints, oil streaks, and Schlieren photographs The basic flat plate, instrumentation, and recording equipment were the same as those described in Ref 2, except that spanwise instrumentation coverage was reduced in favor of greater streamwise coverage The pressure gages were placed along zigzag rows to minimize possible coupling of small self-induced pressure disturbances

Models for the Mach 8 continuous-flow tests were geometrically identical to the Hotshot models but were of the conventional, stainless-steel, thin-skin, thermocouple, transient-heat-transfer type Separate thick-skin pressure and flow visualization models were also constructed

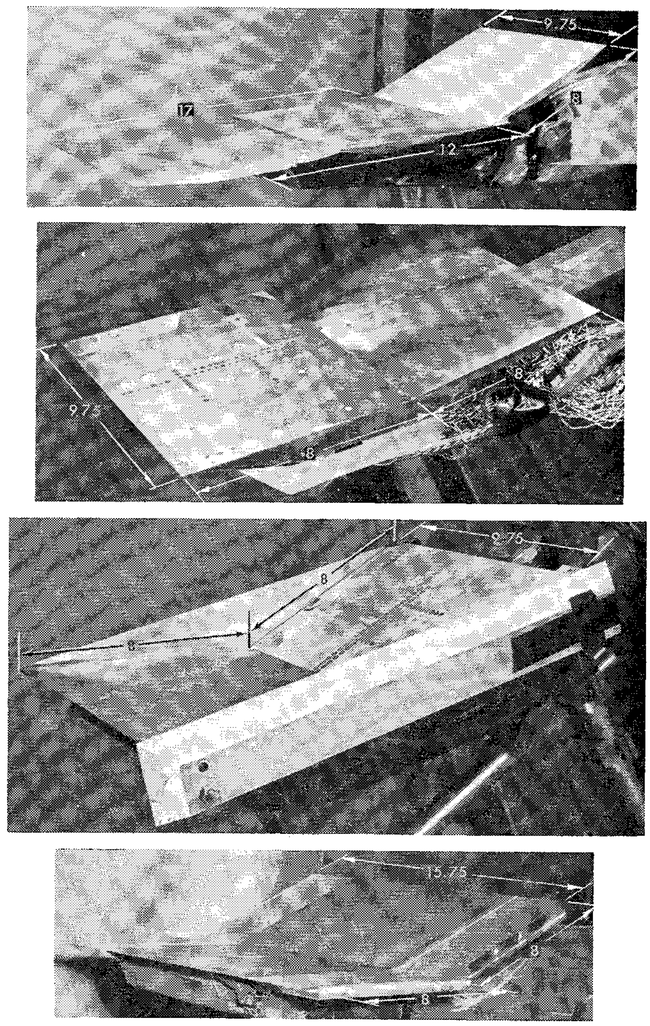

Fig 1 Trailing edge flap models

The tests conducted in the Boeing 44-in Hotshot wind tunnel covered the Mach number range from 14–22 and unit Reynolds numbers ranging from $0\ 4$ to 12×10^5/ft A comprehensive description of this facility is contained in Ref 2 and its bibliography The Mach 8 data were obtained at Arnold Engineering Development Center tunnel "B" at unit Reynolds numbers of 7×10^5 to 30×10^5/ft This facility is described in Ref 3

Discussion of Results: Model Geometry

This discussion of separation phenomena concentrates on describing the influence of configuration and flow variables on detailed pressure and heat-transfer measurements in the separated regions Figure 2 presents Mach 16 test data for a laminar separation caused by a deflected trailing edge flap The relation of pressure and heat transfer to the flow phenomena defined in Fig 2 is typical of all laminar separations observed in the wind tunnel test series being reported

Interpretation of Schlieren Photographs

The significant features of the schlieren photograph of Fig 2a are copied in the line drawing of Fig 2b Both separation and reattachment events are described

The shock off the sharp leading edge of the instrumented plate is defined in the schlieren as the outermost dark line, standing about 0 9 in off the plate surface at the point where the separation shock becomes visible at $X = 5\ 7$ in The boundary-layer edge can be discerned as a light line about midway between the leading edge shock and the plate surface This light line becomes more prominent downstream

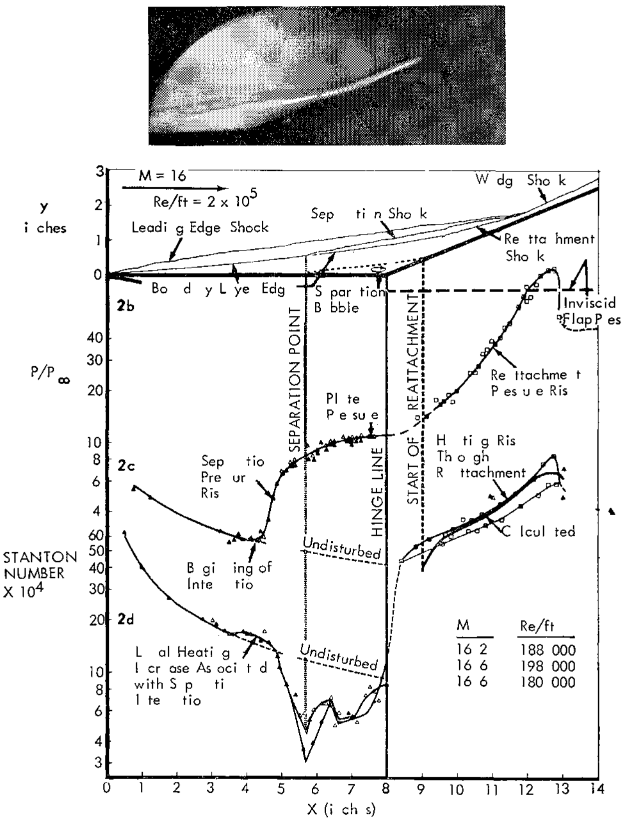

Fig 2 Hypersonic separation phenomena: a) schlieren photograph, b) line drawing, c) pressure, d) heat transfer

of the origin of the separation shock where it is deflected through a compression angle of 7 5° to 8° About $1\frac{1}{2}$ in downstream of the hinge line the reattachment shock becomes visible, and somewhat further downstream the leading edge shock intersects and coalesces with the separation shock This new shock, in turn, intersects and coalesces with the wedge shock off the deflected flap Schlieren resolution is inadequate for defining details such as the vortex streets and weak shocks or expansions known to emanate from intersections of shocks of the same family However, the thin dark line originating at the wall at $X = 12$ 5 in is believed to be a reflection of the pressure waves emanating from one of the forementioned shock intersections

