.I 1

.T

experimental investigation of the aerodynamics of a

wing in a slipstream .

.A

brenckman,m.

.B

j. ae. scs. 25, 1958, 324.

.W

experimental investigation of the aerodynamics of a

wing in a slipstream .

an experimental study of a wing in a propeller slipstream was

made in order to determine the spanwise distribution of the lift

increase due to slipstream at different angles of attack of the wing

and at different free stream to slipstream velocity ratios . the

results were intended in part as an evaluation basis for different

theoretical treatments of this problem .

the comparative span loading curves, together with

supporting evidence, showed that a substantial part of the lift increment

produced by the slipstream was due to a /destalling/ or

boundary-layer-control effect . the integrated remaining lift

increment, after subtracting this destalling lift, was found to agree

well with a potential flow theory .

an empirical evaluation of the destalling effects was made for

the specific configuration of the experiment .

.I 2

.T

simple shear flow past a flat plate in an incompressible fluid of small viscosity .

.A

ting-yili

.B

department of aeronautical engineering, rensselaer polytechnic institute

troy, n.y.

.W

simple shear flow past a flat plate in an incompressible fluid of small viscosity .

in the study of high-speed viscous flow past a two-dimensional body it is usually necessary to consider a curved shock wave emitting from the nose or leading edge of the body . consequently, there exists an inviscid rotational flow region between the shock wave and the boundary layer . such a situation arises, for instance, in the study of the hypersonic viscous flow past a flat plate . the situation is somewhat different from prandtl's classical boundary-layer problem . in prandtl's original problem the inviscid free stream outside the boundary layer is irrotational while in a hypersonic boundary-layer problem the inviscid free stream must be considered as rotational . the possible effects of vorticity have been recently discussed by ferri and libby . in the present paper, the simple shear flow past a flat plate in a fluid of small viscosity is investigated . it can be shown that this problem can again be treated by the boundary-layer approximation, the only novel feature being that the free stream has a constant vorticity . the discussion here is restricted to two-dimensional incompressible steady flow .

.I 3

.T

the boundary layer in simple shear flow past a flat plate .

.A

m. b. glauert

.B

department of mathematics, university of manchester, manchester,

england

.W

the boundary layer in simple shear flow past a flat plate .

the boundary-layer equations are presented for steady

incompressible flow with no pressure gradient .

.I 4

.T

approximate solutions of the incompressible laminar

boundary layer equations for a plate in shear flow .

.A

yen,k.t.

.B

j. ae. scs. 22, 1955, 728.

.W

approximate solutions of the incompressible laminar

boundary layer equations for a plate in shear flow .

the two-dimensional steady boundary-layer

problem for a flat plate in a

shear flow of incompressible fluid is considered .

solutions for the boundary-layer thickness, skin friction, and the velocity

distribution in the boundary

layer are obtained by the karman-pohlhausen

technique . comparison with

the boundary layer of a uniform flow has also

been made to show the effect of

vorticity .

.I 5

.T

one-dimensional transient heat conduction into a double-layer

slab subjected to a linear heat input for a small time

internal .

.A

wasserman,b.

.B

j. ae. scs. 24, 1957, 924.

.W

one-dimensional transient heat conduction into a double-layer

slab subjected to a linear heat input for a small time

internal .

analytic solutions are presented for the transient heat

conduction in composite slabs exposed at one surface to a

triangular heat rate . this type of heating rate may occur, for

example, during aerodynamic heating .

.I 6

.T

one-dimensional transient heat flow in a multilayer slab .

.A

campbell,w.f.

.B

j. ae. scs. 25, 1958, 340.

.W

one-dimensional transient heat flow in a multilayer

slab .

in a recent contribution to the readers'

forum wassermann gave analytic

solutions for the temperature in a double

layer slab, with a triangular heat

rate input at one face, insulated at the other,

and with no thermal resistance

at the interface . his solutions were for the

three particular cases..

i propose here to give the general solution

to this problem, to indicate

briefly how it is obtained using the method of

reference 2, and to point out

that the solutions given by wassermann are

incomplete for times longer

than the duration of the heat input .

.I 7

.T

the effect of controlled three-dimensional roughness

on boundary layer transition at supersonic speeds .

.A

van driest,e.r. and mccauley,w.d.

.B

j. ae. scs. 27, 1960, 261.

.W

the effect of controlled three-dimensional roughness

on boundary layer transition at supersonic speeds .

experiments were performed in the 12-in. supersonic wind

tunnel of the jet propulsion laboratory of the california

institute of technology to investigate the effect of three-dimensional

roughness elements (spheres) on boundary-layer transition on a

tained at local mach numbers of 1.90, 2.71, and 3.67 by varying

trip size, position, spacing, and reynolds number per inch .

the results indicate that (1) transition from laminar to turbulent

flow induced by three-dimensional roughness elements begins

when the double row of spiral vortices trailing each element

contaminates and breaks down the surrounding field of vorticity, (2)

transition appears rather suddenly, becoming more violent with

increasing roughness height relative to the boundary-layer

thickness, (3) after the breakdown of the vorticity field, the strength

of the spiral vortices may still persist in the sublayer of the

ensuing turbulent flow, (4) lateral spacing of roughness elements has

little effect upon the initial breakdown (contamination) of the

laminar flow, and (5) the trip reynolds number where u

and v are the velocity and kinematic viscosity at the outer edge of

the boundary layer and k is roughness height, such that transition

occurs at the roughness position, varies as the position reynolds number to the one-fourth power, viz., where x is

trip position .

.I 8

.T

measurements of the effect of two-dimensional and three-dimensional

roughness elements on boundary layer transition .

.A

klebanoff,p.s.

.B

j. ae. scs. 22, 1955, 803.

.W

measurements of the effect of two-dimensional and three-dimensional

roughness elements on boundary layer transition .

in his study of the effect of roughness on transition, h. l.
dryden found, on the basis of available data, that the effect
of a two-dimensional roughness element such as a /trip wire/
could be represented reasonably well in terms of a functional
relation between and, where is the reynolds number
of transition based on distance from the leading edge, is the
height of the roughness element, and is the boundary-layer
displacement thickness at the position of the element . at his
suggestion some additional data were obtained, primarily to
extend the range to higher values of, during the course of an
investigation of transition on a flat plate conducted at the
national bureau of standards . after the results on the two-
dimensional roughness elements were obtained, it appeared to be
desirable to see whether a row of three-dimensional roughness

elements would behave in the same way .

.I 9

.T

transition studies and skin friction measurements on

an insulated flat plate at a mach number of 5.8 .

.A

korkegi,r.h.

.B

j. ae. scs. 23, 1956, 97.

.W

transition studies and skin friction measurements on

an insulated flat plate at a mach number of 5.8 .

an investigation of transition and skin friction on an insulated flat plate, 5 by 26 in., was made in the galcit 5 by 5 in. hypersonic wind tunnel at a nominal mach number of 5.8 .

the phosphorescent lacquer technique was used for transition detection and was found to be in good agreement with total-head rake measurements along the plate surface and pitot boundary-layer surveys . it was found that the boundary layer was laminar at reynolds numbers of at least 5 x 10 . transverse contamination caused by the turbulent boundary layer on the tunnel sidewall originated far downstream of the flat plate leading edge at reynolds numbers of 1.5 to 2 x 10, and spread at a uniform angle of 5 compared to 9 degree in low-speed flow .

the effect of two-dimensional and local disturbances was investigated . the technique of air injection into the boundary layer as a means of hastening transition was extensively used .

although the onset of transition occurred at reynolds numbers

as low as 10, a fully developed turbulent boundary layer was

not obtained at reynolds numbers much below 2 x 10

regardless of the amount of air injected .

a qualitative discussion of these results is given with emphasis

on the possibility of a greater stability of the laminar boundary

layer in hypersonic flow than at lower speeds .

direct skin-friction measurements were made by means of the

floating element technique, over a range of reynolds numbers

verified as being laminar over the complete range . with air

injection, turbulent shear was obtained only for reynolds

numbers greater than 2 x 10, this value being in good agreement with

earlier results of this investigation . the turbulent skin-friction

coefficient was found to be approximately 0.40 of that for

incompressible flow for a constant value of r, and 0.46 for an effective

reynolds number between 5 and 6 x 10 .

.I 10

.T

the theory of the impact tube at low pressure .

.A

chambre,p.l. and schaaf,s.a.

.B

j. ae. scs. 15, 1948, 735.

.W

the theory of the impact tube at low pressure .

a theoretical analysis has been made for an impact tube of the

relation between free-stream mach number and the impact and free-stream pressures and densities for extremely low pressures .

it is shown that the results differ appreciably from the

corresponding continuum relations .

.I 11

.T

similar solutions in compressible laminar free mixing

problems .

.A

napolitano,l.

.B

j. ae. scs. 23, 1956, 389.

.W

similar solutions in compressible laminar free mixing

problems .

there are in supersonic aerodynamics many situations of

practical interest wherein streams of different velocities and,

in general, different stagnation pressures mix with one another .

in the majority of these problems the interaction between the

two streams takes place in the presence of an axial pressure

gradient . its effect on the characteristics of the mixing may

influence significantly the performances of the devices wherein

the phenomena cited above occur . a theoretical and

experimental program of research to study mixing in the presence of

axial pressure gradients is being carried on at the polytechnic

institute of brooklyn .

.I 12

.T

some structural and aerelastic considerations of high

speed flight .

.A

bisplinghoff,r.l.

.B

j. ae. scs. 23, 1956, 289.

.W

some structural and aerelastic considerations of high

speed flight .

the dominating factors in structural design of high-speed aircraft are thermal and aeroelastic in origin . the subject matter is concerned largely with a discussion of these factors and their interrelation with one another . a summary is presented of some of the analytical and experimental tools available to aeronautical engineers to meet the demands of high-speed flight upon aircraft structures . the state of the art with respect to heat transfer from the boundary layer into the structure, modes of failure under combined load as well as thermal inputs and acrothermoelasticity is discussed . methods of attacking and alleviating structural and aeroelastic problems of high-speed flight are summarized . finally, some avenues of fundamental research are suggested .

.I 13

.T

similarity laws for stressing heated wings .

.A

tsien,h.s.

.B

j. ae. scs. 20, 1953, 1.

.W

similarity laws for stressing heated wings .

it will be shown that the differential equations for a heated

plate with large temperature gradient and for a similar plate at

constant temperature can be made the same by a proper

modification of the thickness and the loading for the isothermal plate .

this fact leads to the result that the stresses in the heated plate

can be calculated from measured strains on the unheated plate by

a series of relations, called the /similarity laws ./ the

application of this analog theory to solid wings under aerodynamic

heating is discussed in detail . the loading on the unheated analog

wing is, however, complicated and involves the novel concept

of feedback and /body force/ loading . the problem of stressing

a heated box-wing structure can be solved by the same analog

method and is briefly discussed .

.I 14

.T

piston theory - a new aerodynamic tool for the

aeroelastician .

.A

ashley,h. and zartarian,g.

.B

j. ae. scs. 23, 1956, 1109.

.W

piston theory - a new aerodynamic tool for the aeroelastician .

representative applications are described which illustrate the extent to which simplifications in the solutions of high-speed unsteady aeroelastic problems can be achieved through the use of certain aerodynamic techniques known collectively as /piston theory ./ based on a physical model originally proposed by hayes and lighthill, piston theory for airfoils and finite wings has been systematically developed by landahl, utilizing expansions in powers of the thickness ratio and the inverse of the flight mach number m . when contributions of orders and are negligible, the theory predicts a point-function relationship between the local pressure on the surface of a wing and the normal component of fluid velocity produced by the wing's motion . the computation of generalized forces in aeroelastic equations, such as the flutter determinant, is then always reduced to elementary integrations of the assumed modes of motion .

essentially closed-form solutions are given for the bending-torsion and control-surface flutter properties of typical section airfoils at high mach numbers . these agree well with results of more exact theories wherever comparisons can be fairly made . moreover, they demonstrate the increasingly important influence of thickness and profile shape as m grows larger, a discovery that would be almost impossible using other available aerodynamic tools . the complexity of more practical flutter analyses-e.g., on three-dimensional wings and panels-is shown to be substantially reduced by piston theory . an iterative procedure is outlined, by which improved flutter eigenvalues can be found through the successive introduction of higher-order terms in and .

other applications to unsteady supersonic problems are reviewed, including gust response and rapid maneuvers of elastic aircraft . steady-state aeroelastic calculations are also discussed, but for them piston theory amounts only to a slight modification of ackeret's formulas .

suggestions are made regarding future research based on the new aerodynamic method, with particular emphasis on areas where computational labor can be reduced with a minimum loss of precision . it is pointed out that a mach number zone exists where thermal effects are appreciable but nonlinear viscous interactions may be neglected, and that in this zone piston theory is the logical way of estimating air loads when analyzing aerodynamic-thermoelastic interaction problems .

.I 15

.T

on two-dimensional panel flutter .

.A

fung,y.c.

.B

j. ae. scs. 25, 1958, 145.

.W

on two-dimensional panel flutter .

theory and experiments of the flutter of a buckled plate are discussed . it is shown that an increase in the initial deviation from flatness or a static pressure differential across the plate raises the critical value of the /reduced velocity ./

the applicability of the galerkin method to the linearized problem of flutter of an unbuckled plate has been questioned by several authors . in this paper the flutter condition was formulated in the form of an integral equation and solved numerically by the method of iteration and the method of matrix approximations, thus avoiding the constraint of assumed modes . for a plate (with finite bending rigidity) the results confirm those given by the galerkin method .

an approximate analysis of the limiting form and amplitude of the flutter motion for a buckled plate is presented .

.I 16

.T

transformation of the compressible turbulent boundary layer .

.A

mager,a.

.B

j. ae. scs. 25, 1958, 305.

.W

transformation of the compressible turbulent boundary layer .

the transformation of the compressible turbulent boundary-layer equations to their incompressible equivalent is demonstrated analytically . the transformation is essentially the same as that for the laminar layer, first given by stewartson, except that the explicit relation between the viscosity and temperature is not required . a key point in the analysis is the modification of the stream function to include a mean of the fluctuating components and the postulate that the apparent turbulent shear, associated with an elemental mass, remains invariant in the transformation .

the values of the incompressible friction coefficients and of pressure rise causing separation thus transformed show good agreement with the experimentally measured and independently reported results . an application of the transformation to the self-preserving boundary layers and to the computations of general boundary-layer flow is shown .

.I 17

.T

remarks on the eddy viscosity in compressible mixing flows .

.A

lu ting and paul a. libby

.B

polytechnic institute of brooklyn, and general applied science laboratories, inc.

.W

remarks on the eddy viscosity in compressible mixing flows .
in connection with a study of the wakes behind bodies in hypersonic flow carried out for the missile and space vehicle division of the general electric company, it was desired to estimate the eddy viscosity in axisymmetric, compressible wakes . because of the lack of applicable experimental data, it was found necessary to make such an estimate by rationally extending the few available data for incompressible flows to the compressible case . this suggested the application and extension of

the transformations applied to turbulent boundary layers in reference

infinitesimal mass are invariant with transformation, mager showed that

the partial differential equations for the compressible turbulent

boundary layer can be transformed to incompressible form . the validity of

this assumption and of the transformations was established for several

boundary-layer flows by comparison with experiment .

.I 18

.T

the flow field in the diffuser of a radial compressor .

.A

rhyming,i.l.

.B

j. ae. scs. 27, 1960, 798.

.W

the flow field in the diffuser of a radial compressor .

this note discusses the two-dimensional diffuser flow field

in a radial compressor outside the impeller wheel . it is

assumed that the diffuser has guide vanes arranged in a circular

row at a radius . the impeller wheel has the radius (see

fig. 1) . the flow in the diffuser starts at the circle with the

radius . the velocity components, and in the r and

directions of the velocity vector on this circle are prescribed

together with the thermal state of the gas . the flow so prescribed

on the radius will, if no disturbances are present (i.e., no

boundary conditions in the flow other than zero velocity at

infinity are to be fulfilled), develop in a spiral flow .

.I 19

.T

an investigation of the pressure distribution on conical bodies in

hypersonic flows .

.A

victor zakkay

.B

aerodynamics laboratory, polytechnic institute of brooklyn, freeport,

n.y.

.W

an investigation of the pressure distribution on conical bodies in

hypersonic flows .

a large amount of work on conical flow fields without axial symmetry

at supersonic speed is presently available . however, no apparent

hypersonic approximation has yet been derived . in this note,

experimental data on two elliptical cones at m = 6 are presented and a

hypersonic approach obtained from physical considerations is suggested .

.I 20

.T

generalised-newtonian theory .

.A

love,e.s.

.B

j. ae. scs. 26, 1959, 314.

.W

generalised-newtonian theory .

author generalizes lees's (amr 10(1957), rev. 2601)

modification of newtonian theory for blunt-nose bodies to apply to pointed-nose bodies as well . the result is expressed by

sin where is the local inclination of the body

surface and the subscript /max/ refers to the maximum local

inclination and pressure coefficient . for blunt-nose bodies

and the generalized theory reverts to lees's blunt-nose

modification with given by normal shock relations . author shows,

by comparison of newtonian and generalized-newtonian theory

with exact solutions, the superiority of generalized-newtonian

theory . he also shows that both two-dimensional and

axisymmetric shapes are correlated by this generalization . results are

presented in two figures that support author's generalization and

indicate the independence of the correlation from variations in

both the hypersonic similarity parameter k = m(d1) and the ratio

of specific heats y .

reviewer believes this generalization should be of interest to

those engaged in development of hypersonic hardware as well as

theory .

.I 21

.T

on heat transfer in slip flow .

.A

stephen h. maslen

.B

lewis flight propulsion laboratory, naca, cleveland, ohio

.W

on heat transfer in slip flow .

a number of authors have considered the effect of slip on the heat

transfer and skin friction in a laminar boundary layer over a flat plate .
reference 1 considers this by a perturbation on the usual laminar
boundary-layer analysis while some other studies.dash e.g., reference
the impulsive motion of an infinite plate .
.I 22
.T
on slip-flow heat transfer to a flat plate .
.A
oman,r.a. and scheuing,r.a.
.B
j. ae. scs. 26, 1959, 126.
.W
on slip-flow heat transfer to a flat plate .
  assuming that continuum flow energy equation in a boundary
layer remains valid well into slip region and taking account of the
temperature jump in a moving rarefied gas and for influence of
large mean free path through appropriate boundary conditions, a
solution is found for the temperature gradient in the slip region .
then from maslen expression (j. aero. sci. 25, 6, 400-401, june
slipping fluid to a flat plate, and behavior confirms results for
small values of knudsen number .
.I 23
.T
skin-friction and heat transfer characteristics of
a laminar boundary layer on a cylinder in axial incompressible
flow .
.A

seban,r.a. and bond,r.