Pressure Measurements

The pressure measurements of Fig 2c are quite similar in form to those noted in supersonic tests and also to the hypersonic data of Sterrett and Emery[4] and Bogdonoff and Vas [5, 6] It will be shown that the separation region pressure data correlate closely with the schlieren observations just described and also with the heat-transfer phenomena that will be discussed Results presented were obtained during three runs that vary slightly in Mach and Reynolds number because of the inability of the Hotshot tunnel to duplicate exactly energy release conditions from run to run The resulting slight differences in pressure and heat-transfer measurements (approximately 6%) are effectively removed by correcting the data to account for the differences due to viscous interaction effects As can be noted, the data follow the distribution for the undisturbed flat plate (obtained from several runs at similar test conditions) until an abrupt rise in pressure occurs $4\frac{1}{2}$ in from the leading edge and somewhat upstream of the separation shock This pressure rise continues to a "knee" from which the increase continues at a lesser slope until the constant pressure laminar "plateau" is reached The measured plateau pressure is in close agreement with the inviscid pressure rise estimated for the $7\frac{1}{2}$° to 8° boundary-layer turning angle noted in the schlieren photograph

The plateau pressure is maintained up to and somewhat beyond the hinge line, where the pressure once again rises This pressure rise results from the reattachment of the boundary layer, which begins approximately 1 in downstream of the hinge line The final value of this rise exceeds the inviscid flap or wedge pressure in the region where the leading edge separation shock intersects the reattachment or wedge shock It is not known to what extent these intersecting shocks affect observed overpressures Although the peak has been reached, the pressure adjustment to final values was not completed on the densely instrumented portion of the flap It was found that, if the initial abrupt pressure rise and the reattachment pressure rise are expressed as $p \sim X^n$, then $n = 6$ to 8 for the former and 3 to 4 for the latter case

Heat-Transfer Measurements

The detailed aerodynamic heat-transfer distributions of Fig 2d provide considerable insight into separation phenomena when viewed in the light of the preceding discussion of pressure and flow phenomena It can be seen that, near the leading edge, the Stanton numbers are in agreement with equivalent flat-plate measurements obtained during previous tests [2] A slight but definite increase above the undisturbed distribution occurs somewhat upstream of the abrupt pressure rise that precedes actual separation and persists until a sudden and rapid falloff to a minimum value The streamwise location of this heat transfer minimum is very nearly coincident with the origin of the separation shock (and the sudden deflection of the boundary layer) and the "knee" of the pressure curve The hypothesis that these events locate the separation point is based on the extension of supersonic correlations[7, 8] of the initial appearance of the separation shock with the "knee" of the pressure curve and a zero value of skin friction which, when coupled with boundary-layer surveys, defines the separation point according to the condition that $\partial u/\partial y = 0$ at $y = 0$ A zero value of heat transfer is not expected because of differences between the wall temperature and the bulk temperature of the fluid in the separated bubble The no-shear condition of the separation point, as opposed to the shear effect of the boundary layer upstream of the separation and the shear effect of the reversed wall flow in the separated bubble, also seems to justify qualitatively the assumption that the separation point and a

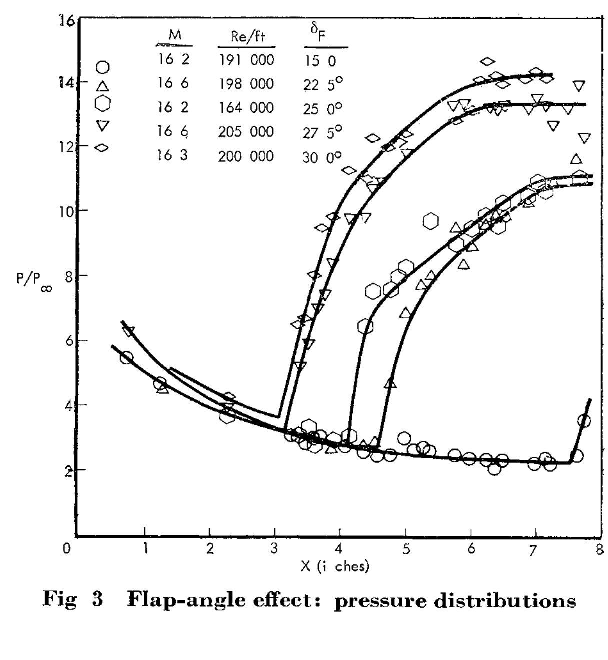

Fig 3 Flap-angle effect: pressure distributions

heat-transfer minimum should be coincident This contention has been substantiated by oil-flow visualization techniques in a limited number of cases

As can be noted, the rise from the minimum to the hinge line occurs in a "sinusoidal" manner The average amount of heat transferred to the separated region per unit time is approximately half of the unseparated flat-plate amount (for the cases shown in Fig 2d) The ratio of the amount of heat transferred in the separated region varies for the two runs in about the same way as the respective stagnation enthalpy ratio varies, although the data from the two runs, both heat transfer and pressure, are in excellent agreement upstream of the interaction The relation of these results to the analysis of Chapman[9] will be discussed for other configurations in a later section

The heating distribution on the deflected flap is similar in form to the flap pressure distribution, with a steady rise to a peak value that is in excess of the wedge solution for the flap An important finding of this program was the good correlation obtained between measured heating distributions on the flap and calculations based on using empirical pressures in the method of Bertram and Feller [10]

The pressure-gradient correction factor K_3 in the expression $St = \bar{S}tK_3(P_W/P_\infty)^{1/2}$ is presented in Ref 10 as a function of n, where n is the exponent in the pressure relation $P \sim X^n$ As noted earlier, n in the reattachment region varies between 3 and 4 The zero-pressure-gradient Stanton number $\bar{S}t$ is taken as the unseparated value just ahead of the hinge line

The increase in measured heating above the calculated, occurring at about $X = 12$ to 13 in, is possibly caused by pressure waves emanating from shock intersections in the same region The possibility that some of this increase, as well as the increase at $X = 3$ 5 to 4 75 in, is caused by a boundary-layer vortex system is discussed in the section on three-dimensional effects However, substantial and rapid changes in pressure, which could also conceivably cause significant heating increases, have been shown by Fitzgerald[11] to result from disturbances caused by similar intersecting shocks (of the same family) generated by a cone-cylinder-flare body at angle of attack

It is reasonable to expect that the influence of these pressure waves will depend on a number of factors affecting the

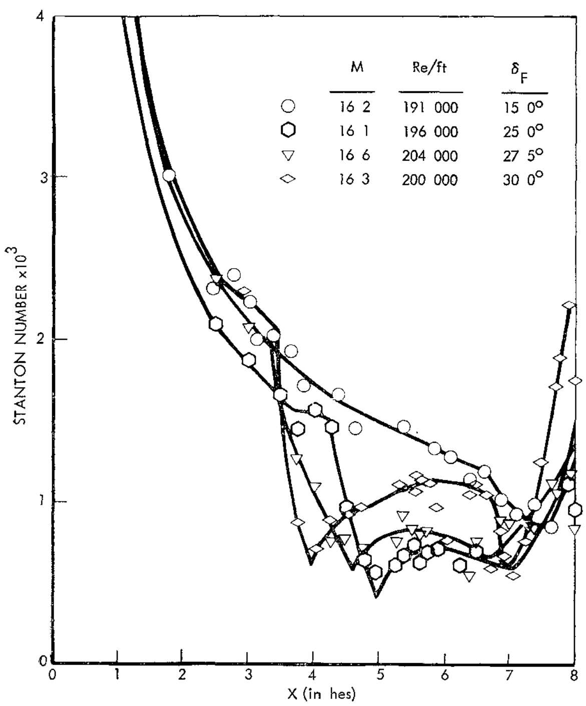

Fig 4 Flap-angle effect: heat-transfer distributions

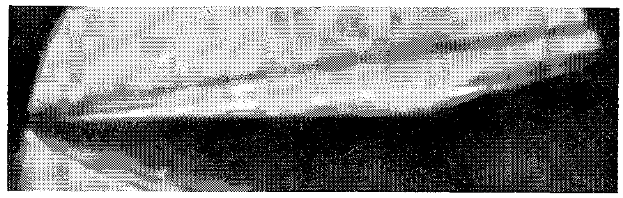

Fig 5 Schlieren photograph: 15° flap deflection

shock interaction, such as shock angle (Mach and Reynolds number effects in high-Mach, low-density flow), Mach number, forebody length, flap length, and flap angle These variables are also important to both basic separation phenomena and reattachment, and for this reason the information described will be based on data obtained with a constant-length flap The flap lengths chosen are sufficient to insure that the reattachment process is completed on the flap This cannot generally be done with flaps whose length is a practical percentage of the total wing chord The definition of flap length effects on separation and reattachment pressures and heat-transfer distributions is a fruitful area for future studies

Effect of Flap Angle

The beginning of the separation interaction and, consequently, the separated length and plateau pressure, is affected significantly if the flap-deflection angle is varied while holding other geometry and flow parameters constant These changes are illustrated in Fig 3, where the pressure distributions on the instrumented flat plate are plotted for flap angles of 15°, 22½°, 25°, 27½°, and 30° for a nominal Mach of 16 and unit Reynolds number of 200,000 ft The changes in pressure in, and in length of, the separated region result in substantial changes in the amount of heat transferred to the separated region, as can be seen from the Stanton number distributions of Fig 4

The correlation of significant pressure and heat-transfer events with schlieren data is consistent with the preceding discussion of data for the 22½° flap, 1 22 aspect ratio plate (Fig 2) For example, the knee in the pressure curve, the minimum in the heat-transfer distribution, and the origin of the separation shock (and the abrupt deflection in the boundary-layer edge) are all coincident for a given flap angle An exception to this occurs with the 15° flap angle As can be seen in the schlieren photograph of Fig 5, there is no separation shock or turning of the boundary layer evident until somewhat downstream of the hinge line There is a definite reduction in heat transfer and an indication of a pressure rise just upstream of the hinge line, however Examination of preliminary data for a 20° flap deflection also shows that the correlations of schlieren with pressure and heat-transfer data, which are successful for relatively large separated bubbles, fail when applied to cases where separation has not developed fully With the 20° flap the pressure definitely shows a rise to a plateau-like region, whereas the heat-transfer distribution is similar to that for the flap-angle range of 22½° to 30° The schlieren photograph for this run definitely fails to indicate any turning of the flow upstream of the hinge line However, with these exceptions, the empirical definition of the separation point through the use of pressure, heat transfer, and schlieren photography is consistent, as shown in Fig 6, where the separation criteria are compared for a large number of cases

Application of the visual criteria to force testing has proved valuable in analysis of control effectiveness experiments It is possible that apparently anomalous force and moment data may be satisfactorily explained through the observation that limited separation can occur without establishing an evident separation shock system

Further examination of the data in Fig 3 reveals that the ratio of the plateau pressure P_p to the pressure at the be-

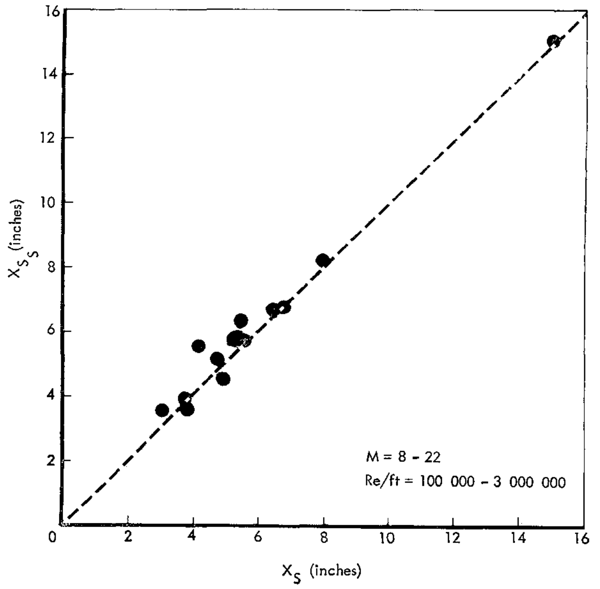

Fig 6 Consistency of separation-point criteria

ginning of interaction P_{X_0} has a constant value, approximately $P_p/P_{X_0} = 3\ 9$, for the conditions tested (with the exception of $\delta_F = 15°$) This pressure ratio is in good agreement with inviscid estimates based on the boundary-layer turning angles measured from schlieren photographs It is concluded that, although systematic variations in P_p/P_{X_0} and δ_{BL} might be expected to result from changes in δ_F, they are sufficiently small that the precision of the measurements is not adequate for their definition

During the course of these experiments, emphasis was not placed on the decreased heat transfer in the separated region However, preliminary analysis of test results reveals information that could be of value in understanding this type of aerodynamic heat-transfer process

Table 1 shows that the average heat transfer in the separated region ranges between 60 and 80% of the attached flat-plate values for the flap angles and Mach-Reynolds conditions considered Making an appropriate correction to the laminar flat-plate heat transfer, $St_{P_p} = St_{FP}(P_p/P_{FP})^{1/2}$, from unseparated flat-plate to separated plateau pressures results in the heat-transfer ratios shown in column 3 Chapman indicates in his analysis of Ref 9 that there is also a dependence on the square root of the Reynolds number, based on length of the separated bubble This is clearly shown to be the case in column 4, where the separation heat-transfer ratio is brought to a common reference length The ratios obtained are, at best, 60% of Chapman's estimated ratio of 0 56, but it must be recognized that they are based on a somewhat fictitious separated length, $X_{HL} - X_{sep}$, and that this could possibly account for some of the difference noted A more important conclusion is that the essentials of Chapman's theory can apparently be extended to very-high-speed flows