.B

j. ae. scs. 18, 1951, 671.

.W

skin-friction and heat transfer characteristics of

a laminar boundary layer on a cylinder in axial incompressible

flow .

a solution is given for the case of the laminar boundary layer

of an incompressible fluid of constant properties on the exterior

of a cylinder with flow parallel to the cylinder axis . this case

differs from the blasius solution for flow along a flat plate by

considering the effect of the curvature in a plane transverse to

the flow direction . the local skin-friction and heat-transfer

coefficients for a prandtl number of 0.715 are evaluated and

compared to the similar magnitudes for flat plate flow, and the

effect of the curvature is shown to be significant in some practical

cases . recovery factors are evaluated, and this quantity is

found to be insensitive to the effect of curvature of the boundary .

.I 24

.T

theory of stagnation point heat transfer in dissociated

air .

.A

fay,j.a. and riddell,f.r.

.B

j. ae. scs. 25, 1958, 73.

.W

theory of stagnation point heat transfer in dissociated air .

the boundary-layer equations are developed in general for the case of very high speed flight where the external flow is in a dissociated state . in particular the effects of diffusion and of atom recombination in the boundary layer are included . it is shown that at the stagnation point the equations can be reduced exactly to a set of nonlinear ordinary differential equations even when the chemical reactions proceed so slowly that the boundary layer is not in thermochemical equilibrium .

two methods of numerical solution of these stagnation point equations are presented, one for the equilibrium case and the other for the nonequilibrium case . numerical results are correlated in terms of the parameters entering the numerical formulation so as not to depend critically on the physical assumptions made .

for the nonequilibrium boundary layer, both catalytic (to atom recombination) and noncatalytic wall surfaces are considered . a solution is represented which shows the transition from the /frozen/ boundary layer (very slow recombination rates) to the equilibrium boundary layer (fast recombination rates) . a recombination rate parameter is introduced to interpret the nonequilibrium results, and it is shown that a scale factor is involved in relating the equilibrium state of a boundary layer on bodies of different sizes .

it is concluded that the heat transfer through the equilibrium stagnation point boundary layer can be computed accurately by a simple correlation formula and that the heat

transfer is almost unaffected by a nonequilibrium state of the

boundary layer provided the wall is catalytic and the lewis

number near unity .

.I 25

.T

inviscid hypersonic flow over blunt-nosed slender bodies .

.A

lees,l. and kubota,t.

.B

j. ae. scs. 24, 1957, 195.

.W

inviscid hypersonic flow over blunt-nosed slender bodies .

at hypersonic speeds the drag area of a blunt nose is much

larger than the drag area of a slender afterbody, and the energy

contained in the flow field in a plane at right angles to the flight

direction is nearly constant over a downstream distance many

times greater than the characteristic nose dimension . the

transverse flow field exhibits certain similarity properties directly

analogous to the flow similarity behind an intense blast wave

found by g. i. taylor, s. c. lin, and a. sakurai . a comparison

with the experiments of hammitt, vas, and bogdonoff on a

flat plate with a blunt leading edge at in helium shows

that the shock-wave shape is predicted very accurately by this

similarity analysis . the predicted surface pressure distribution

is somewhat less satisfactory . experimental results on a

hemisphere-cylinder obtained at in the galcit air tunnel indicate that not only the shock-wave shape but also the surface pressures for this body are given very closely by the similarity theory, except near the hemisphere-cylinder junction .

energy considerations combined with a detailed study of the equations of motion show that flow similarity is also possible for a class of bodies of the form, provided that, where for a two-dimensional body and for a body of revolution . when the shock shape is not similar to the body shape, and the entire flow field some distance from the nose must depend to some extent on the details of the nose geometry .

by again utilizing energy and drag considerations one finds that at hypersonic speeds the inviscid surface pressures generated by a blunt leading edge are larger than the pressures induced by boundary-layer growth on an insulated flat surface for an insulated blunt-nosed slender body of revolution the corresponding distance is given by . (here is free-stream reynolds number based on leading-edge thickness, or nose diameter .) in free flight these constants are replaced by 1,700 and 20, respectively, so that viscous interaction effects are important over the forward portion of a blunt-nosed slender body only for relatively low values of . however, /far downstream/ of the nose the inviscid over-pressure is small and viscous interaction phenomena will have to be taken into account .

.I 26

.T

inviscid leading-edge effect in hypersonic flow .

.A

cheng,h.k. and pallone,a.j.

.B

j. ae. scs. 23, 1956, 700.

.W

inviscid leading-edge effect in hypersonic flow .

current interest in the problem of inviscid-viscous

interaction has led to the realization of the significant effect of

the leading-edge thickness in hypersonic flow . the purpose

of this note is to give an account of the downstream influence

of the blunt leading edge on the basis of the hypersonic small

perturbation theory .

.I 27

.T

newtonian flow theory for slender bodies .

.A

cole,j.d.

.B

j. ae. scs. 24, 1957, 448.

.W

newtonian flow theory for slender bodies .

as an aid to the aerodynamieist in the design of air frames for

hypersonic speeds (speeds faster than about mach 5), newtonian

flow theory is examined from the point of view of gas dynamics

and hypersonic small-disturbance theory . the usual theory is

shown to result as the first approximation of an expansion valid for small . a basic similarity parameter

is introduced . a general solution

of the first approximation for the flow past slender bodies (bodies

which cause only a small disturbance to the stream) at zero angle

of attack is given . an important condition which limits the

application of the theory is noted-namely, that the pressure

coefficient on the surface not fall to zero . the theory is then

applied to cones and to bodies whose shape is .

.I 28

.T

a note on the explosion solution of sedov with application

to the newtonian theory of unsteady hypersonic flow .

.A

freeman,n.c.

.B

j. ae. scs. 27, 1960, 77.

.W

a note on the explosion solution of sedov with application

to the newtonian theory of unsteady hypersonic flow .

an exact analytical solution of the equations of inviscid

compressible unsteady flow has been given by sedov (reference

to the solution may be made through hayes and probstein) .

this solution is the similarity solution for a constant-energy point

explosion . in view of the recent work on problems of hypersonic

flow in the limiting form of the ratio of specific heats near 1

solution in this limit and inquire what form such a solution would

take . einbinder, in a recent note, has examined the solution for various but does not mention the interesting case of .

it may be shown that the convergence to the limit is nonuniform

over the flow field . it is also not difficult to show that the non-

uniform behavior exhibited here is that which one would expect

from the newtonian formulation as derived in reference 3 .

.I 29

.T

a simple model study of transient temperature and thermal

stress distribution due to aerodynamic heating .

.A

isakson,g.

.B

j. ae. scs. 24, 1957, 611.

.W

a simple model study of transient temperature and thermal

stress distribution due to aerodynamic heating .

  the present work is concerned with the determination of

transient temperatures and thermal stresses in simple models intended

to simulate parts or the whole of an aircraft structure of the built-

up variety subjected to aerodynamic heating .

  the first case considered is that of convective heat transfer

into one side of a flat plate, representing a thick skin, and the

effect of the resulting temperature distribution in inducing

thermal stresses associated with bending restraint at the plate edges .

numerical results are presented for the transient temperature

differentials in the plate when the environment temperature first

increases linearly with time and then remains constant, the

period of linear increase representing the time of acceleration of
the aircraft . corresponding thermal stress information is
presented .

the second case is that of the wide-flanged i-beam with
convective heat transfer into the outer faces of the flanges . numerical
results are presented for transient temperature differentials for a
wide range of values of the applicable parameters and for an
environment temperature variation as described above .
corresponding thermal stresses in a beam of infinite length are
determined . a theoretical analysis of the stress distribution in a beam
of finite length is carried out and numerical results obtained for
one case . an experimental investigation of temperatures and
stresses in such a beam is described, and results are presented
which indicate good agreement with corresponding theoretical
results .

.I 30

.T

photo-thermoelastic investigation of transient thermal
stresses in a multiweb wing structure .

.A

gerard,g. and tramposch,h.

.B

j. ae. scs. 26, 1959, 783.

.W

photo-thermoelastic investigation of transient thermal
stresses in a multiweb wing structure .

photothermoelastic experiments were performed on a long multiweb wing model for which a theoretical analysis is available in

the literature . the experimental procedures utilized to simulate

the conditions prescribed in the theory are fully described .

correlation of theory and experiment in terms of dimensionless

temperature, stress, time, and biot number revealed that the

theory predicted values higher than the experimentally observed

maximum thermal stresses at the center of the web . detailed

temperature measurements in the flange suggested that the major

source of this discrepancy can be traced to the one-dimensional

heat conduction analysis of the flange employed in the theory .

.I 31

.T

thermal buckling of supersonic wing panels .

.A

hoff,n.j.

.B

j. ae. scs. 23, 1956, 1019.

.W

thermal buckling of supersonic wing panels .

the temperature and thermal stress distributions are analyzed

in multicellular supersonic wing structures . a buckling criterion

is established for the panels of cover plates subjected to thermal

stresses .

.I 32

.T

the dynamic motion of a missile descending through

the atmosphere .

.A

friedrich,h.r. and dore,f.j.

.B

j. ae. scs. 22, 1955, 628.

.W

the dynamic motion of a missile descending through

the atmosphere .

a method is presented for computing rapidly, yet accurately,

the dynamic motion of a ballistic-type missile descending through

the atmosphere . the equations of motion are separated into a

set of /static/ trajectory equations (zero angle of attack) and a

set of /rotational/ equations describing the oscillatory motion

of the missile about its center of gravity . a transformation

allows the rotational equations to be written in a manner

analogous to the equation for an undamped oscillating spring mass

system with the mass equal to unity and a time variable spring

constant . for given initial conditions this equation can be

solved to obtain the envelope of maximum angle of attack . an

additional transformation allows the calculation of the complete

oscillatory motion at any time during the trajectory as a function

of the maximum angle of attack at that time .

this solution shows that the maximum angle of attack of a

missile descending through the atmosphere at relatively constant

speed is reduced even when the aerodynamic damping is neglected .

.I 33

.T

the prospects for magneto-aerodynamics .

.A

resler,e.j. and sears,w.r.

.B

j. ae. scs. 25, 1958, 235.

.W

the prospects for magneto-aerodynamics .

the equations describing the flow of an electrically conducting fluid in the presence of electric and magnetic fields are written down with the aid of certain simplifications appropriate to aeronautical applications . in order to estimate the probable significance of magneto-aerodynamic effects, some data on conductivity of pure and /seeded/ air are first examined . dimensionless quantities representing the ratios of forces and of currents are then formed and their values studied for conditions of flight in the atmosphere .

some examples of magneto-hydrodynamic and magneto-gasdynamic effects in simple flows are given . these include two cases of poiscuille flow of conducting liquids with applied magnetic fields and the case of quasi-one-dimensional gas flow with applied electrical and magnetic fields . in the last case, attractive possibilities are found for controlled acceleration or deceleration of gas at subsonic and supersonic speeds, even in constant-area channels . the behavior of the flow is characteristically different in different regimes of mach number and flow speed relative to certain /significant speeds/ that are dependent on the ratio of electrical to magnetic field strengths . these are studied, and a chart is constructed to relate the length to the speed ratio of a maximum-acceleration constant-area channel .

it is concluded that the advantages that may accrue from magneto-aerodynamic methods are sufficiently attractive to justify the considerable research and engineering development that will be required . among the unsolved engineering problems are the reduction of surface resistance of electrodes in contact with a conducting gas, development of techniques for seeding, and provision of the required magnetic fields in flight .

.I 34

.T

constant-temperature magneto-gasdynamic channel flow .

.A

kerrebrock,j.p. and marble,f.e.

.B

j. ae. scs. 27, 1960, 78.

.W

constant-temperature magneto-gasdynamic channel flow .

in the course of investigating boundary-layer flow in continuous plasma accelerators with crossed electric and magnetic fields, it was found advantageous to have at hand simple closed-form solutions for the magneto-gasdynamic flow in the duct which could serve as free-stream conditions for the boundary layers . nontrivial solutions of this sort are not available at present, and in fact, as in the work of resler and sears, the variation of conditions along the flow axis must be obtained through numerical integration .

consequently, some simple solutions of magneto-gasdynamic channel flow were sought, possessing sufficient algebraic simplicity

to serve as free-stream boundary conditions for analytic

investigations of the boundary layer in a physically reasonable accelerator .

in particular, since the cooling of the accelerator tube is likely to

be an important physical problem because of the high gas

temperatures required to provide sufficient gaseous conductivity,

channel flow with constant temperature appears interesting .

some simple algebraic solutions for the case of a constant

temperature plasma are developed in the following paragraphs .

.I 35

.T

stagnation point of a blunt body in hypersonic flow .

.A

li,t.y. and geiger,r.e.

.B

j. ae. scs. 24, 1957, 25.

.W

stagnation point of a blunt body in hypersonic flow .

the purpose of this paper is to present a method of calculation

devised to yield all the important information on the symmetric

inviscid hypersonic flow in the stagnation point region of a blunt

body . the problem is the same as that considered by hayes

who used a slightly different approach . it is demonstrated that

hayes' results are valid in the stagnation point region and can

hence be considered a basis for constructing less restricted

solutions .

equations are presented giving velocity, pressure, detachment distance, and vorticity . the values of shock detachment

distance and body pressure coefficient are compared with

experimental data for spheres . the pressure comparison shows that

the results of hayes and the theory presented herein represent a

better approximation than the newtonian impact theory for

hypersonic mach numbers .

in conclusion, the possibility of refinements to this analysis is

discussed .

.I 36

.T

supersonic flow around blunt bodies .

.A

serbin,h.

.B

j. ae. scs. 25, 1958, 58.

.W

supersonic flow around blunt bodies .

the newtonian theory of impact has been shown to be

useful for pressure calculations on the forward facing part

of bodies moving at high speed . it is now a familiar practice

to use this information to calculate nonviscous velocities at the

wall and then to estimate rates of heat transfer . this

procedure is perhaps open to question,. heat-transfer rates depend

on velocity gradients which are not given by the newtonian

analysis . nor can one obtain information on boundary-layer

stability or all the body stability derivatives . it seems,

therefore, inevitable that, as design proceeds with these hypersonic missiles, there will be a greater need for more accurate aerodynamic theories either to predict what will happen in unfamiliar flight conditions or to effect an extrapolation from a known test result to the design condition .

.I 37

.T

a new technique for investigating heat transfer and surface phenomena under hypersonic flow conditions .

.A

ferri,a. and libby,p.a.

.B

j. ae. scs. 24, 1957, 464.

.W

a new technique for investigating heat transfer and surface phenomena under hypersonic flow conditions .

on the forebody of many practically interesting hypersonic vehicles, there is little interaction between the inviscid flow field and the boundary layer . therefore, inviscid flow theory can be used to determine, independent of surface phenomena, the physically interesting quantities such as shock shape, shock detachment distance, sonic line shape, and pressure distribution . furthermore, the pressure distribution so determined can then be used for the study of heat transfer, materials behavior, and other surface phenomena . thus, for these bodies, the prandtl boundary-layer concept can be utilized for the calculation of both the inviscid flow and the boundary-layer behavior .

it is the purpose of this note to point out that this concept can also be applied experimentally in order to provide, in

conjunction with a conventional hypersonic wind-tunnel air supply, a

means for investigating hypersonic heat transfer and surface

phenomena under conditions of flight reynolds numbers .

.I 38

.T

on the prediction of mixed subsonic/supersonic pressure

distributions .

.A

sinnott,c.s.

.B

j. ae. scs. 27, 1960, 767.

.W

on the prediction of mixed subsonic/supersonic pressure

distributions .

high-speed wind-tunnel results are analyzed to derive a

semiempirical scheme for the prediction of transonic pressure

distributions . the supersonic and subsonic parts of the flow are

treated separately, and then linked by an empirical shock

pressure rise relation . the significance of the empirical results is

considered in relation to the physical mechanism of transonic

flows . it is also shown that theoretical solutions can be

improved by introducing the empirical shock relation .

.I 39

.T

on the flow of a sonic stream past an airfoil surface .

.A

sinnott,c.s.

.B

j.ae.scs. 26, 1959, 169.

.W

on the flow of a sonic stream past an airfoil surface .

this study of the flow about an airfoil in a near-sonic stream indicates the important factors determining the pressure distribution on the airfoil . analysis of the mach wave pattern suggests that the supersonic domain of the flow can be derived from two simple-wave flows, one arising from the mach waves reflected at the sonic line and the other from the changes in airfoil surface slope . the compressive effect of the reflected mach waves is determined quantitatively as a function of airfoil leading-edge geometry from an analysis of measured pressure distributions for uncambered airfoils,. and it is shown how this can be superimposed on the wave system from the curved surface to give an equivalent simple-wave flow over the airfoil .

an application of this scheme to the calculation of the pressure distribution over an airfoil in a sonic stream gives results in good agreement with experiment .

.I 40

.T

experiments on boundary layer transition at

supersonic speeds .

.A

van driest,e.r. and boison,j.c.

.B

j. ae. scs. 24, 1957, 885.

.W

experiments on boundary layer transition at

supersonic speeds .

tests were conducted in the 12-in. continuous supersonic wind tunnel of the jet propulsion laboratory, california institute of technology, to determine the effects of surface cooling on boundary-layer transition at supersonic speeds . the effects of cooling were investigated at test section mach numbers of 1.97, smooth cone in the presence of three levels of supply-stream turbulence (0.4, 2, and 9 per cent) and several single-element roughnesses at fixed axial location . transition data were obtained optically by means of a magnified-schlieren system . the results, for the range of mach number investigated, indicate that (1) transition on a smooth cone can definitely be delayed by surface cooling, (2) transition promoted by either supply-stream turbulence or surface roughness can also be delayed by surface cooling depending upon degree of turbulence or relative roughness respectively, and (3) the adverse effects of increased turbulence and roughness decrease with increasing mach number .

.I 41

.T

on transition experiments at moderate supersonic speeds .

.A

morkovin,m.v.

.B

j. ae. scs. 24, 1957, 480.

.W

on transition experiments at moderate supersonic speeds .

studies of transition over a flat plate at mach number 1.76 were carried out using a hot-wire anemometer as one of the principal tools . the nature and measurements of free-stream disturbances at supersonic speeds are analyzed . the experimental results are interpreted in the light of present overall information on transition at supersonic speeds and conclusions as to further fruitful experiments are drawn .

.I 42

.T

the gyroscopic effect of a rigid rotating propeller
on engine and wing vibration modes .

.A

scanlan,r.h. and truman,j.c.

.B

j. ae. scs. 17, 1950, 653.

.W

the gyroscopic effect of a rigid rotating propeller
on engine and wing vibration modes .

in many wing vibration analyses it is found necessary to take into account the effect of flexibly mounted engines . hence, it is reasonable to ask what vibratory gyroscopic effect this flexibility may give rise to when propellers are whirling . an engine mount may be thought of as a horizontal beam cantilevered from the wing, having both horizontal and vertical flexibility . if this beam were infinitely rigid horizontally, then, when it vibrated, the gyroscopic moments induced in the propeller due to the resultant pitching motion of its axis would not produce propeller axis yaw . however, engine-mount lateral stiffness tical stiffness, so that gyroscopic effects will play a role as the propeller axis undergoes pitching vibrations at the tip of the cantilever engine mount . the purpose of this paper is to investigate this role under the assumption that the propeller itself is a rigid disc .

the paper is divided into four parts . part (1) deals briefly with classical gyroscope theory . part (2) presents engine vibration mode studies-experimental photographic techniques on a model gyroscope mounted at the ends of two different cantilever beams . part (3) presents the theory of the coupled motion of an elastic wing upon which a gyroscope is mounted to simulate an engine-propeller system on an airplane . part (4) consists of an example of the theory of part (3), in which, by taking what are thought to be reasonable parameters, results are obtained showing how the whirling of a rigid propeller may materially affect wing normal mode shapes and frequencies .

.I 43

.T

the relation between wall temperature and the effect

of roughness on boundary layer transition .

.A

potter,j.l. and whitfield,j.d.

.B

j. ae. scs. 28, 1961, 663.

.W

the relation between wall temperature and the effect

of roughness on boundary layer transition .

the experimentally demonstrated rise and subsequent

fall of transition reynolds number with decreasing wall-

to-ambient temperature ratio has been the subject of two recent

notes . in both cases it was argued that the increased

effectiveness of roughness due to wall cooling was not sufficient to

explain the transition-reversal phenomenon on nominally smooth

bodies . in one case, the criterion for transition reversal was

taken to be and in the other values of as low as

eter is a reynolds number formed from velocity and

kinematic viscosity based on calculated conditions at the height of

roughness element k in the undisturbed, laminar boundary layer

at the station of roughness location . the present note is

submitted to show that another method for evaluating the effect

of roughness on transition leads to an opposite conclusion .

.I 44

.T

tip-bluntness effects on cone pressures at m=6.85 .

.A

bertram,m.h.

.B

j.ae.scs. 23, 1956,898.

.W

tip-bluntness effects on cone pressures at m=6.85 .

there is, at present, considerable interest in the characteristies of blunted bodies from both an aerodynamic and a heat-transfer standpoint . the use of blunt shapes is contemplated to reduce the heat-transfer problem at body noses, but there are also applications for blunt noses which occur from mainly aerodynamic considerations . an actual reduction in drag may be the beneficial result of blunting the nose of a cone or a similar slender shape under certain conditions . although the sphere has received considerable treatment, the nose shapes are not necessarily tangent spheres . in the case, let us say, of a total head tube situated in the nose of a given body, the blunting may be quite flat, and nose sections blunter than spherical shape may conceivably be desirable, in some cases, from the heat-transfer standpoint .

the purpose of the present investigation is to examine the aerodynamic effect of a simple type of nose blunting on a basic body .

the incompressible flow of an electrically conducting fluid past a porous plate y = 0 with constant suction velocity in the presence of a transverse uniform strength has recently been investigated by gupta . in this note, the problem is generalized to take into account the effect of free convection, when a body force g per unit mass is acting in the negative x-direction parallel to the wall . the fluid is assumed to be semi-incompressible as usual . in addition to the obvious practical significance, this problem is also interesting in the sense that it provides another exact solution of the magnetohydrodynamic equations, since the only electromagnetic assumptions involved are constant properties and freedom from excessive charges .

.I 45

.T

an investigation of separated flows, part ii: flow

in the cavity and heat transfer .

.A

charwat,a.f.

.B

j. ae. scs. 28, 1961, 513.

.W

an investigation of separated flows, part ii: flow

in the cavity and heat transfer .

the first portion of this paper describes studies of the internal structure of the separated flow in a notch at a free-stream mach number of 3 . observations include.. flow visualization, spark-schlieren pictures of the fluctuations of the free shear layer, and studies of the diffusion of heat from sources placed in the separated region . the second part describes measurements of local heat transfer to the wall .

the external mach number, the length-to-depth ratio of the cavity, the ratio of the oncoming boundary layer thickness to the notch depth (in the turbulent flow region), the thermal to-momentum thickness ratio of the boundary layer and, finally, the geometry of the internal boundary of the separated region are varied as systematically as possible . on the basis of these observations, a simple model of the flow in and the heat transfer across the separated region is formulated .

.I 46

.T

some comments on the inversion of certain large matrices .

.A

bertram klein

.B

convair, a division of general dynamics corp., san diego, calif.

.W

some comments on the inversion of certain large matrices .

the subject of matric structural analysis has been treated in two

recently published papers in the journal . the authors of these papers

have made a number of statements about the inversion of certain large

matrices . it is the purpose of this note to bring to the attention of

the reader certain facts that shed new light on this important problem .

it is shown here that the situation is not as hopeless as the above-

mentioned authors intimate .

.I 47

.T

analysis of low-aspect-ratio aircraft structures .

.A

samson,s.h. and bergmann,h.w.

.B

j. ae. scs. 27, 1960, 679.

.W

analysis of low-aspect-ratio aircraft structures .

two methods are presented for the analysis of complex low-

aspect-ratio aircraft structures . both methods provide for arbitrary external loading, are general with respect to the orientation of structural members, and permit arbitrary boundary conditions . for purposes of analysis a structure is idealized as a network of flexural members with interconnected torsion boxes .

in the first method, sets of linear equations are obtained by expressing boundary conditions, member deflection equations, equilibrium requirements, and slope-compatibility relationships in terms of deflections and internal forces . the solution for deflections and internal forces is then formed as the product of an inverse structural matrix and a column matrix of load functions .

in the second method, the conditions at a given boundary are assembled as a column matrix and are transferred in a step by-step fashion over the entire structure to an opposite boundary . the transfer is accomplished by successive multiplications of square matrices composed independently for the different transfer ranges . the final operation is the inversion of a relatively small matrix and provides the solution for the unknown boundary conditions .

comparisons of theoretical results with experimental data and electric-analog solutions are favorable .

.I 48

.T

supersonic flow at the surface of a circular cone at angle of attack .

.A

willett,j.e.

.B

j. ae. scs. 27, 1960, 907.

.W

supersonic flow at the surface of a circular cone at

angle of attack .

formulas for the inviscid flow properties on the surface of a cone at angle of attack are derived for use in conjunction with the m.i.t. cone tables . these formulas are based upon an entropy distribution on the cone surface which is uniform and equal to that of the shocked fluid in the windward meridian plane . they predict values for the flow variables which may differ significantly from the corresponding values obtained directly from the cone tables . the differences in the magnitudes of the flow variables computed by the two methods tend to increase with increasing free-stream mach number, cone angle and angle of attack .

.I 49

.T

temperature and velocity profiles in the compressible

laminar boundary layer with arbitrary distribution

of surface temperature .

.A

chapman,d. and rubesin,m.

.B

j. ae. scs. 16, 1949, 547.

.W

temperature and velocity profiles in the compressible

laminar boundary layer with arbitrary distribution of surface temperature .

an analysis is presented which enables the temperature profiles, veiocity profiles, heat transfer, and skin friction to be calculated for laminar flow over a two-dimensional or axially symmetric surface without pressure gradient but with an arbitrary analytic distribution of surface temperature . the general theory is applicable to a gas of any prandtl number, although the numerical results given herein have been computed for air . the predictions of the theory for the special case of constant surface temperature are compared with the calculations of crocco . on the basis of this comparison, it is inferred that the present theory enables heat-transfer and skin-friction calculations accurate to within about 5 per cent to be made for flight conditions up to mach numbers near 5 and to within about 1 or 2 per cent for supersonic wind-tunnel conditions up to considerably higher mach numbers .

a particular effort has been made to present the results, which are simple considering their generality, in a form that can be used readily in practical applications . from the mathematical point of view, the theory is applicable to an arbitrary analytic distribution of surface temperature, but in any given practical case it is necessary that the surface-temperature distribution be approximated by a polynomial . the only unknowns in the final equations developed are the coefficients of this polynomial, so that the work involved in applying the theory in any given case depends entirely on the work involved in approximating a given surface-temperature distribution by a polynomial .

an example is worked out in detail which illustrates some of the
principal effects of variable surface temperature . it is shown
that both positively infinite and negatively infinite heat-transfer
coefficients can occur . the anomaly of infinite and negative
heat-transfer coefficients is discussed and attributed to the
customary definition of the heat-transfer coefficient, which is shown
to be fundamentally inappropriate for flows with variable surface
temperature . in the particular example considered, a
conventional method for calculating the net heat transferred yields
completely incorrect results . a brief qualitative discussion of the
possible effects of the heat transfer on flow separation is given .
in order to facilitate the use of the results, all of the principal
equations developed are collected and summarized in the section
entitled /practical use of results ./

.I 50

.T

investigation of laminar boundary layer in compressible
fluids using the crocco method .

.A

van driest,e.r.

.B

naca tn.2597, 1952.

.W

investigation of laminar boundary layer in compressible
fluids using the crocco method .

in the present investigation of the flow of air in a thin laminar
boundary layer on a flat plate, the crocco method has been used to solve

the simultaneous differential equations of momentum and energy involved

in such flow . the crocco method was used because it gave accurate

results for arbitrary prandtl number near unity . the prandtl number

was taken at 0.75, the specific heat was held constant, and the

sutherland law of viscosity-temperature variation was assumed to

represent the viscosity data starting with an initial ambient

temperature of -67.6 f . the main results presented here are the

skin-friction and heat-transfer coefficients as functions of reynolds number,

mach number, and wall-to-free-stream temperature ratio . variations of

shear, velocity, temperature, and mach number across the boundary layer

are included . the crocco method is discussed in detail .

.I 51

.T

theory of aircraft structural models subjected to aerodynamic

heating and external loads .

.A

o'sullivan,w.j.

.B

naca tn.4115, 1957.

.W

theory of aircraft structural models subjected to aerodynamic

heating and external loads .

the problem of investigating the simultaneous effects of transient

aerodynamic heating and external loads on aircraft structures for the

purpose of determining the ability of the structure to withstand flight

to supersonic speeds is studied . by dimensional analyses it is shown

that ..

constructed of the same materials as the aircraft will be thermally

similar to the aircraft with respect to

the flow of heat through the structure

will be similar to those of the aircraft when the structural model is

constructed at the same temperature as the aircraft .

external loads will be similar to those of the aircraft .

subjected to heating and cooling that correctly simulate the aerodynamic

heating of the aircraft, except with respect to angular velocities and

angular accelerations, without requiring determination of the heat flux

at each point on the surface and its variation with time .

acting on the aerodynamically heated structural model to those acting

on the aircraft is determined for the case of zero angular velocity and

zero angular acceleration, so that the structural model may be subjected

to the external loads required for simultaneous simulation of stresses

and deformations due to external loads .

.I 52

.T

procedure for calculating flutter at high supersonic

speed including camber deflections, and comparison

with experimental results .

.A

morgan,h.g.

.B

naca tn.4335, 1958.

.W

procedure for calculating flutter at high supersonic

speed including camber deflections, and comparison with experimental results .

a method which may be used at high supersonic mach numbers is described for calculating the flutter speed of wings having camber in their deflection modes . the normal coupled vibration modes of the wing are used to derive the equations of motion . chord deflections of the vibration modes are approximated by polynomials . the wing may have a control surface and may carry external stores although no aerodynamic forces on the stores are presented . the aerodynamic forces that are assumed to be acting on the wing are obtained from piston theory and also from a quasi-steady form of a theory for two-dimensional steady flow . airfoil shape and thickness effects are taken account of in the analysis .

the method is used to calculate the flutter speed of some wings which had been previously tested at mach numbers of 1.3 to 3.0 . comparison of the calculations and experiment is made for flat-plate 60 and 45 delta wings and also for an untapered 45 sweptback wing .

.I 53

.T

transition reynolds numbers of separated flows at
supersonic speeds .

.A

larson,h.k. and keating,s.j.

.B

nasa tn.d349, 1960.

.W

transition reynolds numbers of separated flows at
supersonic speeds .

experimental research has been conducted on the effects of wall cooling, mach number, and unit reynolds number on the transition reynolds number of cylindrical separated boundary layers on an ogive-cylinder model . results were obtained from pressure and temperature measurements and shadowgraph observations . the maximum scope of measurements encompassed mach numbers between 2.06 and 4.24, reynolds numbers (based on length of separation) between 60,000 and 400,000, and ratios of wall temperature to adiabatic wall temperature between 0.35 and 1.0 . within the range of the present tests, the transition reynolds number was observed to decrease with increasing wall cooling, increase with increasing mach number, and increase with increasing unit reynolds number . the wall-cooling effect was found to be four times as great when the attached boundary layer upstream of separation was cooled in conjunction with cooling of the separated boundary layer as when only the separated boundary layer was cooled . wall cooling of both the attached and separated flow regions also caused, in some cases, reattachment in the otherwise separated region . cavity resonance present in the separated region for some model configurations was accompanied by a large decrease in transition reynolds number at the lower test mach numbers .

.I 54

.T

method for calculation of compressible laminar boundary

layer characteristics in axial pressure gradient with

zero heat transfer .

.A

morduchow,m. and clarke,j.h.

.B

naca tn.2784, 1952.

.W

method for calculation of compressible laminar boundary

layer characteristics in axial pressure gradient with

zero heat transfer .

the karman-pohlhausen method is extended primarily to sixth-degree velocity profiles for determining the characteristics of the compressible laminar boundary layer over an adiabatic wall in the presence of an axial pressure gradient . it is assumed that the prandtl number is unity and that the coefficient of viscosity varies linearly with the temperature . a general approximate solution which permits a rapid determination of the boundary-layer characteristics for any given free-stream mach number and given velocity distribution at the outer edge of the boundary layer is obtained . numerical examples indicate that this solution will in practice lead to results of satisfactory accuracy, including the critical reynolds number for stability . for the special purpose of calculating the location of the separation point in an adverse pressure gradient, a short and simple method, based on the use of a seventh-degree velocity profile, is derived . the numerical example given here indicates that this method should in practice lead to sufficiently accurate results . for the special case of flow near a forward stagnation point it is shown that the karman-pohlhausen method with the usual fourth-degree profiles leads to results of adequate accuracy, even for the critical reynolds number .

.I 55

.T

separation, stability and other properties of compressible laminar boundary layer with pressure gradient and heat transfer .

.A

morduchow,m. and grape,r.g.

.B

naca tn.3296, 1955.

.W

separation, stability and other properties of compressible laminar boundary layer with pressure gradient and heat transfer .

a theoretical study is made of the effect of pressure gradient, wall temperature, and mach number on laminar boundary-layer characteristics and, in particular, on the skin-friction and heat-transfer coefficients, on the separation point in an adverse pressure gradient, on the wall temperature required for complete stabilization of the laminar boundary layer, and on the minimum critical reynolds number for laminar stability . the prandtl number is assumed to be unity and the coefficient of viscosity is assumed to be proportional to the temperature, with a factor arising from the sutherland relation . a simple and

accurate method of locating the separation point in a compressible flow

with heat transfer is developed . numerical examples to illustrate the

results in detail are given throughout .

.I 56

.T

an analysis of the applicability of the hypersonic

similarity law to the study of the flow about bodies

of revolution at zero angle of attack .

.A

ehret,d.m.

.B

naca tn.2250, 1950.

.W

an analysis of the applicability of the hypersonic

similarity law to the study of the flow about bodies

of revolution at zero angle of attack .

the hypersonic similarity law as derived by tsien has been

investigated by comparing the pressure distributions along bodies of

revolution at zero angle of attack . in making

these comparisons, particular

attention was given to determining the limits of mach number and fineness

ratio for which the similarity law applies . for the purpose of this

investigation, pressure distributions

determined by the method of

characteristics for ogive cylinders for

values of mach numbers and fineness

ratios varying from 1.5 to 12 were compared .

pressures on various cones

and on cone cylinders were also compared in this study .

the pressure distributions presented demonstrate that the hypersonic

similarity law is applicable over a

wider range of values of mach numbers

and fineness ratios than might be expected from the assumptions made in

the derivation . this is significant since within the range of

applicability of the law a single pressure

distribution exists for all similarly

shaped bodies for which the ratio of

free-stream mach number to fineness

ratio is constant . charts are presented

for rapid determination of

pressure distributions over ogive cylinders for any combination of mach

number and fineness ratio within defined limits .

.I 57

.T

applicability of the hypersonic similarity rule to

pressure distributions which include the effects of

rotation for bodies of revolution at zero angle of

attack .

.A

rossow,v.j.

.B

naca tn.2399, 1951.

.W

applicability of the hypersonic similarity rule to

pressure distributions which include the effects of

rotation for bodies of revolution at zero angle of

attack .

the analysis of technical note 2250, 1950, is extended to include

the effects of flow rotation . it is

found that the theoretical pressure

distributions over ogive cylinders can be related by the hypersonic

similarity rule with sufficient accuracy for most engineering purposes .

the error introduced into pressure distributions and drag of ogive

cylinders by ignoring the rotation term in the characteristic equations

is investigated . it is found that

the influence of the rotation term on

pressure distribution and drag depends only upon the similarity

parameter k (mach number divided by fineness ratio) .

although the error in

drag, due to neglect of the rotation term, is negligible at k=0.5, the

error is about 30 percent at k=2.0 .

charts are presented for the rapid determination of pressure

distributions for rotational flow over

ogive cylinders for all values of

the similarity parameter between 0.5 and

of mach number and fineness ratio .