The question of the appropriateness of using the hinge line for a reference can be considered for the case of a very large deflection of a short-chord flap In this situation, use of the hinge line as a reference is justified Bogdonoff and Vas[5, 6] have shown separation region pressure distributions for a short flap which are similar to present data The heat-transfer data that they present for a $\delta_F = 10°$ flap also are similar in form to those shown in Figs 2 and 4 The separation they achieved with this relatively small flap angle can be attributed to Mach and Reynolds number effects as well as the possibility that helium vs air might also be a factor

Considering that short-span flaps can produce large regions of relatively high plateau pressure separated flow and that the initial portion of a deflected flap seems relatively ineffective, it appeared desirable to examine the pressure distributions resulting from the deployment of trailing edge spoilers A distinct advantage of spoiler-type controls is the elimination of the hinge moment changes due to separation and reattachment that occur on flap-type controls

Table 1

1 δ_F	2 $\bar{S}t_{sep}/\bar{S}t_{FP}$	3 $\bar{S}t_{s\ p}/\bar{S}t_{P_p}$	4 $(3) \times \left[\dfrac{(X_{HL} - X_s)_{\delta F=30}}{(X_{HL} - X\ )_{\delta_F}}\right]^{1/2}$
30	0 79	0 335	0 335
$27\frac{1}{2}$	0 66	0 303	0 325
25	0 61	0 285	0 327
$22\frac{1}{2}$	0 60	0 277	0 355

Effects of Spoiler Height

The data in Fig 7 present the pressure distributions for spoiler models with spoiler-height-to-chord ratios of $h/c = 0\ 0625$ and $0\ 094$ These spoiler heights range from just under to somewhat greater than a boundary-layer thickness The nominal conditions tested, Mach 16 and unit Reynolds number 200,000/ft, permit direct comparison with the preceding flap data Such a comparison reveals considerable similarity in the shape of the distribution and, further, shows that the ratio of plateau pressure to pressure at the start of interaction, P_p/P_{X_0}, is essentially the same for the spoilers and the ramps This is not surprising in that Chapman et al have shown such similarity in separation pressure data for ramp and step data at moderate supersonic speeds[7] It can also be seen that, based on the length of the interaction and the plateau pressure, the $h/c = 0\ 094$ spoiler produces a separation condition that is virtually identical to that of the 30° flap angle Similarly, the $h/c = 0\ 0625$ spoiler pressures are essentially equivalent to the $22\frac{1}{2}$ flap pressures

Up to this point, only the case of the unswept control surface has been considered Sweep of the trailing edge control is not only a possibility but likely, and, therefore, consideration of resulting three-dimensional effects definitely is required

Three-Dimensional Effects

Results of theoretical and experimental studies of swept separations in supersonic turbulent flows have been reported by Stalker[12] His conclusions were that significant pressures in the separation interaction can be readily related to two-

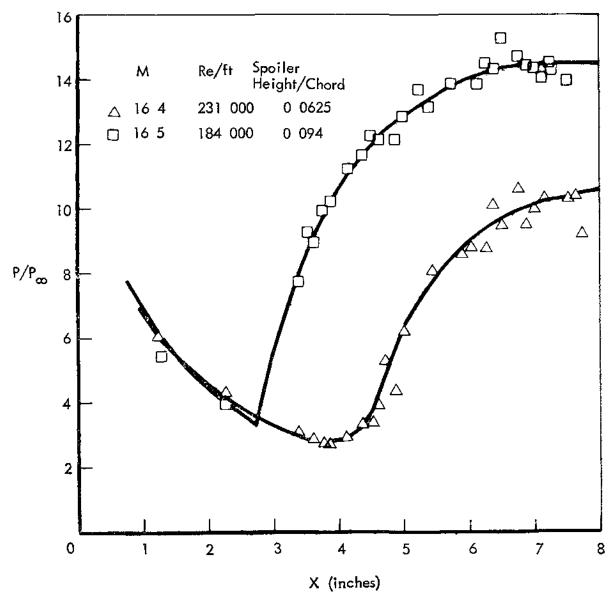

Fig 7 Spoiler height effect: pressure distribution

dimensional separations; roughly, the turbulent peak pressure depended only on the normal component of the Mach number The applicability of these conclusions to the laminar hypersonic conditions of this test is suggested by the data in Fig 8 Here the pressure distribution normal to a spoiler with $h/c = 0\ 0625$ and a sweep of 45° is compared with data normal to a similar unswept spoiler The decrease in the ratio P_p/P_{X_0} for the swept case, compared to the zero sweep spoiler, is approximately what would be expected for a decrease to the normal Mach number $M_n = 0\ 707\ M_\infty$ The similarity existing between ramp-like flaps and step-like spoilers, noted earlier, further suggests the application of Stalker's method to flaps whose hinge line is swept Reference 13 presents an analysis of this problem

Another three-dimensional boundary-layer effect observed in connection with separation and reattachment (on two-dimensional geometrical models) is believed to be caused by the presence of a system of boundary-layer vortices having a streamwise orientation of their longitudinal axes The existence of such a system is implied by striation patterns burnt into the surface of unswept hinge line flap models Use of scorch patterns to define the aerodynamic heating effects of such a vortex system is an analogical extension of supersonic sublimation and oil-flow techniques to high-enthalpy hypervelocity flows The patterns on the model of Fig 9 result from the reaction of a temperature-sensitive paint at a test condition of Mach 8 and unit Reynolds number of 2×10^6/ft Similar patterns were scorched into the stainless-steel surface of flap models during Hotshot wind tunnel tests

Occurrence of such vortex systems is of considerable interest because of the association of similar systems with transition of laminar boundary layers, i e , subsonic Taylor-Goertler vortices Hopkins[17] and Ginoux[18] have reported on the possible existence of supersonic boundary-layer vortices in the reattachment region downstream of an aft-facing step Their visual studies used sublimation and fluorescent oil techniques, supplemented by Ginoux' total pressure surveys, to define spacings and areas affected by these regularly spaced longitudinal perturbations The characteristics that they observed in the reattachment areas were similar to subsonic Taylor-Goertler vortices on concave walls, and it was hypothesized that flow curvature was sufficient to support such a system

The small radius flow curvature at separation and reattachment appears to satisfy the necessary condition of flow concavity in a manner similar to that noted for supersonic reattaching flows over an aft-facing step It is not known if the eventual fading of the striation patterns indicates dissipation of vortex action as normal flat-plate flow is re-established