.I 58

.T

pressure measurements on sharp and blunt 5 and 15 half-angle

cones at mach number 3.86 and angles of attack to

100 .

.A

amick,j.l.

.B

nasa tn.d753, 1961.

.W

pressure measurements on sharp and blunt 5 and 15 half-angle

cones at mach number 3.86 and angles of attack to

100 .

measured pressure distributions on cones are compared with modified

newtonian theory . deviations as large as 14 percent of the stagnation

pressure behind a normal shock are found .

by combining empirical results

for cylinders normal to the flow with

newtonian concepts, a method of

calculating pressures on cones at high angles

of attack is developed .

calculations by this method differ from the

experimental results on sharp cones

by only 2 percent of the stagnation

pressure behind a normal shock . for

blunted cones, additional deviations

up to 8 percent are noted near the

nose .

schlieren pictures of the flow show an attached shock on the sharp

of attack . detachment of the shock

appears to be associated with the

attainment of sonic speed immediately

behind the shock .

an orifice size effect is found which can increase the indicated

pressure above the true value, if

the orifice width is greater than

one-tenth the local radius of curvature .

.I 59

.T

tables of exact laminar-boundary layer solutions when

the wall is porous and fluid properties are variable .

.A

brown,w.d. and donoughe,p.l.

.B

naca tn.2479, 1951.

.W

tables of exact laminar-boundary layer solutions when

the wall is porous and fluid properties are variable .

the three partial differential equations of the laminar boundary

layer for two-dimensional steady-state compressible flow have been

transformed into two ordinary differential equations by the method of

pohlhausen, falkner, and skan . the ordinary equations include

parameters for expressing the simultaneous effects of pressure gradient in

the main-stream flow through a porous wall and property changes in the

fluid due to large temperature differences between the wall and the

free stream .

a total of 58 cases have been solved numerically by the method of

picard . the euler number (nondimensional pressure-gradient parameter) ranges in value from 1 (stagnation-point value) to the negative values found at the laminar separation points . three rates of flow through the porous wall were considered (including the impermeable case where the flow rate is 0) . five temperature ratios (stream temperature divided by wall temperature) were used .. the uncooled and unheated case (temperature ratio of 1), two cooled cases (temperature ratios of ture ratios of and) . velocity, weight-flow, and temperature distributions are tabulated as are the dimensionless stream function of falkner and skan and its derivatives and the dimensionless temperature function of pohlhausen and its derivatives .

for each case, displacement, momentum, and convection thicknesses, as well as nusselt number and coefficient of friction at the wall, were computed .

.I 60

.T

estimation forces and moments due to rolling for several slender tail configurations at supersonic speeds .

.A

bobbitt,p.j. and malvestuto,f.s.

.B

naca tn.2955, 1953.

.W

estimation forces and moments due to rolling for several slender tail configurations at supersonic speeds .

the velocity potentials, span loadings, and corresponding force and moment derivatives have been theoretically evaluated for a number of slender-tail arrangements performing a steady rolling motion at supersonic speeds .

the method of analysis is based upon an application of

conformal-transformation techniques . the utilization of these techniques allows

the simple determination of the complex potentials for various types

of two-dimensional boundary-value problems .

in addition, two simple and often-used approximations to the

rolling derivatives have been compared with the corresponding exact

values determined by the method presented in this report .

in order to show the importance of wing-tail interference, the

effect of the flow field behind a rolling wing on the tail

characteristics has been illustrated for a simple wing-tail arrangement .

.I 61

.T

on flow of electrically conducting fluids over a flat

plate in the presence of a transverse magnetic field .

.A

rossow,v.j.

.B

naca tn.3971, 1957.

.W

on flow of electrically conducting fluids over a flat

plate in the presence of a transverse magnetic field .

the use of a magnetic field to control the motion of electrically

conducting fluids is studied . the boundary-layer solutions are found

for flow over a flat plate when the magnetic field is fixed relative to

the plate or to the fluid . the

equations are integrated numerically for the effect of the transverse magnetic

field on the velocity and temperature

profiles, and hence, the skin friction and rate of heat transfer .

it is concluded that the skin friction and the heat-transfer rate are

reduced when the transverse magnetic

field is fixed relative to the plate

and increased when fixed relative to the fluid . the total drag is

increased in all the cases studied .

.I 62

.T

similar solutions for the compressible laminar boundary

layer with heat transfer and pressure gradient .

.A

cohen,c.b. and reshotko,e.

.B

naca tn.3325, 1955.

.W

similar solutions for the compressible laminar boundary

layer with heat transfer and pressure gradient .

stewartson's transformation is applied to the laminar compressible

boundary-layer equations and the

requirement of similarity is introduced,

resulting in a set of ordinary nonlinear differential equations

previously quoted by stewartson, but unsolved . the requirements of the

system are .. prandtl number of 1.0, linear viscosity-temperature

relation across the boundary layer, an

isothermal surface, and the particular

distributions of free-stream velocity
consistent with similar solutions .
this system admits axial pressure
gradients of arbitrary magnitude, heat
flux normal to the surface, and arbitrary mach numbers .
  the system of differential equations is transformed to an integral
system, with the velocity ratio as
the independent variable . for this
system, solutions are found for pressure gradients varying from that
causing separation to the infinitely favorable gradient and for wall
temperatures from absolute zero to twice the free-stream stagnation
temperature . some solutions for separated flows are also presented .
  for favorable pressure gradients, the solutions are unique . for
adverse pressure gradients, where the solutions are not unique, two
solutions of the infinite family of possible solutions are identified as
essentially viscid at the outer edge of the boundary layer and the
remainder essentially inviscid . for
the case of favorable pressure gradients
with heated walls, the velocity within
a portion of the boundary layer is
shown to exceed the local external velocity .
the variation of a reynolds
analogy parameter, which indicates the ratio of skin friction to heat
transfer, is from zero to 7.4 for a surface of temperature twice the
free-stream stagnation temperature, and from zero to 2.8 for a surface
held at absolute zero where the value 2 applies to a flat plate .
.I 63

.T

hypersonic viscous flow over slender cones .

.A

talbot,l.

.B

naca tn.4327, 1958.

.W

hypersonic viscous flow over slender cones .

viscous self-induced pressures on 3 -semivertex-angle cones were

measured over the range 3.7 free-stream mach number 5.8 and 0.5

viscous-interaction parameter 2.3 . the data were found to be in good

agreement with results obtained by talbot on 5 cones in the range

rameter 3.5 . all these data were correlated reasonably well by the

viscous-interaction parameter, which is defined as

where and are the mach number and reynolds number based on

ideal taylor-maccoll flow conditions and c is the chapman-rubesin

factor .

a new method for calculating self-induced pressures is presented

which takes into account the interaction between boundary-layer growth

and the inviscid-flow field at the outer edge of the boundary layer .

pressures calculated by this method were only 10 to 20 percent higher

than the measured values .

.I 64

.T

unsteady oblique interaction of a shock wave with plane

disturbances .

.A

moore,f.k.

.B

naca tn.2879, 1953.

.W

unsteady oblique interaction of a shock wave with plane

disturbances .

analysis is made of the flow field produced by oblique impingement

of weak plane disturbances of arbitrary

profile on a plane normal shock .

three types of disturbance are considered ..

moves . the sound wave refracts either

as a simple isentropic sound wave

or as an attenuating isentropic pressure wave, depending on the angle

between the shock and the incident

sound wave . a stationary vorticity

wave of constant pressure appears behind the shock .

reflects as a sound wave, and a stationary vorticity wave is produced .

the shock . the incident wave refracts as a stationary vorticity wave,

and either a sound wave or attenuating pressure wave is also produced .

computations are presented for the first two types of incident wave,

over the range of incidence angles, for shock mach numbers of 1, 1.5,

and .

.I 65

.T

convection of a pattern of vorticity through a shock

wave .

.A

ribner,h.s.

.B

naca tn.2864, 1953.

.W

convection of a pattern of vorticity through a shock

wave .

an arbitrary weak spatial distribution of vorticity can be

represented in terms of plane sinusoidal shear waves of all orientations and

wave lengths (fourier integral) . the analysis treats the passage of a

single representative weak shear wave through a plane shock and shows

refraction and modification of the shear wave with simultaneous

generation of an acoustically intense sound

wave . applications to turbulence

and to noise in supersonic wind tunnels are indicated .

.I 66

.T

some effects of joint conductivity on the temperature

and thermal stresses in aerodynamically heated skin-stiffener

combinations .

.A

griffith,g.e. and miltonberger,g.h.

.B

naca tn.3609, 1956.

.W

some effects of joint conductivity on the temperature

and thermal stresses in aerodynamically heated skin-stiffener

combinations .

temperatures and thermal stresses in typical skin-stiffener

combinations of winglike structures subjected to aerodynamic heating have

been obtained with the aid of an electronic differential analyzer .

variations were made in an aerodynamic

heat-transfer parameter, in a joint

conductivity parameter, and in the ratio

of skin width to skin thickness .

the results, which are presented in nondimensional form, indicate that

decreasing the joint conductivity parameter lowers both the interior

and the average temperature ratios, increases the peak thermal stress

ratios in the skin, and may considerably increase the peak stiffener

stress ratios,. increasing the aerodynamic heat-transfer parameter

decreases the interior and average temperature ratios, increases the

peak skin stress ratios somewhat,

but greatly increases the peak

stiffener stress ratios,. and increasing the ratio of skin width to skin

thickness produces only moderate decreases in the peak skin stress

ratios while moderately increasing the peak stiffener stress ratios .

.I 67

.T

dynamic stability of vehicles traversing ascending

or descending paths through the atmosphere .

.A

tobak and allen.

.B

naca tn.4275, 1958.

.W

dynamic stability of vehicles traversing ascending

or descending paths through the atmosphere .

an analysis is given of the oscillatory motions of vehicles which

traverse ascending and descending paths through the atmosphere at high

speed . the specific case of a skip path is examined in detail, and

this leads to a form of solution for the oscillatory motion which should

recur over any trajectory . the distinguishing feature of this form is

the appearance of the bessel rather than the trigonometric function as

the characteristic mode of oscillation .

.I 68

.T

some aspects of air-helium simulation and hypersonic

approximations .

.A

love,e.s.

.B

nasa tn.d49, 1959.

.W

some aspects of air-helium simulation and hypersonic

approximations .

some illustrations of the differences that may be expected between

results obtained in hypersonic wind tunnels that employ air and results

obtained in those that employ helium as the test medium (imperfect-gas

effects are not considered) are compiled and presented herein . simple

expressions are presented that demonstrate the possibility of simulating

air results in helium tests and of transforming helium data to

equivalent air data . nonviscous and viscous simulations are considered . in most cases, the methods and the general forms of the expressions for

simulation that are derived are applicable to any two ideal gases having

different ratios of specific heats .

.I 69

.T

predicted shock envelopes about two types of vehicles

at large angles of attack .

.A

kaattari,g.e.

.B

nasa tn.d860, 1961.

.W

predicted shock envelopes about two types of vehicles

at large angles of attack .

methods based on oblique- and normal-shock relationships and the

continuity of mass flow through suitably chosen volume elements between

the shock and body were developed to predict shock envelopes about two

types of vehicles being considered for atmosphere entry . one type is a

high-drag capsule shape . the other type is essentially a slender

triangular wing capable of providing high lift or high drag, depending on

the angle of attack . predicted and measured shock envelopes were

compared for a mach number range of 3 to 15 for vehicles at high angles of

attack,. good agreement was found . most of the available experimental

data were in a speed and temperature range in which no important

real-gas effects occurred .

.I 70

.T

a study of flow changes associated with airfoil section

drag rise at supercritical speeds .

.A

nitzburg,g.e. and crandall,s.

.B

naca tn.1813, 1949.

.W

a study of flow changes associated with airfoil section

drag rise at supercritical speeds .

a study of experimental pressure distributions and section characteristics for several moderately thick airfoil sections was made . a correlation appears to exist between the drag-divergence mach number and the free-stream mach number for which sonic velocity occurs at the airfoil crest, the chordwise station at which the airfoil surface is tangent to the free-stream direction . it was found that, since the mach number for which sonic velocity occurs at the airfoil crest can be estimated satisfactorily by means of the prandtl-glauert rule, a method is provided whereby the drag-divergence mach number of an airfoil section at a given angle of attack can be estimated from the low-speed pressure distribution and the airfoil profile . this method was used to predict with a reasonable degree of accuracy the drag-divergence mach number of a considerable number of airfoil sections having diverse shapes and a wide range of thickness-chord ratios .

the pressure distributions and section force characteristics of several moderately thick airfoil sections at mach numbers above the drag-divergence mach number were analyzed . some of the characteristics of the flow over these airfoils at supercritical mach numbers are discussed .

.I 71

.T

laminar boundary layer behind shock advancing into

stationary fluid .

.A

mirels,h.

.B

naca tn.3401, 1955.

.W

laminar boundary layer behind shock advancing into

stationary fluid .

a study was made of the laminar compressible boundary layer induced

by a shock wave advancing into a stationary fluid bounded by a wall .

for weak shock waves, the boundary layer is identical with that which

occurs when an infinite wall is impulsively set into uniform motion

shocks .

velocity and temperature profiles, recovery factors, and

skin-friction and heat-transfer coefficients are tabulated for a wide range

of shock strengths .

.I 72

.T

boundary layer behind shock or thin expansion wave

moving into stationary fluid .

.A

mirels,h.

.B

naca tn.3712, 1956.

.W

boundary layer behind shock or thin expansion wave

moving into stationary fluid .

the boundary layer behind a shock or thin expansion wave advancing into a stationary fluid has been determined . laminar and turbulent boundary layers were considered . the wall surface temperature behind the wave was also investigated . the assumption of a thin expansion wave is valid for weak expansions but becomes progressively less accurate for strong expansion waves .

the laminar-boundary-layer problem was solved by numerical integration except for the weak wave case, which can be solved analytically . integral (karman-pohlhausen type) solutions were also obtained to provide a guide for determining expressions which accurately represent the numerical data . analytical expressions for various boundary-layer parameters are presented which agree with the numerical integrations within 1 percent .

the turbulent-boundary-layer problem was solved using integral methods similar to those employed for the solution of turbulent compressible flow over a semi-infinite flat plate . the fluid velocity, relative to the wall, was assumed to have a seventh-power profile . the blasius

equation, relating turbulent skin friction

and boundary-layer thickness, was

utilized in a form which accounted for compressibility .

  consideration of the heat transfer to the wall permitted the wall

surface temperature, behind the wave,

to be determined . the wall

thickness was assumed to be greater than the

wall thermal-boundary-layer

thickness . it was found that the wall

temperature was uniform (as a

function of distance behind the wave)

for the laminar-boundary-layer case

but varied with distance for the turbulent-boundary-layer case .

.I 73

.T

investigation of the stability of the laminar boundary

layer in a compressible fluid .

.A

lees,l. and lin,c.c.

.B

naca tn.1115, 1946.

.W

investigation of the stability of the laminar boundary

layer in a compressible fluid .

  in the present report the stability of two-dimensional laminar

flows of a gas is investigated by the method of small perturbations .

the chief emphasis is placed on the case of the laminar boundary layer .

part 1 of the present report deals with the general mathematical theory . the general equations governing one normal mode of the small velocity and temperature disturbances are derived and studied in great detail . it is found that for reynolds numbers of the order of those encountered in most aerodynamic problems, the temperature disturbances have only a negligible effect on those particular velocity solutions which depend primarily on the viscosity coefficient (/viscous solutions/) . indeed, the latter are actually of the same form in the compressible fluid as in the incompressible fluid, at least to the first approximation . because of this fact, the mathematical analysis is greatly simplified . the final equation determining the characteristic values of the stability problem depends on the /inviscid solutions/ and the function of tietjens in a manner very similar to the case of the incompressible fluid . the second viscosity coefficient and the coefficient of heat conductivity do not enter the problem,. only the ordinary coefficient of viscosity near the solid surface is involved .

part 2 deals with the limiting case of infinite reynolds numbers . the study of energy relations is very much emphasized . it is shown that the disturbance will gain energy from the main flow if the gradient of the product of mean density and mean vorticity near the solid surface has a sign opposite to that near the outer edge of the boundary layer .

a general stability criterion has been obtained in terms of the gradient of the product of density and vorticity, analogous to the rayleigh-tollmien criterion for the case of an incompressible fluid . if this gradient vanishes for some value of the velocity ratio of the main flow exceeding 1-1/m (where m is the free stream mach number) .

.I 74

.T

an experimental study of the turbulen coundary layer

on a shock tube wall .

.A

gooderum,p.n.

.B

naca tn.4243, 1958.

.W

an experimental study of the turbulen coundary layer

on a shock tube wall .

interferometric measurements were made of the density profiles of

an unsteady turbulent boundary layer on the flat wall of a shock tube .

the investigation included both subsonic and supersonic flow (mach

numbers of 0.50 and 1.77) with no pressure gradient and with heat transfer

to a cold wall . velocity profiles and average skin-friction

coefficients were calculated . effects on the velocity profile of

surface roughness and flow length are examined .

.I 75

.T

studies of structural failure due to acoustic loading .

.A

hess,n.w.

.B

naca tn.4050, 1957.

.W

studies of structural failure due to acoustic loading .

some discussion of the acoustic fatigue problem of aircraft structures is given along with data pertaining to the acoustic inputs from some powerplants in common use . comparisons are given for results of some fatigue tests of flat panels and cantilever beams exposed to both random- and discrete-type inputs . in this regard it appears that both the stress level of the test and the type of model are significant,. hence, no generalization can be made at this time . with regard to increasing the fatigue life, it was noted that increased stiffening of a panel due to curvature and pressure differential is particularly beneficial .

.I 76

.T

flight measurement of wall pressure fluctuations and boundary-layer turbulence .

.A

mull,h.r. and algranti,j.s.

.B

nasa tn.d280, 1960.

.W

flight measurement of wall pressure fluctuations and boundary-layer turbulence .

the results are presented for a flight test program using a fighter type jet aircraft flying at pressure altitudes of 10,000, 20,000, and apparatus was used to measure and record the output of microphones and hot-wire anemometers mounted on the forward-fuselage section and wing of the airplane . mean-velocity profiles in the boundary layers were obtained from total-pressure measurements .

the ratio of the root-mean-square fluctuating wall pressure to the free-stream dynamic pressure is presented as a function of reynolds number and mach number . the longitudinal component of the turbulent-velocity fluctuations was measured, and the turbulence-intensity profiles are presented for the wing and forward-fuselage section .

in general, the results are in agreement with wind-tunnel measurements which have been reported in the literature . for example, the variation of (is the root mean square of the wall-pressure fluctuation, and q is the free-stream dynamic pressure) with reynolds number was found to be essentially constant for the forward fuselage-section boundary layer, while variations at the wing station were probably unduly affected by the microphone diameter, which was large compared with the boundary-layer thickness .

.I 77

.T

a comparative analysis of the performance of long range hypervelocity vehicles .

.A

eggers,a.j.

.B

naca tn.4046, 1957.

.W

a comparative analysis of the performance of long range hypervelocity vehicles .

long-range hypervelocity vehicles are studied in terms of their motion in powered flight, and their motion and aerodynamic heating in

unpowered flight . powered flight is
analyzed for an idealized propulsion
system which rather closely approaches
present-day rocket motors .
unpowered flight is characterized by a return
to earth along a ballistic, skip,
or glide trajectory . only those
trajectories are treated which yield the
maximum range for a given velocity at the end of powered flight .
aerodynamic heating is treated in a manner
similar to that employed previously
by the senior authors in studying ballistic missiles (naca tn 4047),
with the exception that radiant as well as convective heat transfer is
considered in connection with glide and skip vehicles .
  the ballistic vehicle is found to be the least efficient of the
several types studied in the sense
that it generally requires the highest
velocity at the end of powered flight in order to attain a given range .
this disadvantage may be offset, however, by reducing convective heat
transfer to the re-entry body through
the artifice of increasing pressure
drag in relation to friction drag - that
is, by using a blunt body . thus
the kinetic energy required by the vehicle at the end of powered flight
may be reduced by minimizing the mass of coolant material involved .
  the glide vehicle developing lift-drag ratios in the neighborhood
of and greater than 4 is far superior

to the ballistic vehicle in ability

to convert velocity into range . it has the disadvantage of having far

more heat convected to it,. however, it has the compensating advantage

that this heat can in the main be radiated

back to the atmosphere .

consequently, the mass of coolant material may be kept relatively low .

the skip vehicle developing lift-drag ratios from about 1 to 4 is

found to be superior to comparable ballistic and glide vehicles in

converting velocity into range . at

lift-drag ratios below 1 it is found to

be about equal to comparable ballistic

vehicles while at lift-drag ratios

.I 78

.T

an analytical treatment of aircraft propeller precession

instability .

.A

reed,w.h. and bland,s.r.

.B

nasa tn.d659, 1961.

.W

an analytical treatment of aircraft propeller precession

instability .

an analytical investigation is made of a precession-type instability

which can occur in a flexibly supported aircraft-engine-propeller

combination . by means of an idealized

mathematical model which is comprised

of a rigid power-plant system flexibly

mounted in pitch and yaw to a fixed

backup structure, the conditions required for neutral stability are

determined . the paper also examines the sensitivity of the stability

boundaries to changes in such parameters

as stiffness, damping, and

asymmetries in the engine mount, propeller

speed, airspeed, mach number,

propeller thrust, and location of pitch and yaw axes . stability is found

to depend strongly on the damping and stiffness in the system .

with the use of nondimensional charts theoretical stability

boundaries are compared with experimental results obtained in wind-tunnel

tests of an aeroelastic airplane model . in general, the theoretical

results, which do not account for wing response, show the same trends

as observed experimentally,. however,

for a given set of conditions

calculated airspeeds for neutral stability

are consistently lower than the

measured values . evidently, this result is due to the fact that wing

response tends to add damping to the system .

.I 79

.T

effects of extreme surface cooling on boundary layer

transition .

.A

jack,j.r.

.B

naca tn.4094, 1957.

.W

effects of extreme surface cooling on boundary layer

transition .

an investigation was made to determine the combined effects of surface cooling, pressure gradients, nose blunting, and surface finish on boundary-layer transition . data were obtained for various body shapes at a mach number of 3.12 and reynolds

numbers per foot as high as 15x10 .

previous transition studies, with moderate cooling, have shown agreement with the predictions of stability theory . for surface roughnesses ranging from 4 to 1250 microinches the location of transition was unaffected with moderate cooling . with extreme cooling, an adverse effect was observed for each of the parameters investigated . in general, the transition reynolds number decreased with

decreasing surface temperature .

in particular, the beneficial effects of a favorable pressure gradient obtained with moderate cooling disappear with extreme cooling, and a transition reynolds number lower than

that observed on a cone is obtained .

further, an increase in the nose bluntness decreased the transition reynolds number under conditions of extreme cooling .

.I 80

.T

effect of distributed three-dimensional roughness and

surface cooling on boundary layer transition and lateral

spread of turbulence at supersonic speeds .

.A

braslow,a.l.

.B

nasa tn.d53, 1959.

.W

effect of distributed three-dimensional roughness and

surface cooling on boundary layer transition and lateral

spread of turbulence at supersonic speeds .

an investigation was made in the langley 4 by 4-foot supersonic

pressure tunnel at mach numbers of 1.61 and 2.01 to determine (1) the

effect of distributed roughness on boundary-layer transition with the

model surface at adiabatic wall temperature and cooled and (2) the

effect of surface cooling on the lateral spread of turbulence . both

distributed granular-type and single spherical roughness particles were

used, and transition of the boundary layer was determined by hot-wire

anemometers . the transition-triggering mechanism of the

three-dimensional roughness at supersonic speeds appeared to be the same as

that previously observed at subsonic speeds . in fact, the critical

value of the roughness reynolds number parameter (that is,

the value at which turbulent spots are initiated by the roughness) was

found to be approximately the same at supersonic and subsonic speeds

when complete local conditions at the top of the roughness, including

density and viscosity, were considered in the formulation of the

roughness reynolds number . for three-dimensional roughness at a reynolds

number less than its critical value, the roughness introduced no

disturbances of sufficient magnitude to influence transition . surface

cooling, although providing a theoretical increase in stability to small disturbances, did not increase to any important extent the value of the

critical roughness reynolds number for three-dimensional roughness

particles . cooling, therefore, because of its effect on the

boundary-layer thickness, density, and viscosity actually promoted transition due

to existing three-dimensional surface roughness for given mach and

reynolds numbers . the measured lateral spread of turbulence in the

boundary layer appeared to be unaffected by the increased laminar

stability derived from the surface cooling .

.I 81

.T

compressible laminar flow and heat transfer about a

rotating isothermal disk .

.A

ostrach,s. and thornton,p.

.B

naca tn.4320, 1958.

.W

compressible laminar flow and heat transfer about a

rotating isothermal disk .

the flow and heat transfer about a rotating isothermal disk are

re-examined to include the effects of compressibility and property

variations . if viscous dissipation is neglected,

the compressible problem is

correlated to the incompressible problem by assuming linear variations

of viscosity and thermal conductivity with temperature . certain

inaccuracies in several previous incompressible solutions are noted and

corrected herein . the effect of compressibility appears as a

distortion of the normal coordinate and normal velocity component and as
a multiplicative factor in the heat-transfer coefficient, the nusselt
number, and in the expressions for the skin-friction components and
torque required to rotate the disk .

.I 82

.T

theoretical investigation of the ablation of a glass-type
heat protection shield of varied material properties
at the stagnation point of a re-entering irbm .

.A

adams,e.w.

.B

nasa tn.d564, 1961.

.W

theoretical investigation of the ablation of a glass-type
heat protection shield of varied material properties
at the stagnation point of a re-entering irbm .

the melting-type heat protection at the stagnation point of a
re-entering irbm is treated by employing homogeneous, opaque, and
nondecomposing glass shields which do not exceed a temperature of
some effects due to variations of the glass properties . the ballistic
re-entry vehicle has a nose diameter of 0.635 m, a ballistic factor
of 3.5 x 10, a re-entry angle of 124.9 (from the
vertical) at an altitude of 100 km, and a re-entry speed of 4.5 .
the performance of 36 different glass shields with assumed
combinations of material properties is investigated by employing a
calculation method which yields practically exact, transient solutions

for the problem . as a corollary, results for a certain steady flight

state are also given . the discussions made it possible to derive

under realistic flight conditions some thermal characteristics for the

employment of thin, or light-weight, glass shields .

investigation of these hypothetical glass shields leads to the

conclusion that a low thermal conductivity and a high specific heat,

and thus, a small thermal diffusivity are most desirable . a small

thermal diffusivity yields high surface temperatures, causing a high

radiative heat transfer out of the shield,. and steep temperature

profiles normal to the surface, causing a small thermal penetration across

the shield with little total ablation of the shield . results show that

for the assumed irbm re-entry, the necessary thickness of the employed

glass shields increases monotonically with thermal diffusivity which is

the only material parameter affecting this thickness .

a high viscosity level and a high emissivity constant of the

surface of the supposedly opaque shield are also desirable,. although,

these two properties exert a comparatively small influence on the

overall performance when disregarding glass shields with an extremely

low viscosity level .

.I 83

.T

discussion of solar proton events and manned space

flights .

.A

anderson,k. and sinchtel,c.d.

.B

nasa tn.d671, 1961.

.W

discussion of solar proton events and manned space flights .

as a result of studies made during the international geophysical year (igy) and the international geophysical cooperation (igc), it is known that a considerable fraction of large solar flares give rise to almost pure streams of protons which reach the earth and continue to arrive for as long as 11 days . the energies of these particles lie within a very steep spectrum extending from 20 to least 500 mev . because of the frequency of large flares during times of high solar activity, and owing to the long duration of each solar proton emission, these particles were present in detectable intensity near the top of the earth's atmosphere for about 15 percent of the time from 1957 to 1960 . the number of large flares that accelerated and released these particles during this three-year period was about 30 .

the event that began on august 22, 1958 contributed greatly toward the understanding of the solar and terrestrial sequence of events, and in addition provided the first identification of the emitted particles . a flare on may of protons in the neighborhood of the earth that

this phenomenon was recognized as an
additional radiation hazard to manned vehicles
in the high atmosphere and in most parts
of the solar system . the three very intense
events that occurred in july, 1959 further
supported this conclusion, and the possibility
of predicting such events became an
important consideration . in addition to its value
in the protection of human beings, effective
forecasting clearly would be of great value in
the detailed scientific study of this
phenomenon .

this paper presents a preliminary
discussion of some aspects of predicting the
arrival of protons at the earth following the
appearance of solar activity features and,
equally important, of forecasting the periods
when this penetrating radiation is unlikely
to occur .

.I 84

.T

experimental investigation of the downstream influence
of stagnation point mass transfer .

.A

libby,pa. and cresci,r.j.

.B

j. ae. scs. 28, 1961, 51.

.W

experimental investigation of the downstream influence

of stagnation point mass transfer .

this report presents the results of an experimental

investigation of the downstream influence of localized mass transfer in the

stagnation region of a blunt body under hypersonic flow

conditions . the coolant is injected through a porous plug coaxial

with the centerline of symmetry of the model . the tests were

carried out in a wind tunnel with a mach number of 6.0,

stagnation temperatures of approximately 1,600 r., and a stagnation

pressure of approximately 600 psia . four different gases were

injected over a range of mass flows . the heat transfer on the

impermeable section was measured under isothermal wall

conditions,. for the higher rates of mass flow, adiabatic surface

temperatures were also determined . the theoretical analysis of the

boundary-layer flow is investigated in order to establish the

similarity parameters for the flow system . these parameters permit

the extrapolation of the test results to other flow conditions,

provided that laminar flow prevails . helium is found to be the

most efficacious coolant .

.I 85

.T

on trails of axisymmetric hypersonic blunt bodies flying

through the atmosphere .

.A

feldman,s.

.B

j. ae. scs. 28, 1961, 433.

.W

on trails of axisymmetric hypersonic blunt bodies flying through the atmosphere .

the trail left in the atmosphere by a body moving at hypersonic speeds is the subject of theoretical treatment . the times required for ionization and dissociation (and their inverse processes) to go to completion, when compared to the flow times of a gas particle, are important in determining the observable effects of hypersonic trails-i.e., emitted thermal radiation and reflection of electromagnetic waves from the trail .

in order to simplify the theoretical treatment, the trail is divided into two regions .. (1) the expansion-controlled trail, which treats the behavior of the wake behind the body up to a point, along the direction of flight, where the pressure decays to the free-stream value and cooling is controlled principally by the expansion of the flow, and (2) the conduction-controlled trail, where the trail cools mainly by diffusion of heat away from the high-temperature core .

the influence of the details of the body shape on the observables are discussed and a simple computational procedure for the behavior of the conduction-controlled trail is developed based on integral methods . results of calculations that assume thermodynamic equilibrium of the flow field give the values of the thermodynamic variables in the trail of a sphere, axial distributions of emitted thermal radiation, and maps of electron density distribution . it is shown that the cooling of the conduction-controlled trail is essentially due to conduction of heat and that

viscous effects are not important . it is found that this portion

of the trail does not widen as one proceeds downstream . flight

velocities considered vary between 15,000 and 35,000 ft sec and

altitudes range between 100,000 and 250,000 ft .

.I 86

.T

inviscid-incompressible flow theory of static peripheral

jets in proximity to the ground .

.A

strand,t.

.B

j. ae. scs. 1961, 27.

.W

inviscid-incompressible flow theory of static peripheral

jets in proximity to the ground .

an /exact/ flow theory of peripheral jets issuing

symmetrically from a hovering aerial-ground vehicle is presented . the

theory is exact insofar as no simplifying assumptions have been

made in obtaining a solution of the governing inviscid,

two-dimensional hydrodynamical flow equations . the results are

valid for all jet thickness vehicle height ratios . the limit of

applicability of existing theories (very low thickness height

ratios) are defined . jet reaction, lift, and power coefficients for

static conditions are introduced and computed . lift

augmentation and lift power ratios are also calculated .

applications to three-dimensional vehicles with rotational symmetry are indicated .

.I 87

.T

free-convection magnetohydrodynamic flow past a porous flat plate .

.A

pau-chang lu

.B

graduate assistant, department of mechanical engineering, case

institute of technology, cleveland, ohio

.W

free-convection magnetohydrodynamic flow past a porous flat plate .

the incompressible flow of an electrically conducting fluid past a

porous plate with constant suction velocity in the presence of a

transverse uniform strength has recently been investigated by gupta . in

this note, the problem is generalized to take into account the effect of

free convection, when a body force is acting parallel to the wall . the

fluid is assumed to be semi-incompressible as usual . in addition to

the obvious practical significance, this problem is also interesting in

the sense that it provides another exact solution of the

magnetohydrodynamic equations, since the only electromagnetic assumptions involved

are constant properties and freedom from excessive charges .

.I 88

.T

magnetohydrodynamic free-convection pipe flow .

.A

cramer,k.r.

.B

j. ae. scs. 28, 1961, 736.

.W

magnetohydrodynamic free-convection pipe flow .

it has been shown that transverse magnetic fields of practical

strengths exert considerable influence on liquid-metal,

free-convection, vertical, flat-plate and parallel-plate flow fields .

the extent of influence was determined by the magnitude of a

nondimensional parameter a which is the ratio of the hartmann

number to the fourth root of the grashof number, and is a

measure of the relative influence of the magnetic and buoyant

forces . in this note the steady, fully developed, laminar,

free-convection flow of a fluid of electrical conductivity through

a fully submerged, open-ended, constant-temperature, vertical

pipe located in a transverse magnetic field of strength is

analyzed in terms of the same parameter . the magnitude of

its influence on the velocity and temperature profiles, the surface

shear and heat transfer, and the volumetric flow rate is

determined .

.I 89

.T

an investigation of separated flows, part i: the pressure

field .

.A

charwat,a.f.

.B

j. ae. scs. 28, 1961, 457.

.W

an investigation of separated flows, part i: the pressure field .

the present article describes an investigation of several types of separated regions such as blunt-base wakes and cavities formed in cutouts in the boundaries and ahead of or behind two dimensional steps in supersonic (mach numbers 2 to 4) and subsonic flow . the conditions for the existence, the geometry, and the pressure field are described in this paper .

a second article (to be published) will describe investigations of the internal flow and the heat transfer across such separated regions .

it is found that there is a maximum (critical) ratio of the length of the separated free-shear layer to the depth of the depression in the boundary beyond which the cavity collapses, leaving mutually independent separated regions at each protrusion . this critical length changes greatly upon laminar-turbulent transition in the oncoming boundary layer,. in either laminar or turbulent flow it is approximately independent of mach and reynolds numbers . a semiempirical correlation predicting the conditions under which the flow will span a depression of arbitrary depth is proposed .

detailed pressure distributions along the boundaries of a cavity (in turbulent flow) are presented as a function of the ratio of the cavity length to the critical length, which is found to be the pertinent similarity parameter . for short notches the impact pressure due to the reversal of the inner portion of the shear layer at recompression tends to thicken the shear layer and a type of boundary layer-free stream interaction governs the pressure field . the pressure in the cavity is nearly constant and can be higher than free-stream . in long notches the shear layer bends inward at separation and curves back gradually ahead of the recompression point . the floor-pressure variation is pronounced and the recovery pressure at reattachment is small . the variation of the drag coefficient with mach number reflects the change from one to the other mechanism of recompression .

detailed surveys of the mach-number distributions in a blunt-body wake and the mixing region behind its throat, as well as in the shear layer spanning a cutout in a wall, are presented and analyzed . it is found that, in general, the assumptions of the simple supersonic-wake models which rely on a principle of steady flow with mass conservation in the cavity are not adequate for cavities in which there is recompression against a boundary .

results showing the influence of the thickness of the initial boundary layer (in the range of 0.3 to 3 times the notch depth) and of the geometry of the notch are also presented .

.I 90

.T

periodic temperature distributions in a two-layer composite slab .

.A

stonecypher,t.e.

.B

j. ae. scs. 27, 1960, 152.