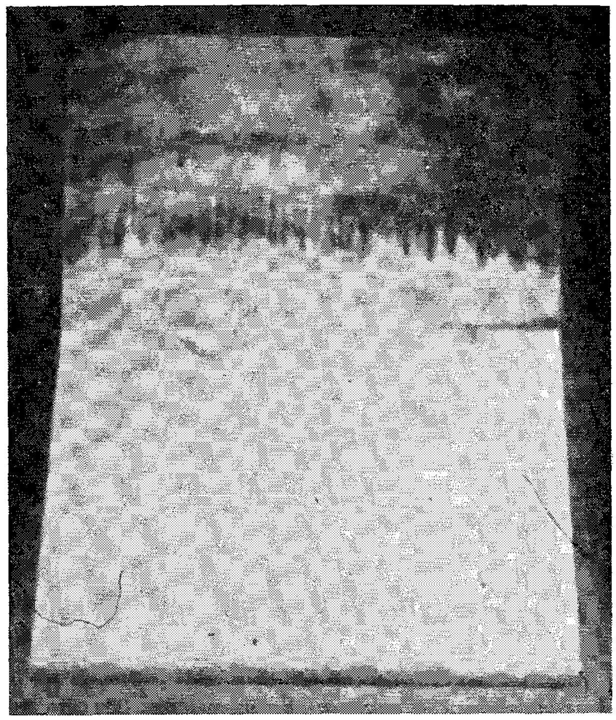

Fig 9 **Mach 8 reattachment heating patterns**

or if it merely indicates the threshold of material temperature sensitivity

Correlation of striation spacing with theory and estimates of the radius of concavity has not yet been possible because of the complexity of the interaction studied

It is concluded that aerodynamic heating evidence points to the existence of streamwise vortices in high-speed boundary layers at reattachment and that these vortices have characteristics similar to those noted at much lower speeds The existence of these vortices in the region of flow curvature at the start of the separation interaction could provide an explanation of the heating increase that precedes the initial pressure rise as described in Figs 2 and 4 The contribution of these vortices to the heating problem is not clear Eggers, for example, has shown that certain types of shed vortices could provide heating relief[13] In other cases it has been established that vortices are responsible for increases in aerodynamic heating[14] This is an area requiring further work because it is not inconceivable that these three-dimensional flow details could have an independent influence on separation characteristics as well as producing local heating increases

One other three-dimensional effect of considerable importance to these studies is the spanwise flow caused by pressure differences between the plateau and the tips Related to this are the tip disturbance patterns sweeping back from the leading edge corners These clearly generate an effect in the reattachment region, as can be seen from the curvature, near the tips or edges of the flap, of the reattachment heating patterns in Fig 9 Curvatures have also been observed in the separation region, as can be seen in the Mach 8 flap oil flows of Fig 10 Scorch patterns, indicating a curved separation line, have also been observed in Mach 16 Hotshot tests These curvatures imply the possibility of tip or edge effects that significantly influence separation on low-aspect-ratio wings

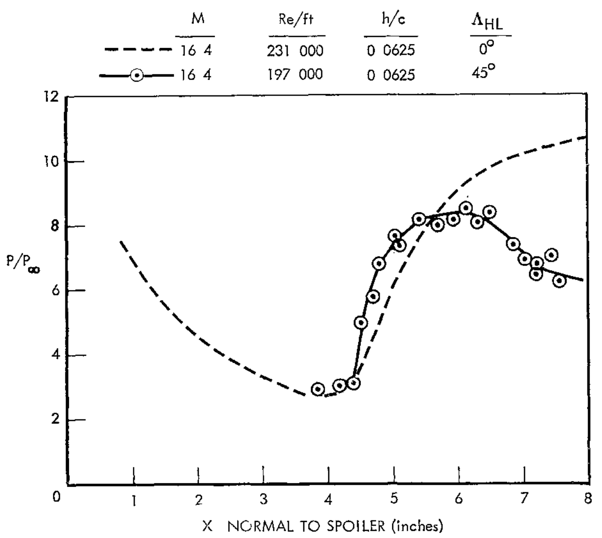

Fig 8 **Spoiler sweep effect**

Effects of Aspect Ratio

The models of Fig 1 define the range of aspect ratios studied during this investigation All the data discussed previously were obtained with the 1 22 aspect-ratio rectangular wing model The plateau pressure and separation-point

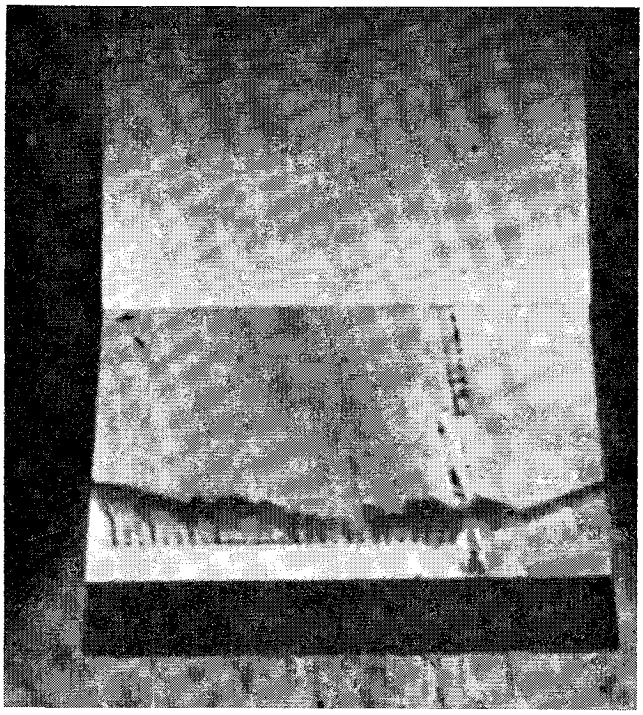

Fig 10 Mach 8 oil-streak definition of separation line

location changes resulting from an increase to an aspect ratio of 1 97 (X_{HL} = 8 in) were not substantial The use of tip plates on the 8-in hinge line model, the classical method of obtaining an infinite aspect ratio, did result in substantial forward movement of the separation point as well as an increase in plateau pressure Canning and DeRose[15] obtained a 20% increase in rolling-moment coefficient when they added similar tip plates to their Mach 5 trailing edge spoiler free-flight model However, data in Ref 2 show substantial pressure and heat-transfer effects in the corner formed by two perpendicular flat plates aligned with the freestream velocity vector This injects the question of whether the