.W

periodic temperature distributions in a two-layer composite

slab .

an investigation to determine the feasibility of using an

insulating thermal barrier to protect exposed solid

propellant motors from atmospheric or environmental temperature

variations has recently been completed . in one portion of this

study, a solution was developed for the periodic temperature

distribution in a two-layer composite slab . one exposed surface of

this composite slab was adiabatic, and the other exposed surface

was subjected to a sinusoidal temperature variation . the

technique used in the analysis was similar to that of grober . in this

note, pertinent features of the development of the solution are

given .

.I 91

.T

periodic temperature distribution in a two-layer composite slab .

.A

w. f. campbell

.B

national aeronautical establishment, ottawa, ont., canada

.W

periodic temperature distribution in a two-layer composite slab .

in a recent contribution to the reader's forum, under the above title,

stonecypher outlined a method for finding the periodic temperature

distribution in a two-layer composite slab, one exposed surface of the slab

being insulated and the other subject to a sinusoidal temperature variation . perfect thermal contact between the two layers, and constant

thermal properties were assumed .

two years ago i drew attention in these pages to a method for

determining the transient temperature in such a two-layer slab resulting from a

triangular heat-input pulse . i should like to point out that this same

method also is applicable to the case where one external face is given

a sinusoidal temperature variation with time . the method is based on

the analogy between one-dimensional heat flow and the flow of an

electric current in a simple transmission line having only series resistance

and parallel capacitance .

.I 92

.T

the analysis of redundant structures by the use of

high-speed digital computers .

.A

chrichlow,w.j. and haggenmacher,g.w.

.B

j. ae. scs. 27, 1960, 595.

.W

the analysis of redundant structures by the use of

high-speed digital computers .

large-scale redundant structure analyses are currently

feasible by the use of modern high-speed digital computers .

this capability opportunely meets the urgent need to solve

complex problems which otherwise would be hopelessly beyond

the capacity of the hand desk computer . however, the difficulties

have now shifted from tedious hand computations to the problems of adequately representing the structure by a model and

of the peculiarities of irregular geometrical configurations .

a wide scope of problem types can be handled by a generalized

program approach . matrix formulation is used for the organization

of input data and for handling data transfer in the large

complex of subroutines, including the formation of equilibrium

and continuity conditions to the final loads and deflections .

simultaneous treatment of thermal expansions and plasticity

is included .

the use of minimum-size redundant systems is emphasized,

starting from the philosophy of cutting members to provide a

statically determinate structure . improved numerical accuracy

and problem size capacity is gained for a given computer .

examples are discussed ranging from simple plane-load diffusion

problems to pressurized fuselage cutouts and complex

wing-fuselage-shell intersection-type problems .

.I 93

.T

the supersonic blunt body problem - review and extensions .

.A

van dyke,m.d.

.B

j. ae. scs. 25, 1958, 485.

.W

the supersonic blunt body problem - review and extensions .

a survey of existing analytical treatments of the supersonic or

hypersonic blunt-body problem indicates that none is adequate for predicting the details of the flow field . reasons are given for

the failure of various plausible approximations . a numerical

method, which is simpler than others proposed, is set forth for

solving the full inviscid equations using a medium-sized electronic

computer . results are shown from a number of solutions for

bodies that support detached shock waves described by conic

sections .

.I 94

.T

the transverse curvature effect in compressible axially

symmetric laminar boundary layer flow .

.A

probstein,r.f. and elliott,d.

.B

j. ae.scs. 28, 1956, 206.

.W

the transverse curvature effect in compressible axially

symmetric laminar boundary layer flow .

  the viscous transverse curvature effect in compressible axially

symmetric laminar boundary-layer flow has been investigated,

and it is found that the effect is characterized by the parameter

which is essentially the ratio of the boundary-layer thickness

to body radius . it is shown that the busemann and crocco

integrals of the two-dimensional energy equation for are

still valid for axially symmetric flow in which the transverse

curvature effects are considered . by a generalization of

mangler's transformation it is then shown that the boundary-layer equations are reducible to an almost two-dimensional form, making the analysis simpler for two asymptotic flow regions characterized by and less than or of the order of unity . it is with the latter region that the present paper is primarily concerned, and for this case it is shown that the additional term in the momentum and energy equations, which differentiates them from the two-dimensional form, behaves like an external favorable pressure gradient .

except for certain special cases it is necessary to obtain the of the order of unity by means of asymptotic expansions in ascending powers of a parameter that is small compared to unity but proportional to . it is shown how the asymptotic solutions can be found for (1) the velocity and temperature distributions for the compressible zero pressure gradient case when the body shapes are given by and and (2) the velocity distribution for incompressible flow with an external velocity of the form past a body given by . the zeroth approximation is the mangler result . for the cases of a linear external velocity distribution, similar profiles can be found for all values of . more generally it is shown that similar profiles exist if the exponents n and m satisfy the condition that . here, similar is used in the restricted meaning that the distributions are derivable from ordinary differential equations .

in the case of the cone and cylinder with zero pressure gradient where the equations have been numerically integrated for, the first-order correction to the mangler formulation shows that the effect on both the skin-friction coefficient and heat-transfer

rate can become appreciable in the range where is less than

or of the order of unity . at a constant, the effects are

increased in magnitude when either the ratio of wall to free-stream

temperature, or mach number, is increased . also, all other

conditions being equal, for the same value of the skin-friction

coefficient and heat-transfer increase on the cylinder is greater

than that on the cone .

for flows with pressure gradient, the transverse curvature term

behaves again like a favorable pressure gradient and tends to

delay both separation and transition when compared with axially

symmetric flows in which the transverse curvature effect is

neglected .

.I 95

.T

temperature distribution and thermal stresses in a

model of a supersonic wing .

.A

pohle,f.v. and oliver,h.

.B

j. ae. scs. 21, 1954, 8.

.W

temperature distribution and thermal stresses in a

model of a supersonic wing .

the transient temperature distribution and the thermal

stresses in an idealized wing structure considered by hoff and

torda in reference 1 are determined . only the effects of aerodynamic heating and of heat conduction are included,. radiation

and convection effects are neglected . the present work differs

from that of reference 1 in that the conduction from the cap to

the web is considered when the temperature of the cap is

calculated, and the spar cap temperature is assumed to be a function

of both space and time . graphs of temperature and thermal

stress distributions are presented, and the results are compared

with those of reference 1 .

.I 96

.T

review of published data on the effect of roughness on transition from

laminar to turbulent flow .

.A

hugh l. dryden

.B

national advisory committee for aeronautics

.W

review of published data on the effect of roughness on transition from

laminar to turbulent flow .

a review is presented of the published data on the effect of roughness,

especially single roughness elements, on transition from laminar to

turbulent flow, in which an attempt is made to reanalyze and correlate the

available information . the reanalysis shows that the transition

reynolds number of a flat plate with zero pressure gradient is a

function of the ratio of the height of the roughness element to the

displacement thickness of the boundary layer at the element, this functional

relation being a better representation of the data than a constant

critical reynolds number of the roughness element . other data show that the
effects of roghness are similar in streams of different initial
turbulence and that a plot of the ratio of transition reynolds number of the
rough plate to that for the smooth plate against the ratio of the height
of the roughness element to displacement thickness of the boundary layer
at the element gives good correlation of all the data for a given shape
when transition occurs downstream from the
roughness element . at a certain value of the height-thickness ratio
dependent on the stream speed, location of roughness element, and
airstream turbulence, the transition position reaches the element and
remains there as the height or the stream speed is further increased .
the paper also discusses available data on the effect of distributed
roughness on transition on a flat plate, as well as some of the
published data on roughness effects on transition on air-foils .

.I 97

.T

a mixing theory for the interaction between dissipative
flows and nearly isentropic streams .

.A

crocco,l. and lees,l.

.B

j. ae. scs. 19, 1952, 649.

.W

a mixing theory for the interaction between dissipative
flows and nearly isentropic streams .

by means of a simplified theoretical /model,/ the present paper
treats the general class of flow problems characterized by the

interaction between a viscous or dissipative flow near the surface

of a solid body, or in its wake, and an /outer/ nearly isentropic

stream . for the present, the external flow is taken to be a plane,

steady, supersonic flow, which makes a small angle with a plane

surface or plane of symmetry, although the methods used can be

extended to curved surfaces, to axially symmetric supersonic

flows, and also to subsonic flows . the internal dissipative flow

is regarded as quasi-one-dimensional and parallel to the surface

on the average, with a properly defined mean velocity and mean

temperature . the nonuniformity of the actual velocity

distribution is taken into account only approximately by means of a

relation between mean temperature and mean velocity .

mixing, or the transport of momentum from outer stream to

dissipative flow, is considered to be the fundamental physical process

determining the pressure rise that can be supported by the flow .

with the aid of this concept, a large number of flow problems is

shown to be basically similar, such as boundary-layer

shockwave interaction, wake flow behind blunt-based bodies (base

pressure problem), flow separation in overexpanded supersonic

nozzles, separation on wings and bodies, etc .

.I 98

.T

heat transfer by laminar flow to a rotating plate .

.A

millsaps,k. and pohlhausen,k.

.B

j. ae. scs. 19, 1952, 120.

.W

heat transfer by laminar flow to a rotating plate .

an exact solution of the heat-transfer problem for the von

karman example of the laminar flow of a viscous fluid over a

rotating plate is given in dimensionless form and physically

discussed . the solution is explicitly given for a constant

temperature on the plate with viscous dissipation included . the

numerical results are given for prandtl numbers from 0.5 to 10 .

.I 99

.T

the fundamentals of the statistical theory of turbulence .

.A

th. von karman

.B

california institute of technology

.W

the fundamentals of the statistical theory of turbulence .

statistical theory in general considers mean values of certain

quantities . in the case of the turbulent motion one is interested in mean

values of velocities and of their derivatives, and in mean values of

squares and products of velocities and their derivatives . it was o.

reynolds who first expressed the so-called apparent or turbulent

stresses by the mean values of the products of the velocity components . the

different theories suggested so far have as their common objective the

establishment of relations between certain mean values, e.g. between the

turbulent shear stresses given by the mean products of velocity

fluctuations and the derivatives of the mean velocities, i.e. the measured mean velocity gradients . in this sort of investigations the conception of

the /correlation/ is of paramount importance . the late a. friedman

tried to introduce the correlations as unknown variables in the

hydrodynamic equations., however, he could not carry his investigations to

practical results, i.e., to results which can be compared with the

experimental evidence . recently, g. i. taylor had success in his

analysis of /isotropic/ turbulence by means of correlation calculations, and

was able to discuss, theoretically, the problem of the decay of

turbulence in a windstream behind a turbulence producing device . his theory

raised considerable interest because it is concerned with the important

problem of wind-tunnel turbulence and its results could be compared

directly with experimental work done by dryden in this country and by

fage, townend and simmons in england .

the present paper is concerned with two fundamental problems.. with

uniform isotropic turbulence and with the turbulent friction in a

parallel stream . first, the general theory of isotropic turbulence is

developed . this general theory includes taylor's consideration as a

special case . however, it

.I 100

.T

vibration isolation of aircraft power plants .

.A

taylor,e.s. and browne,k.a.

.B

j. ae. scs. 6, 1938, 43.

.W

vibration isolation of aircraft power plants .

vibration in aircraft structure can almost always be traced to vibratory forces originating from the power plant . these forces are transmitted to the aircraft in two ways .. (1) by the action of air forces upon the surfaces of the aircraft in, or adjacent to, the slip stream of the propeller, and (2) by direct transmission of unbalanced forces from the power plant through the engine mounting . the latter has always caused the preponderance of disturbance . vibratory stresses induced in the engine mounting structure occasionally produce fatigue failures in the associated parts, and always shorten the useful life of the entire aircraft structure . more important, however, are the psychological and physiological effects of continuous vibration and its attendant noise on the passengers and crew . this may very likely be the major source of the rapid fatigue which is so intimately associated with flying . the importance and desirability of drastically reducing vibration can hardly be questioned .

this paper is limited to a consideration of the directly transmitted forces and, further, considers the power plants as rigid bodies attached by flexible means to the aircraft which is also considered as a rigid body of relatively large mass . it is also limited to the case of engines and engine supporting structures having axial symmetry (radial engines), although the methods employed could easily be extended to other cases .

.I 101

.T

laminar heat transfer over blunt-nosed bodies at hypersonic flight

speeds .

.A

lester lees

.B

the ramo-wooldridge corporation, los angles, and california institute

of technology, pasadena, california

.W

laminar heat transfer over blunt-nosed bodies at hypersonic flight

speeds .

this paper deals with two limiting cases of laminar heat transfer over

blunt-nosed bodies at hypersonic flight speeds, or high stagnation

temperatures.. (a) thermodynamic equilibrium, in which the chemical

reaction rates are regarded as /very fast/ compared to the rates of diffusion

across streamlines., (b) diffusion as rate-governing, in which the

volume recombination rates within the boundary layer are /very slow/

compared to diffusion across streamlines . in either case the gas density

near the surface of a blunt-nosed body is much higher than the density

just outside the boundary layer, and the velocity and stagnation

enthalpy profiles are much less sensitive to pressure gradient than in the

more familiar case of moderate temperature differences . in fact, in

case (a), the nondimensionalized enthalpy gradient at the surface is

represented very accurately by the /classical/ zero pressure gradient

value, and the surface heat-transfer rate distribution is obtained directly in terms of the surface pressure distribution . in order to illustrate the method, this solution is applied to the special cases of an unyawed hemisphere and an unyawed, blunt cone capped by a spherical segment .

in the opposite limiting case where diffusion is rate-controlling the diffusion equation for each species is reduced to the same form as the low-speed energy equation, except that the prandtl number is replaced by the schmidt number . the simplifications introduced in case (a) are also applicable here, and the expression for surface heat transfer rate is similar., the maximum value of the ratio between the rate of heat transfer by diffusion alone and by heat conduction alone in the case of thermodynamic equilibrium is given by.. (prandtl no./schmidt no .) when the diffusion coefficient is estimated by taking a reasonable value of atom-molecule collision cross section this ratio is 1.30 .

additional theoretical and (especially) experimental studies are clearly required before these simple results are accepted .

.I 102

.T

advantages and limitations of models .

.A

sobey,a.j.

.B

j. r. ae. s. 63, 1959, 646.

.W

advantages and limitations of models .

summary .. the use of models for structural test investigations in the presence of kinetic heating effects is examined . the principal features of the complex process to be

represented are discussed under the classifications

external air flow, internal heat transfer,

elastic response . of these the second is found

to influence most model design, and an

analysis of a typical structure is included to

illustrate the various contributions to

internal heat transfer .

.I 103

.T

theory of mixing and chemical reaction in the opposed

jet diffusion flame .

.A

spalding,d.b.

.B

a.r.s. jnl. 31, 1961, 763.

.W

theory of mixing and chemical reaction in the opposed

jet diffusion flame .

an idealization of the flow system used by potter

and butler is analyzed . the differential

equation of mixing is solved exactly, to give the location

of, and burning rate in, the flame . the solutions

to the chemical kinetic differential equation

are discussed, relations being derived between the jet

flow rate at extinction, the chemical kinetic

constants and the laminar flame speed in premixed gases . it is shown that the jet flow rate at

extinction is independent of the transport properties .

comparison is made with the experimental

data of potter, heimel and butler . it is argued that

experiments must be carried out at higher

reynolds numbers if the measurements are to be

quantitatively analyzable .

.I 104

.T

similar solutions of a free convection boundary layer

equation for an electrically conducting fluid .

.A

reeves,b.l.

.B

a.r.s. jnl. 31, 1961, 517.

.W

similar solutions of a free convection boundary layer

equation for an electrically conducting fluid .

author investigates the existence of a class of similar solutions

for free convection from a vertical flat plate, such as are known for

free convection in a nonconducting fluid . the magnetic field acts

transversely to the fluid motion and is assumed to remain constant

in the direction perpendicular to the plate . this introduces into

the momentum equation a retarding force which is a function only

of x, the distance along the plate length . for similarity it is

found that the magnetic inductance must vary as . if

the plate temperature is constant . if n = 0, the magnetic inductance is constant while the plate temperature increases linearly

with x .

.I 105

.T

the asymptotic boundary layer on a circular cylinder

in axial incompressible flow .

.A

stewartson,k.

.B

q.app.math. 13, 1955, 113.

.W

the asymptotic boundary layer on a circular cylinder

in axial incompressible flow .

in this paper the incompressible boundary layer over a

circular cylinder in an axial flow is investigated far from the

leading edge . if u and v are the velocity components in the

x and r direction respectively and a stream function is

introduced by and, then

for a constant free-stream velocity has the

following asymptotic form ..

where the p's are determined successively, first for s=1 and

all t, then s=2 and all t, etc., from ordinary differential

equations . here and log c=euler's

constant . it is shown that the effect of the curvature of the

body (in planes perpendicular to the flow) is to increase

the skin friction . also the case in which the free-stream

velocity is proportional to (at the

method breaks down), is studied . it is concluded that the

effect of the curvature of the cylinder, when the boundary

layer has a thickness comparable with its radius of

curvature, is to delay separation .

.I 106

.T

the transverse potential flow past a body of revolution .

.A

campbell,i.j.

.B

q.j.mech.app.math. 9, 1956, 140.

.W

the transverse potential flow past a body of revolution .

it is shown that in the potential flow of

an incompressible inviscid fluid past a

body of revolution set with its axis at right

angles to the stream, the velocity

components at the surface along and perpendicular

to the meridians vary with azimuthal

angle round the body in a simple manner .

this is shown by entirely elementary

considerations .

.I 107

.T

on the mixing of two parallel streams .

.A

ting,lu.

.B

j.math.phys. 38, 1959, 153.

.W

on the mixing of two parallel streams .

using the techniques of boundary-layer theory, the proper third boundary condition for the mixing of two parallel streams is derived from the compatibility condition of the higher order approximation . it is shown that the commonly adopted third boundary condition of balancing of transverse momentum is correct only for the mixing problem of two semi-infinite incompressible streams . for the fulfillment of the proper third boundary condition, the possibility of introducing the similar solution of blasius type is examined for various cases .

.I 108

.T

properties of the confluent hypergeometric function .

.A

a. d. macdonald

.B

.W

properties of the confluent hypergeometric function .

the confluent hypergeometric functions have proved useful in many branches of physics . they have been used in such problems involving diffusion and sedimentation, as isotope separation and protein molecular weight determinations in the ultracentrifuge . the solution of the equation for the velocity distribution of electrons in high frequency gas discharges may frequently be expressed in terms of these functions .

the high frequency breakdown electric field may then be predicted

theoretically for gases by the use of such solutions together with kinetic

theory .

this report presents some of the properties of the confluent

hypergeometric functions together with six-figure tables of the functions .

.I 109

.T

the production of uniform shear flow in a wind tunnel .

.A

owen,p.r. and zienkiewicz,h.k.

.B

j.fluid mech. 2, 1957, 521.

.W

the production of uniform shear flow in a wind tunnel .

a nearly uniform shear flow was obtained in the working

section of a wind tunnel by inserting a grid of parallel rods with

varying spacing .

the function of such a grid is to impose a resistance to the

flow, so graded across the working section as to produce a linear

variation in the total pressure at large distances downstream

without introducing an appreciable gradient in static pressure near

the grid . a method of calculating a suitable arrangement of the

rods is described . although this method is strictly applicable

only to weakly sheared flows, an experiment made with a grid

designed for a shear parameter as large as 0.45 gave results in

close agreement with the theory . there was no evidence from

the experiment of any large-scale secondary flow accompanying the shear--a danger inherent in an empirical attempt to grade the

resistance of the grid--nor was any tendency observed for the

shear to decay with increasing distance from the grid .

.I 110

.T

dynamics of a dissociating gas .

.A

lighthill,m.j.

.B

j.fluid mech. 2, 1957, 1.

.W

dynamics of a dissociating gas .

this is a lucid introduction to the effects of dissociation

in gas dynamics . the problem in view is that of air flow

past a bluff body at speeds somewhat above 2 km sec .

thermodynamic equilibrium is assumed,. theories of near

equilibrium for transport properties and of large

departures from equilibrium being promised in parts 2 and 3 .

following a survey of the equilibrium statistical

thermodynamics of a pure dissociating diatomic gas, a

new model is introduced . this /ideal dissociating gas/ is

characterized by only three constants, the characteristic

temperature, density and internal energy for dissociation .

physically, it may be regarded as having its vibrational

modes always just half excited (so that at low

temperatures the ratio of specific heats approaches 4 3 rather

then 7 5) . thermodynamic properties of the ideal gas are derived, and the oblique shock wave relations deduced in the /strong-shock/ approximation (including an elegant relation between the principal curvatures of any bow shock and the subsequent vorticity) . useful relations are given for the isentropic changes that take place along streamlines between shocks .

various of these results are applied to the problem typified by a sphere flying at high mach number . the newtonian impact theory and its empirical modification are dismissed as lacking theoretical basis, in favor of the limit for large values of both mach number and density ratio across the shock . it is suggested that the zero surface pressure sometimes predicted by the latter theory corresponds to separation not of the flow but of the shock wave from the surface . an estimate is given for the subsequent shape of the shock . finally, another approximation is applied to the region near the stagnation streamline . the fluid is assumed incompressible, but rotational in accord with the shock relations,. and it is shown that a spherical shock corresponds to a concentric spherical body . the resulting surface pressure is within 1 per cent of that predicted by freeman's second approximation based on the newtonian-plus-centrifugal solution (same j. 1 (1956),

.I 111

.T

the laminar boundary layer equation: a method of solution by means of an automatic computer .

.A

leigh,d.c.f.

.B

proc. cam. phil. s. 51, 1955, 320.

.W

the laminar boundary layer equation: a method of solution

by means of an automatic computer .

a method, very suitable for use with an automatic computer,

of solving the hartree-womersley approximation to the

incompressible boundary-layer equation is developed .

it is based on an iterative process and the choleski method

of solving a simultaneous set of linear algebraic equations .

the programming of this method for an automatic computer is

discussed . tables of a solution of the boundary-layer

equation in a region upstream of the separation point are

given . in the upstream neighbourhood of separation

this solution is compared with goldstein's

asymptotic solution and

the agreement is good .

.I 112

.T

steady motion of conducting fluids in pipes under transverse

magnetic fields .

.A

shercliff,j.a.

.B

proc. cam. phil.s. 49, 1953, 136.

.W

steady motion of conducting fluids in pipes under transverse

magnetic fields .

this paper studies the steady

motion of an electrically conducting, viscous fluid

along channels in the presence of an imposed

transverse magnetic field when the walls do not

conduct currents . the equations which determine

the velocity profile, induced currents and

field are derived and solved exactly in the case

of a rectangular channel . when the imposed

field is sufficiently strong the velocity profile is

found to degenerate into a core of uniform flow

surrounded by boundary layers on each wall .

the layers on the walls parallel to the imposed

field are of a novel character . an analogous

degenerate solution for channels of any symmetrical

shape is developed . the predicted pressure

gradients for given volumes of flow at various field

strengths are finally compared with experimental results for square and

circular pipes .

.I 113

.T

acoustical signal detection in turbulent airflow .

.A

smith,m.w. and lambert,r.f.

.B

j.acous.s.am. 32, 1960, 858.

.W

acoustical signal detection in turbulent airflow .

improvement in detected signal-to-noise ratio is obtained

for a periodic signal masked by additive noise

and turbulent noise backgrounds . comparisons are made

between autocorrelation, crosscorrelation, and a

combination of frequency filtering and crosscorrelation .

although the latter method provided the greatest

improvement, the crosscorrelation technique was the

most successful single method . it turned out that

the maximum improvement obtainable was limited by

the dynamic range of the correlator computer and

not by errors due to finite averaging time and scanning

the delay . the improvement for signals masked by

turbulent noise was found to be about 5 db less than that

obtained for additive noise .

.I 114

.T

response of plates to a decaying and convecting randon

pressure field .

.A

dyer,i.

.B

j.acous.s.am. 31, 1959, 922.

.W

response of plates to a decaying and convecting randon pressure field .

following the methods of lyon, an analysis of the vibratory response of a plate to a random pressure field is given . the pressure correlation of the random field is assumed to have a scale small compared to the plate size, to decay exponentially, and to convect with constant speed over the plate . two cases are considered, one in which the convection speed is much less than the speed of free flexural waves in the plate, the other in which the convection speed is the same order as the flexural wave speed . the mean square plate displacement is shown to be relatively independent of convection for speeds much less than the flexural wave speed, and to increase significantly for speeds in the order of the flexural wave speed . it is shown that damping is usually, but not always, an effective means of vibration reduction . in the case of convection speeds much smaller than the flexural speed, the use of hysteretic damping for reduction of the displacement response is shown to be limited by the decay of the assumed random pressure field .

.I 115

.T

on turbulent lubrication .

.A

constantinescu,v.n.

.B

proc.inst.mech.e. 173, 1959, 881.

.W

on turbulent lubrication .

the paper concerns the hydrodynamic turbulent motion in the lubricant layer . proceeding from the reynolds equations and introducing the approximations currently used in lubrication problems, owing to the lubricant film thickness, the general motion equations for turbulent lubrication are written .

using the prandtl mixing length hypothesis, exact and approximate solutions are obtained for the velocity distribution into the lubricant layer . the results are discussed by pointing out the pressure gradient and the reynolds number influence on the velocity distributions, as well as the differences with respect to the laminar flow .

in order to obtain simple formulae, the exact dependence of the rate of flow on the pressure gradient into a dimensionless form is replaced by a linear relation, the slope of which depends on the reynolds number . this approximation allows the obtainment of the pressure differential equation under a simple form . the pressure equation is integrated in case of journal bearings, by assuming a constant or a variable viscosity of the lubricant .

the results are compared to the experimental

data obtained by m. i. smith and d. d.

fuller and the good qualitative agreement is pointed out .

.I 116

.T

the elliptic cylinder in a shear flow with hyperbolic

velocity profile .

.A

jones,e.e.

.B

q.j.mech.app.math. 12, 1959, 191.

.W

the elliptic cylinder in a shear flow with hyperbolic

velocity profile .

the stream function for the shear flow with hyperbolic

velocity profile past an elliptic cylinder has been determined

as an infinite series of mathieu functions . it is found that

the stagnation streamline of the flow is displaced towards

a region of higher velocity, this displacement increasing

the main stream, (2) as the stream becomes progressively

non-uniform, (3) with increase of minor axis length when

the major axis length remains invariant . in each case the

displacement reaches a limiting value as the cylinder moves away

from the axis of symmetry of the stream . these limiting

values are reached at critical distances from the axis of symmetry,

which decrease as the stream becomes progressively non-uniform, but these distances are approximately independent of incidence .

the pressure coefficients and the resultant force and moment

coefficients associated with the cylinder have also been obtained,

and investigated numerically for the flat plate type of cylinder .

.I 117

.T

the motion of a viscous liquid past a paraboloid .

.A

mather,d.j.

.B

q.j.mech.app.math. 14, 1961, 423.

.W

the motion of a viscous liquid past a paraboloid .

an approximate solution for the steady

flow of incompressible viscous liquid past

a paraboloid of revolution is described .

an assumption is made for the form of the

stokes stream function and substituted

into the navier-stokes equations using

paraboloidal coordinates . after making

suitable approximations, a non-linear

differential equation for a function f is

deduced . the solutions of this equation

depend on the reynolds number of the

flow considered . examples found by

numerical integration are given to illustrate

the properties of the function f for

reynolds numbers varying from 0.0001 to

is found, and it is shown that this approximate

solution tends to the perfect fluid

flow away from the boundary, allowance

being made for the displacement effect of

what may be called the boundary layer .

.I 118

.T

the transonic flow of a compressible fluid through

an axially symmetrical nozzle .

.A

tomotika,s. and hasimoto,z.

.B

j.math.phys. 29, 1950, 105.

.W

the transonic flow of a compressible fluid through

an axially symmetrical nozzle .

by a method similar to that developed by s. tomotika

and k. tamada (quart. appl. math. 7, 381-397 (1950),.

these rev. 11, 275) for computing two-dimensional mixed

isentropic flows in the sonic region, the flow in the vicinity

of the throat of an axially symmetrical nozzle is studied .

several exact solutions to von karman's equation for axially

symmetrical transonic flows are obtained and the one that

gives flows through a converging and diverging nozzle is

considered in detail . this solution consists of four branches

of which two are rejected because of singularities . of the remaining two branches, one gives pure supersonic flow and

the other gives taylor's type of flow with a local supersonic

region in the throat . by varying a parameter, the latter

branch approaches two asymptotes which yield meyer's

type of asymmetrical flows .

.I 119

.T

conduction of fluctuating heat flow in a wall consisting

of many layers .

.A

vodicka,v.

.B

app.sc.res. 5, 1955, 108.

.W

conduction of fluctuating heat flow in a wall consisting

of many layers .

van gorcum has pointed to interesting and important analogies

between the theory of a passive four-pole and the conduction of heat

waves through stratiform bodies . this paper generalizes in certain

regards van gorcum's ideas and draws their consequences for the case of

a solid, bounded by two infinite parallel planes and consisting of any

number of layers made from different materials .

.I 120

.T

measurement of convective heat transfer by means of

the reynolds analogy .

.A

granville,r.a. and boxall,g.

.B

brit.j.app.phys. 11, 1960, 471.

.W

measurement of convective heat transfer by means of

the reynolds analogy .

preston's method for measuring skin friction in pipes has

been extended to include non-uniform flow, with and

without pressure gradients, over flat surfaces . by means of a

modified form of the reynolds analogy, the local

convective heat transfer coefficient can be related to the skin

friction, and it is proposed that the method be used in

aerodynamic models of furnaces and in heat transfer plant

of simple geometry . more investigations are required of

the effects of fluid turbulence, surface roughness and

surface curvature on convective heat transfer and skin

friction .

.I 121

.T

a theory for base pressures in transonic and supersonic

flow .

.A

korst,h.h.

.B

j.app.mech. 23, 1956, 593.

.W

a theory for base pressures in transonic and supersonic

flow .

a physical flow model is devised based on the concepts of interaction between the dissipative shear flow and the adjacent free stream and the conservation of mass in the wake . four flow components are integrated in the model,. namely, the flow approaching the trailing edge, the expansion around the trailing edge, the mixing within the free-jet boundary, and the recompression at the end of the wake . a unique and stable solution results for the base pressure . theoretical results obtained for thin approaching boundary layer do not require empirical information and are, therefore, best suited to evaluate the merits of the theory . here emphasized is the case of isoenergetic constant-pressure mixing in the turbulent free-jet boundary and agreement is found between theory and experimental data .

.I 122

.T

a simplified approximate method for the calculation of the pressure around conical bodies of arbitrary shape in supersonic and hypersonic flow .

.A

willi f. jacobs

.B

lockheed aircraft corporation, georgia division

.W

a simplified approximate method for the calculation of the pressure around conical bodies of arbitrary shape in supersonic and hypersonic flow .

exact conical-flow solutions are available only for circular cones at zero angle of attack . for nonaxisymmetric cones or cones at angle of attack, only approximate methods exist . these methods are generally quite complicated and further limited to certain body shapes or certain mach-number ranges . a great need was therefore felt for a simple approximate method applicable to any arbitrarily shaped conical body at zero incidence as well as at angle of attack .

such a method has been developed recently at lockheed and is presented here in abbreviated form . the method is based on the /equivalent-cone/ theory . this theory determines the pressure on a conical body utilizing information for a symmetric cone at zero angle of attack with the same normal component of the free stream with respect to the surface as the local element of the body considered . this method works relatively well at high mach numbers . however, it is quite inconsistent at lower mach numbers, especially for bodies which deviate considerably from circular cones . the equivalent-cone method does not give satisfactory results, mainly due to the fact that it considers only the local surface element on the body independent of the other body elements in the newtonian-theory manner .

.I 123

.T

the downstream influence of mass transfer at the nose

of a slender cone .

.A

cresci,r.j. and libby,p.a.

.B

j.aer.scs. 29, 1962, 815.

.W

the downstream influence of mass transfer at the nose of a slender cone .

the influence of localized mass transfer at the nose of a slender cone under hypersonic flow conditions has been studied by experimental and theoretical means . two gaseous coolants, nitrogen and helium, are injected through a porous plug subtending a half angle of 30 . the effect of the mass transfer on the shock shape, pressure distribution, heat transfer, and transition are investigated . the experimental work involved tests in the mach-number-8.0 tunnel at pibal . the theoretical analysis involved a study of the effect of mass transfer on the shock stand-off distance and leads to an inviscid-flow parameter permitting the experimentally determined shock shape and pressure distribution to be extrapolated to other than test conditions and to other coolant gases . there is obtained the maximum value of this parameter resulting in no significant alteration of the pressure distribution on the cone and thus defining the flows in which boundary-layer-type similarity applies .

significant reductions in heat transfer are obtained with injection . indeed, with small amounts of helium injection the peak heating is found to occur downstream on the cone and to be an order of magnitude less than would occur at the stagnation point without mass transfer . with nitrogen early transition is found to occur, so that local heating rates are actually increased over those prevailing at the same reynolds number without

injection .

.I 124

.T

a summary of the supersonic pressure drag of bodies

of revolution .

.A

morris,d.n.

.B

j.aero.scs. 28, 1961, 563.

.W

a summary of the supersonic pressure drag of bodies

of revolution .

a number of approximate theories for supersonic and

hypersonic flow over bodies of revolution at zero angle of attack are

appraised by a critical comparison with characteristics and

second-order results, with the use of hypersonic similarity as a

basis for the comparison . most of the approximate theories

are inadequate except over very limited ranges of fineness ratio

and mach number . the combination of second-order

supersonic theory and second-order shock-expansion theory provides

consistently good results throughout the supersonic speed range .

on the basis of exact (or nearly exact) supersonic solutions and

a limited amount of test data and theory in the transonic region,

summary design curves are developed that give the pressure

drag of conical and ogive noses and conical and ogive boattails

over the complete range of transonic, supersonic, and hypersonic mach numbers . other shapes can be analyzed in the same

manner, provided that an equivalent amount of data is available .

the analysis is made with the assumption of inviscid flow,

so that the effects of boundary-layer growth, shock

boundary-layer interaction, and flow separation are not included . the

present correlations provide a sound basis of inviscid-flow results

from which these additional viscous effects can be evaluated .

.I 125

.T

measurements of skin friction of the compressible turbulent

boundary layer on a cone with foreign gas injection .

.A

pappas,c.c. and okuno,a.f.

.B

j.aero.scs. 27, 1960, 321.

.W

measurements of skin friction of the compressible turbulent

boundary layer on a cone with foreign gas injection .

measurements of average skin friction of the turbulent

boundary layer have been made on a 15 total included angle cone with

foreign gas injection . measurements of total skin-friction drag

were obtained at free-stream mach numbers of 0.3, 0.7, 3.5, and

x 10 with injection of helium, air, and freon-12

through the porous wall . substantial reductions in skin

friction are realized with gas injection within the range of mach

numbers of this test . the relative reduction in skin friction is

in accordance with theory--that is, the light gases are most effective when compared on a mass flow basis . there is a marked

effect of mach number on the reduction of average skin friction,.

this effect is not shown by the available theories . limited

transition location measurements indicate that the boundary layer

does not fully trip with gas injection but that the transition point

approaches a forward limit with increasing injection . the

variation of the skin-friction coefficient, for the lower injection rates

with natural transition, is dependent on the flow reynolds

number and type of injected gas,. and at the high injection rates the

skin friction is in fair agreement with the turbulent

boundary-layer results .

.I 126

.T

an investigation of two-dimensional supersonic base

pressures .

.A

charwat,a.f. and yakura,j.k.

.B

j.aero.scs. 25, 1958, 122.

.W

an investigation of two-dimensional supersonic base

pressures .

an investigation of the base pressure behind wedges at mach

numbers 2 and 3 in the laminar and the transitional regime is

reported . temperature and velocity traverses through the

mixing zone are shown and exploratory investigations of the

wake vortex by use of hot wires and flow-visualization techniques are described . it is found that the laminar two-dimensional base

pressure agrees well with chapman's theoretical predictions .

the shear layer exhibits gross velocity distributions

characteristic of the free jet mixing zone, but also shows disturbances that

originate in the expansion-turning of the oncoming boundary

layer . an interesting trailing vortex is observed, which is

explained in terms of nonuniform mixing rate in the wake .

.I 127

.T

supersonic axially symmetric nozzles .

.A

clippinger,r.f.

.B

b. r. l. r794, 1951.