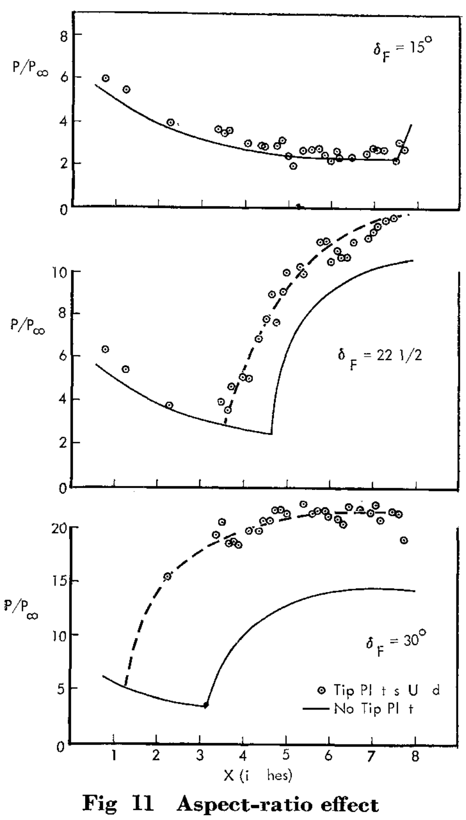

Fig 11 Aspect-ratio effect

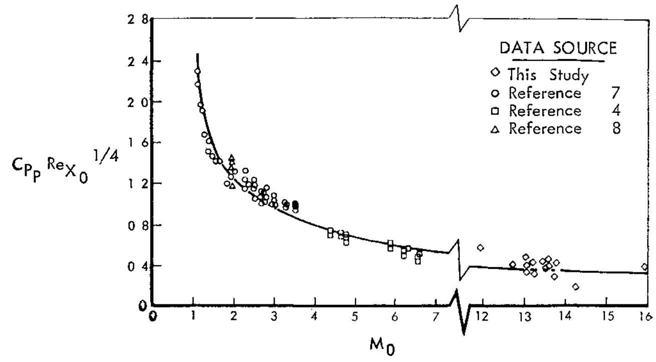

Fig 12 Variation of plateau pressure parameter with Mach number

changes result from a truly two-dimensional separation or if the corner interaction is a significant influence This question is not answered conclusively, but it does not appear from the comparison of the 15° flap-deflection data (Fig 11) that there are perceptible tip-plate influences in this essentially unseparated situation It is concluded, therefore, that the tip-plate data for the $22\frac{1}{2}°$ and 30° flap deflections is essentially representative of a two-dimensional separation

The increased plateau pressure and the forward movement of the separation point as shown in Fig 11 is significant because winged hypersonic vehicle designs generally have tip-mounted vertical tails that, with the body, effectively form tip plates

Models with smaller aspect ratios were also tested, primarily to study Reynolds number effects to occur with increased chord lengths

Discussion of Results: Mach and Reynolds Numbers

Influence of Mach Number

Chapman, Kuehn, and Larson,[7] Erdos and Pallone,[16] and others[8] have developed simplified analyses for flow through the free interaction region of a separated flow In these analyses, a linearized equation of motion relating the pressure coefficient to the rate of change of displacement thickness of the boundary layer is written for the flow external to the boundary layer This external pressure is assumed to be impressed at the wall The momentum equation of the boundary layer is then written for the flow near the wall Order-of-magnitude considerations of the two equations reduce them to a form that allows them to be combined so as to yield an expression for pressure coefficient as a function of the coefficient of friction at the wall and the Mach number of the external flow at the beginning of the interaction

The analysis by Erdos and Pallone, for instance, leads to expressions for laminar plateau and separation coefficients of the form

$$C_{P_p} = (Re_{X_0})^{-1/4} \xi^{1/2} f_3(1) \frac{[2C_{f_0} Re_{X_0}{}^{1/2}]^{1/2}}{[M_0{}^2 - 1]^{1/4}} \quad (1)$$

$$C_{P_s} = (Re_{X_0})^{-1/4} \xi^{1/2} f_3 \left(\frac{X_s - X_0}{l_i}\right) \frac{[2C_{f_0} Re_{X_0}{}^{1/2}]^{1/2}}{[M_0{}^2 - 1]^{1/4}} \quad (2)$$

where $\xi^{1/2}$ is a correction factor that may be applied to allow for the deviations from linearized flow assumed in the analysis

Erdos and Pallone show that data from Refs 4, 7, and 8 follow the preceding relationships reasonably well for Mach numbers in the range $1\ 1 < M_0 < 6\ 5$ if $f_3(1) = 1\ 47$, $f_3[(X_s - X_0)/l_i] = 0\ 81$, and $\xi = 1$

They suggest that extension of the preceding relationships to high-speed flow may be made if allowance is made for the viscous interaction effects in evaluating pressure and skin-friction coefficient at the beginning of the interaction

Such an extension has been made for the data obtained in this investigation The results are shown in Fig 12, where $C_{P_p} Re_{X_0}{}^{1/4}$ has been plotted as a function of M_0 Local values at the edge of the boundary layer were determined by using isentropic flow relationships with $\gamma = 1\ 4$ and the measured wall pressures Allowance for dissipative effects associated with flow through the bow shock could, of course, affect the plotted values substantially Also shown on Fig 12 are the lower-speed data from Refs 4, 7, and 8 which were given by Erdos and Pallone

The curve shown in Fig 12 represents extension to the high Mach number range of the relationships given in Eqs (1) and (2) Van Driest's relationship for skin-friction coefficient, assuming adiabatic flow, has been used in determining the curve No adjustment has been made for deviations of ξ from a value of unity Clearly, the use of other methods of determining C_{f_0} could lead to curves that might differ from the one shown Also, allowance for heat transfer and for deviations of ξ from unity would alter the curves As the figure shows, the data do not differ too greatly from the values that would be computed from Eq (1) Allowances for heat transfer and for deviation of ξ from unity would shift the curve upward and bring it into closer agreement with the data The magnitude of the deviations of the data from an average value is about the same as that observed at lower Mach numbers; however, the percentage deviation is larger

When Mach number is held constant in Eq (1), the pressure coefficient becomes proportional to the square root of the Stanton number if Reynolds' analogy is assumed to hold up to the beginning of interaction The data in Fig 13 tend to support use of the analogy

In connection with the influence of Mach number on separation, it is interesting to note that increasing the Mach number while maintaining flap angle and Reynolds number per foot constant results in a rearward displacement of the separation point For example, in tests with a 15° flap at $M_\infty = 8$ and a Reynolds number per foot of approximately 700,000, separation has been observed at a considerable distance forward of the flap On the other hand, for flow at M_∞ of approximately 16 over the same model and at a comparable Reynolds number per foot, separation did not occur until just ahead of the hinge line This same behavior was observed in other tests

Influence of Reynolds Number

Figure 14 shows the effect observed in this investigation of changing Reynolds number per foot while keeping the freestream Mach number constant and maintaining a given geometrical configuration As indicated in the figure, the beginning of the free interaction moved forward as the Reynolds number increased In previous investigations of laminar separation at moderate supersonic speeds, an opposite effect of increasing Reynolds number was observed It is not possible at this point to give reasons for the differences observed between the high-speed and lower-speed flows It is possible to show in a qualitative way, however, that a displacement forward of the interaction with increasing Reynolds number may not be unreasonable

Consider the flow model shown in Fig 2 Assume that the correlation of $C_{P_p} Re_{X_0}{}^{1/4}$ is reasonably well established (Fig 12), that the mixing profiles adjacent to the reversed flow region have a chance to become fully developed and otherwise independent of upstream Reynolds number influence, and either isentropic or oblique-shock compressions of the external flow at separation and reattachment With these assumptions, the location of the beginning of interaction may be predicted for a given flap angle δ_F Conversely, the angle required to cause the free interaction to begin at a given point may be determined For example, for given

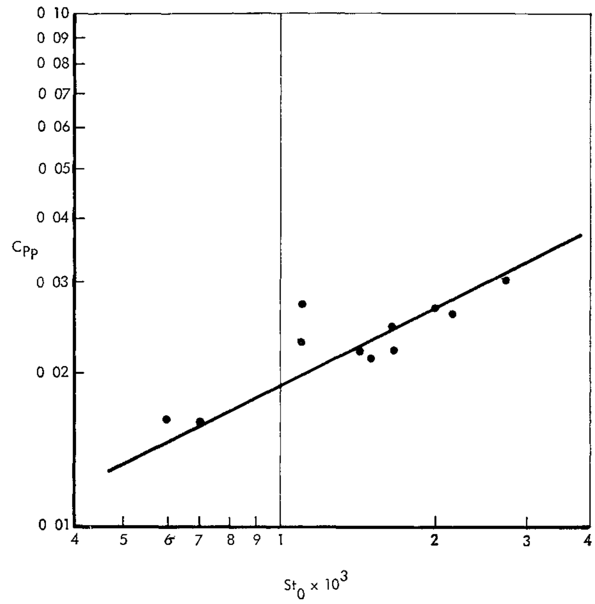

Fig 13 Plateau pressure correlation at constant Mach number

freestream flow conditions and an assumed value of X_0, C_{P_p} may be determined The Mach number and flow direction downstream of the separation shock may then be evaluated From a solution for the mixing layer, the total pressure on the separating streamline may be determined If this is now set equal to the computed value of static pressure downstream of the reattachment shock, the flap angle δ_F required to position the beginning of interaction at the chosen value of X_0 may be determined

Consider the influence of increasing Reynolds number per foot while M_∞ and the geometrical configuration are kept constant Suppose that X_0 remains unchanged; since Re_{X_0} would now be larger, C_{P_p} and, in turn, P_p/P_0 should be smaller This would require the external flow to turn through a smaller angle at separation This, in turn, would require the change in flow direction at reattachment to be larger Although the computed ratio of total pressure to static pressure on the dividing streamline should now be higher than before, the increased turning angle at reattachment would produce a pressure ratio across the reattachment shock of such magnitude that flow on the separating streamline would not be able to pass on downstream This would then require that the separation point or the beginning of the interaction move forward to a point such that P_p/P_0

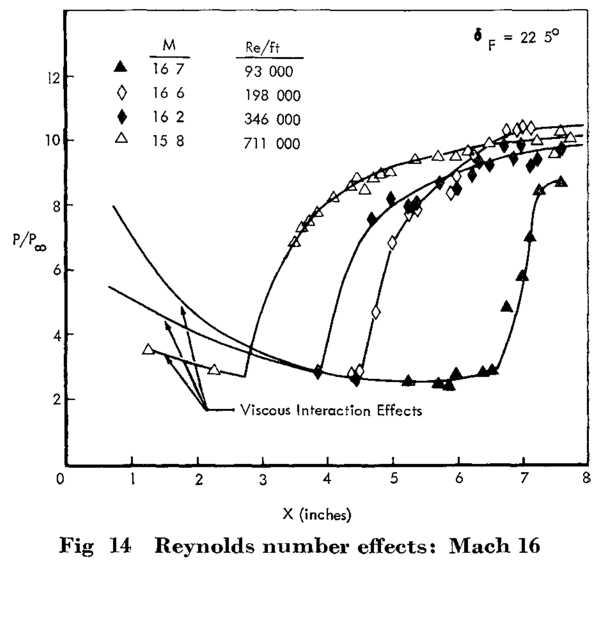

Fig 14 Reynolds number effects: Mach 16

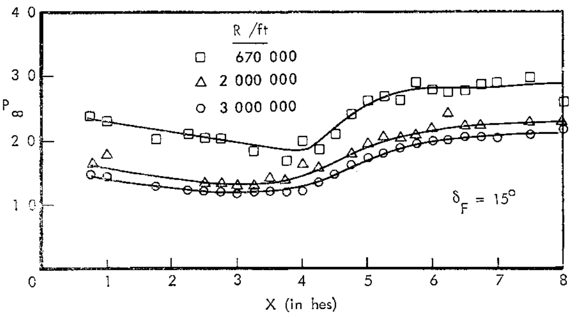

Fig 15 Reynolds number effects: Mach 8

and δ_{BL} increase sufficiently to allow $\delta_F - \delta_{BL}$ to decrease sufficiently that the flow on the dividing streamline is able to penetrate the compression at reattachment and pass on downstream

Clearly, quantitative agreement between observed values and values of X_0 predicted by the method just given should not be expected There are at least two reasons for this First, although reasonably good correlation has been found for the quantity $C_{P_p}Re_{X_0}{}^{1/4}$ as a function of M_0, deviations of $C_{P_p}Re_{X_0}{}^{1/4}$ of as much as 10% from an average value have been found for a given Mach number These deviations are even greater at Mach numbers in the range of those used in these tests Clearly, the predicted value of X_0 would depend strongly on the choice of the value of $C_{P_p}Re_{X_0}{}^{1/4}$ Second, and probably far more important, the method just described is based on the assumption that mixing profiles are fully developed and free of upstream Reynolds number influence It is not likely that this condition would be satisfied except for large ratios of separation length to initial boundary-layer thickness

The effect of upstream Reynolds number influence on the mixing process should be to reduce the ratio of stagnation to static pressure on the dividing streamline, thereby requiring that separation occur farther upstream than predicted by a method that ignores this upstream influence In fact, it is possible that changes in the mixing process caused by vari ation of Reynolds number might have a far greater influence on the location of separation than corresponding changes in P_p/P_0, δ_{BL}, and $\delta_F - \delta_{BL}$ Perhaps this accounts for the difference between results obtained in this investigation and those obtained in Ref 7 at much lower Mach numbers It is of interest to note that some recent data on laminar separation caused by a 15° flap at $M_\infty = 8$ show the location of the beginning of interaction to be independent of Reynolds number per foot, as shown in Fig 15

In the case of the results reported here, there are at least two other factors that may have influenced the location of separation As discussed earlier, the bow shock and separation shock interacted with each other and with the reattachment shock Schlieren photographs indicate that these interactions occurred near the surface of the flap This complex interaction could have had a substantial influence on the reattachment process and, thereby, on the location of separation It is not possible at this point to assess the effect that this interaction may have had In addition, differences from run to run in the ratio of wall temperature to freestream temperature which produced differences in Reynolds number could have influenced the location of separation Calculations indicate that this effect should be small for the variations in T_W/T_∞ which occurred in these tests, but whether this is in fact so is not yet known

Separation Length

Correlation of a characteristic length of the separated region with strength of the pressure disturbance is shown in Fig 16, where the characteristic lengths have been plotted as a function of Reynolds number per foot for several equivalent disturbance strengths Except as noted on the figure, the data are for a freestream Mach number of approximately 16 Such correlation is suggested by the conclusions of Ref 8 Although Ref 8 is concerned with incident-shock separation phenomena, the basic similarity of the phenomena at lower speeds, regardless of configuration, suggests the applicability of such conclusions to a flap configuration as well

In Ref 8, a length parameter was successfully correlated with a group of flow parameters according to an approximate analytical treatment Numerically, the correlation of the type made in Ref 8, Eq (44), was not found applicable to present data without a large change in constant of proportionality Application of the analysis to the data of the present tests disclosed that the dominant variables were the influenced length and the effective driving-pressure difference causing the separation In Ref 8, the driving-pressure difference was taken as the difference between a "final" pressure corresponding to reattachment and the undisturbed value upstream of the beginning of the separation process

For this analysis, in light of the results of Ref 8, the "final" pressures were taken to be the ideal-gas values corresponding to the flap angle in an inviscid flow For the driving pressures, the differences between these values and the corresponding values at the beginning of interaction are used The latter pressures correspond to undisturbed flat-plate values and, thus, could readily be determined In general, the beginning of interaction was far enough from the leading edge of the model that the initial pressure was influenced only moderately by viscous interaction effects and, thus, varied only slightly as the flap angle was varied Therefore, the driving-pressure difference was left significantly dependent only on the "final" pressure and, thus, on the flap angle

In Ref 8, the length correlated is that of the constant pressure region, which extends from just past the separation point to just ahead of the point of reattachment Because of the relative difficulty of pinpointing the location of reattachment

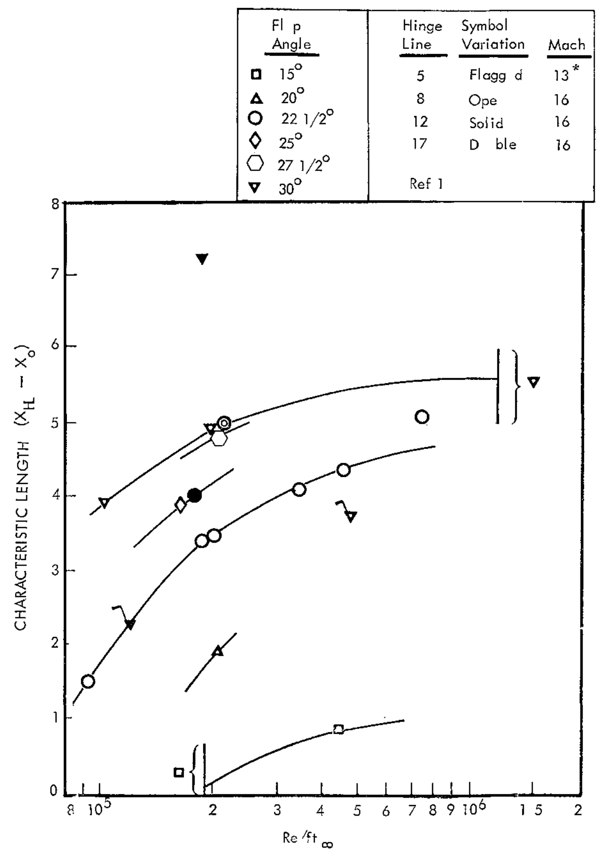

Fig 16 Characteristic length correlation

on the flap in most cases in the current tests, the significant length was taken, instead, as that from the beginning of interaction to the hinge line That this artifice is successful indicates, as do the conclusions of Ref 8, that separation phenomenon may be studied in two parts, with reattachment a subject in its own right, and that, in a flap configuration, the hinge line is a natural and appropriate dividing line

Conclusions

The influences on laminar separation phenomena of model geometry, Mach number, and Reynolds number are described Detailed pressure and heat-transfer measurements are correlated with flow visualization data, schlieren photographs, and oil-streak techniques to pinpoint the location of separation The heat-transfer measurements, in this situation, represent an extension to high speeds of the use of zero skin-friction criteria to define the separation point

It is shown that the high heating at reattachment can be calculated with reasonable accuracy using measured pressures The heat transferred to the wall in the separated region is a function of the plateau pressure and the length of the separated region, in accord with Chapman's analysis The ratio of the separated heat transfer to the unseparated heat transfer (corrected to plateau pressure conditions) is approximately 60% of Chapman's prediction in the case of the Mach 16 data presented

Important three-dimensional effects are considered, and a moderately strong influence of aspect ratio is indicated Sweep effects are also illustrated Evidence of a boundary-layer vortex system affecting heating at reattachment, and possibly just upstream of separation, is discussed It is concluded that, although the basic two-dimensional separation is not yet fully understood, it is necessary to devote more attention to three-dimensional aspects of separation

The wide ranges of Mach number and Reynolds number of these tests reveal interesting effects of unit Reynolds number variations At low supersonic speeds, it has been shown that increasing Reynolds number per foot decreases the separated length and the plateau pressure At Mach 8 no effects of unit Reynolds number on separated length are observed, whereas at Mach 16 separated length increases with increasing unit Reynolds number However, the plateau pressure and Reynolds number based on beginning of interaction, $C_{P_p} Re_{X_0}{}^{1/4}$, are shown to correlate with theory over the range of Mach numbers tested These successful correlations indicate that separations at small angles of attack can be evaluated through the use of appropriate edge conditions It is inferred that larger flap deflections are required to separate flow as Mach number increases when data are compared at common Reynolds numbers The influences of flap length and of wall temperature on separation are not well defined, though, and these also mark areas requiring further research

It is possible, based on these results, to synthesize a separation pressure distribution that could be of considerable value in force testing, especially control effectiveness studies Plateau pressures can be evaluated for given combinations of Mach and Reynolds numbers, whereas characteristic separation shock patterns appearing in schlieren photographs or shadowgraphs can be used to fix location of separation The problems of scaling separation lengths to flight conditions are not understood, and full-scale separations cannot necessarily be inferred from such wind tunnel tests until these scaling problems are solved

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