.W

supersonic axially symmetric nozzles .

at each of twenty-one exit mach numbers, ranging

from 1.008 to 8.238, ten supersonic axially symmetric

nozzle shapes with plane sonic surfaces have been computed

on the eniac by the method of characteristics . the

boundary of the shortest of each group of ten has a

sharp edge at the sonic plane, while the others have

smooth boundaries . this report describes the computational

procedures and presents a sample of the results for twenty

nozzles .

more extensive and elaborate tables of the results of the

entire computations are available at the ballistic research laboratories . nozzle contours can be obtained

accurately from them by interpolation for exit mach numbers

between 1.479 and 8.238 for a wide range of ratios of nozzle

length to throat diameter .

.I 128

.T

effects of free stream vorticity on the behaviour of

a viscous boundary layer .

.A

li,t-y.

.B

j.aero.scs. 23, 1956, 1128.

.W

effects of free stream vorticity on the behaviour of

a viscous boundary layer .

  theoretical investigation is considered of the two-dimensional

steady flow field at large distance from a finite object set in a

viscous incompressible fluid . study is made of coordinate-type

expansions for pressure and velocity for large r, uniformly in, for

fixed reynolds number, assuming exact boundary conditions at

infinity and regularity of flow with zero net mass flow across a

simple curve enclosing the object .

  mathematical nature of the distinction between parameter and

coordinate-type expansions is discussed with description of inner

and outer expansions and matching techniques .

  a feature of the expansion procedure is the introduction of an

artificial parameter . inner and outer expansions are matched with the aid of known solutions of the navier-stokes equations .

analysis requires simple consideration of the heat and laplace

equations without resort to special methods .

paper is worth studying by those interested in asymptotic

expansion procedures .

.I 129

.T

an investigation of the noise produced by a subsonic air jet .

.A

j. h. gerrard

.B

university of manchester

.W

an investigation of the noise produced by a subsonic air jet .

to investigate the theoretical predictions of lighthill on aerodynamic

sound, measurements have been made of the sound field of a 1 in. air jet

issuing from a long pipe . the measurements have been made over a wide

frequency band (30 to 10,000 cycles/sec.) and in one-third octave bands

in this frequency range . the mean mach number at the pipe orifice was

varied from 0.3 to 1.0 .

the dependence of the apparent position of the noise sources on

frequency and jet speed was investigated . at a given frequency a source is

situated farther from the jet orifice the higher the jet speed . lower

frequency sources appear farther downstream than ones of higher

frequency, consistent with their association with larger eddies . the

directional characteristics of the sound field at different frequencies

and jet speeds are illustrated by means of scale diagrams showing lines of constant sound intensity . these sound fields are analyzed in terms

of the moving quadrupole sources of lighthill's theory and good

agreement obtained . it is shown that the apparent spread of the sources at

low frequencies is due to the doppler effect . at low frequency

relative to the frequency of maximum power output) the radiation is

predominantly that of three mutually orthogonal longitudinal quadrupoles

which, except for the effect of convection upon it, has a sound field

like a monopole source . at higher frequencies the sound fields of

lateral and longitudinal quadrupoles predominate .

.I 130

.T

the behaviour of non-linear systems .

.A

clauser,f.h.

.B

j.aero. scs. 23, 1958, 411.

.W

the behaviour of non-linear systems .

many of the phenomena that occur in the world around us are

governed by nonlinear relationships . in the development of the

mathematical sciences, the difficulties of nonlinear analysis have

hindered the formulation of nonlinear concepts that would

permit us to understand such phenomena . in the present article,

our progress in understanding the behavior of nonlinear systems

is reviewed and an attempt is made to present the resulting

concepts in such a way that they may be applied with some

generality to other problems .

.I 131

.T

two-dimensional jet mixing of a compressible fluid .

.A

pai,s.i.

.B

j.aero.scs. 16, 1949, 463.

.W

two-dimensional jet mixing of a compressible fluid .

the mixing and divergence of a supersonic jet exhausting into a supersonic stream are investigated theoretically .

in the first part of this paper, the flow is assumed to be laminar . when the velocity and temperature in the jet are different slightly from those of the surrounding stream, by the method of small perturbations and under ordinary boundary layer assumptions, the equation of motion of two-dimensional flow will be reduced to a form of the well-known equation of heat conduction, whose solution is known for any given boundary conditions . it has also been shown that the exact solution of the two dimensional jet mixing of viscous compressible fluids can be obtained by successive approximations starting with the solution of small perturbations .

velocity and temperature distributions for two cases--one is the mixing of two-uniform flows and the other is the mixing of a jet of compressible fluid from a two-dimensional nozzle with full expansion exhausting into a supersonic stream--have been calculated . the properties of the jet mixing depend mainly on the momentum of the jet regardless of whether the change of

momentum is due to the change of velocity or the change of

temperature--i.e., the change of density . compressibility has a

considerable effect on the properties of the jet .

in the second part, the cases of turbulent flow are investigated .

by means of reichardt's theory of free turbulence, the turbulent

shearing stress may be expressed as

it has been shown in this paper that

where is a constant that can be determined experimentally .

the value of n lies between 0 and 1 . the exact value of n

depends on the condition of mixing .

when the expression of turbulent shearing stress given above

is used instead of the viscous stress in the equation of motion,

by suitable transformation of variables, it has been shown that

the equation of two-dimensional turbulent jet mixing is identical

to that of the laminar case . hence, the solution of the first part

of this paper can be applied to the turbulent case, provided that

the characteristic constants and n have been properly chosen .

.I 132

.T

viscosity effects in sound waves of finite amplitude:

in survey in mechanics .

.A

lighthill,m.j.

.B

ed. by g.k.batchelor and r.m.davies. c.u.p. 1956.

.W

viscosity effects in sound waves of finite amplitude:

in survey in mechanics .

this article has as its subject /the conflicting influence on sound propagation of convection on the one hand, and of diffusion and relaxation on the other/, whose importance in the determination of the structure of shock waves was first appreciated clearly by sir geoffrey taylor . as an essential introduction to the main topics, author gives an exceptionally clear and valuable account of the physical mechanisms of viscosity, thermal conductivity, and other diffusion effects, including relaxation . the classical theory of shock-wave formation is then discussed, and some extensions are made .

the remainder of the article is based on the demonstration that the nonlinear equation for plane progressive sound waves, in which convection and diffusion are taken into account to a first approximation, can be transformed into burgers's equation, the general solution of which was given by hopf and cole . this approach, in which all flows are continuous (they become discontinuous at shock waves in the limit as viscosity, etc., tend to zero), allows the author to re-derive and extend whitham's theory of the formation and decay of weak plane shock waves, and to derive many new results, such as the velocity distributions during the union of two shock waves and during the formation of a shock wave . the application of the same idea to non-plane shock waves is also discussed, but more briefly,. in these cases, burgers's equation is not quite such a good approximation as before .

the article concludes with sections on sound waves whose reynolds numbers based on the length scale of the flow and the velocity amplitude are comparable with unity, and on the effects of relaxation on the properties of shock waves . the whole is much more than a survey, and represents a very substantial advance in the theory of sound waves . it is the finest possible tribute to sir geoffrey taylor that he should be able to inspire articles such as this and the others in this volume .

.I 133

.T

some effects of surface curvature on laminar boundary layer flow .

.A

murphy,j.s.

.B

j.aero.scs. 20, 1953, 338.

.W

some effects of surface curvature on laminar boundary layer flow .

the laminar flow of a viscous incompressible fluid over a two-dimensional curved surface is investigated for two cases, one in which the curvature is /large/ and the other in which it is cases are obtained as approximations from the exact equations of motion by an order-of-magnitude analysis . these equations are solved for flow over a particular surface with zero surface pressure gradient . in this analysis, the pressure gradient normal to the surface is included, and the outer boundary conditions are modified in accordance with the requirements of flow over a curved surface .

the results indicate that for equal reynolds numbers, the stress on convex surfaces is less than the flat-plate value, while the stress on concave surfaces is greater than for a flat plate . the most important effect of surface curvature, for the cases considered, is the modification of the shape of the velocity profile near the /outer edge/ of the boundary layer . the requirement that a smooth transition exist between the viscous flow and the potential flow at the outer edge of the layer causes the profile to have a negative slope near the outer edge for convex surface curvature and a positive slope for concave surface curvature .

.I 134

.T

note on an interaction between the boundary layer and the inviscid flow .

.A

antonio ferri and paul a. libby

.B

department of aeronautical engineering and applied mechanics, polytechnic institute of brooklyn, brooklyn, n.y.

.W

note on an interaction between the boundary layer and the inviscid flow .

according to the classical boundary-layer theory the flow about bodies at reynolds numbers of aeronautical interest can be considered as composed of two regimes.. an outside inviscid flow and a thin boundary-layer region adjacent to the body . this point of view leads to the approximation that, on a slightly curved surface, throughout the layer

is negligibly small . the additional assumption that the inviscid flow

is irrotational leads to the requirement that is zero at the outer edge

of the boundary layer . in this theory any interaction between the two

regimes is accountable by a simple correction to the body shape based on

the boundary-layer displacement thickness .

recently, in connection with hypersonic laminar boundary layers, this

classical point of view has been modified., an interaction between the

two flow regimes leading to a self-induced axial pressure gradient has

been considered . it is the purpose of the present note to point out

another type of interaction which may be of practical importance and of

fundamental interest even at mach numbers below those considered in the

hypersonic boundary-layer theory and which may have to be considered in

that theory .

.I 135

.T

the calculation of wall shearing stress from heat-transfer measurements

in compressible flows .

.A

nick s. diaconis

.B

lewis flight propulsion laboratory, naca, cleveland, ohio

.W

the calculation of wall shearing stress from heat-transfer measurements

in compressible flows .

it has been shown by ludwieg that the wall shearing stress of a laminar

or turbulent boundary layer in an incompressible flow can be determined from a heat-transfer measurement at the surface . the instrument

used in that investigation was essentially a small, locally insulated,

heating element embedded in the test surface . the size of the

instrument was restricted by the condition that the thermal boundary layer

generated by the heating element be contained locally within the laminar

sublayer . in the present analysis ludweig's theory for such an

instrument is extended to compressible flow over an insulated flat plate .

with the same limitations on the design and operation of the instrument

as mentioned above, it can also be assumed for compressible laminar

and turbulent boundary layers that only the flow in the immediate

vicinity of the wall or the laminar sublayer will be affected in the region

of the heated element . this assumption then permits the use of the

laminar boundary-layer equations as the governing equations for this

analysis for both laminar and turbulent boundary layers .

.I 136

.T

recent developments in rocket nozzle configurations .

.A

roa,g.v.r.

.B

a.r.s.jnl. 31, 1961,1488.

.W

recent developments in rocket nozzle configurations .

existing configurations of supersonic portion of rocket nozzles

are described and compared . survey covers bell-type conical and

contoured nozzles, annular nozzles, plug nozzles, and the author's

own /e-d/ (expansion-deflection) nozzle . the latter is a bell-type nozzle in which the gases are first deflected radially outward

by a small central plug, then expanded radially inward around the

base of the plug, and finally deflected back to a nearly axial

direction by the nozzle wall, in compressive turning .

.I 137

.T

the generation of sound by aerodynamic means .

.A

curle,n.

.B

j.roy.ae.s. 65, 1961, 724.

.W

the generation of sound by aerodynamic means .

a summary is given of some of the more important experimental results

relating to the noise radiated from a cold subsonic turbulent jet .

these are then related to the predictions of lighthill's general theory

of aerodynamic noise .

.I 138

.T

wakes in axial compressors .

.A

pearson,h. and mckenzie,a.b.

.B

j.roy.ae.s.63, 1959, 415.

.W

wakes in axial compressors .

the tendency in the past has been to assume that when wakes or non-uniform total head profiles are fed into an axial compressor then substantially constant static pressure prevails at the entry, the variations in total head appearing as variations in velocity . this variation in velocity causes variation in incidence on the early stage blade rows and thus can give rise to excitation of blade vibration . this assumption is implicit, for instance, in references 1 and 2, but we think has been a common assumption by most of the people working in this field .

where the compressor is fed by a duct of substantially parallel walls for a reasonable length ahead, such an assumption appeared justifiable . such a duct when given an air flow test with its outlet discharging, for instance, to atmosphere instead of to the compressor, then the distribution assumed would normally be obtained and in fact many surveys of such ducts have been represented in this fashion . the object of this note is to show that, in fact, this distribution will not normally occur when the compressor is present and we may normally expect much more nearly a constant velocity into the compressor with attendant static pressure distributions to match with the total head variations ahead of the intake, with of course, the attendant curved flow to support the static pressure gradients .

.I 139

.T

viscous effects on pitot tubes at low speeds .

.A

mcmillan,f.a.

.B

j.roy.ae.s.58, 1954, 570.

.W

viscous effects on pitot tubes at low speeds .

measurements were made of the pressure in a blunt-nosed pitot

tube, in an air stream at reynolds numbers from about 15 to 1000 .

the results are expressed in terms of a pressure coefficient

density of the fluid, and p and v are the static pressure and

velocity in the undisturbed stream . as found in previous

investigations, becomes greater than 1 at low reynolds numbers, the

increase being about at a reynolds number of 50 (based

on external tube radius) . in disagreement with the work of hurd,

chesky, and shapiro, no decrease of below 1 was found at any

reynolds number .

when the values of found by various experiments are

plotted against reynolds numbers based on internal tube radius,

it is found that the curves are in closer agreement than when the

external radius is used .

.I 140

.T

the determination of turbulent skin friction by means

of pitot tubes .

.A

preston,j.h.

.B

j.roy.ae.s. 58, 1954, 109.

.W

the determination of turbulent skin friction by means

of pitot tubes .

a simple method of determining

local turbulent skin friction on a smooth

surface has been developed which utilises a

round pitot tube resting on the surface .

assuming the existence of a region near the

surface in which conditions are functions

only of the skin friction, the relevant physical

constants of the fluid and a suitable length,

a universal non-dimensional relation is

obtained for the difference between the total

pressure recorded by the tube and the static

pressure at the wall, in terms of the skin

friction . this relation, on this assumption,

is independent of the pressure gradient .

the truth and form of the relation were first

established, to a considerable degree of

accuracy, in a pipe using four geometrically

similar round pitot tubes--the diameter

being taken as representative length . these four

pitot tubes were then used to determine

the local skin friction coefficient at three stations

on a wind tunnel wall, under varying

conditions of pressure gradient . at each station, within the limits of experimental accuracy, the deduced skin friction coefficient was found to be the same for each pitot tube, thus confirming the basic assumption and leaving little doubt as to the correctness of the skin friction so found . pitot traverses were then made in the pipe and in the boundary layer on the wind tunnel wall . the results were plotted in two non-dimensional forms on the basis already suggested and they fell close together in a region whose outer limit represented the breakdown of the basic assumption, but close to the wall the results spread out, due to the unknown displacement of the effective centre of a pitot tube near a wall . this again provides further evidence of the existence of a region of local dynamical similarity and of the correctness of the skin friction deduced from measurements with round pitot tubes on the wind tunnel wall . the extent of the region in which the local dynamical similarity may be expected to hold appears to vary from about to of the boundary-layer thickness for conditions remote from, and close to, separation respectively .

.I 141

.T

free-flight techniques for high speed aerodynamic research .

.A

hamilton,j.a. and hufton,p.a.

.B

j.roy.ae.s. 60, 1956, 151.

.W

free-flight techniques for high speed aerodynamic research .

the development rocket-borne and rocket-launched high-speed airplane model test is described . details of airborne components, telemetering units, tracking, and their calibration are also discussed . tests on controls, drag measurements, longitudinal stability evaluations, lift measurements, pressure measurements, aeroelastic estimations, and sonic bang recordings are effected . the reynolds numbers involved are much higher than are usual in the wind tunnel, and extensions of mach numbers are obtained beyond the tunnel limits, both free of the tunnel wall interference .

.I 142

.T

the problem of aerodynamic heating .

.A

van driest,e.r.

.B

aero.eng.rev. 15, 1956.

.W

the problem of aerodynamic heating .

paper is a good review of knowledge to date on convective heat transfer to objects moving through air at low and high speeds .

theoretical and experimental information is given on recovery

factors and heat-transfer coefficients for isothermal surfaces of

unswept flat plates, wedges and cones with attached shock waves,

and stagnation points of blunt bodies of revolution, for both

laminar and turbulent boundary layers . a convenient nomograph for

calculating flat plate turbulent boundary-layer heat-transfer

coefficients is given . effects of surface cooling, surface roughness,

and supply stream turbulence on transition are discussed and

shown graphically .

.I 143

.T

interplanetary orbits .

.A

vertregt,m.

.B

j.brit.inter.s. 16, 1958, 326.

.W

interplanetary orbits .

the basic equations under simplified

conditions for interplanetary flight are derived .

for a voyage from planet to planet an

unlimited number of orbits is possible . in

order to give a clear survey of these possible

orbits a diagram is developed from which the

approximate energy-requirement, the duration,

and other particulars of a voyage can be easily found .

.I 144

.T

heat flow in composite slabs .

.A

mayer,e.

.B

j.am.r.s. 22, 1952, 150.

.W

heat flow in composite slabs .

this paper presents the solution of the heat flow problem in composite walls under heat transfer conditions which are typical of uncooled rocket engine walls . analytic expressions in the form of fourier sums are obtained for the temperature distribution in a composite wall consisting of an inner (refractory) medium and an outer metallic) medium under newtonian heat transfer into the first medium with negligible heat transfer from the second medium to the exterior . the expressions obtained are based on a plane parallel composite slab as a representative model for relatively thin cylindrical walls, with thickness-to-radius ratio not exceeding 0.2 . the general results for the composite slab are simplified for the limiting cases of a thin refractory shield with a thick shielded medium and a thick refractory shield with a thin shielded medium .

.I 145

.T

skin friction in the laminar boundary layer in compressible

flow .

.A

young,a.d.

.B

aero.quart. 1, 1949, 137.

.W

skin friction in the laminar boundary layer in compressible

flow .

from an analysis of the work of

crocco and others, semi-empirical formulae

are derived for the skin friction on a

flat plate at zero incidence with a laminar

boundary layer . these formulae are

for the general case of heat transfer, and

when there is no heat transfer .

the problem of heat transfer and

the effect of radiation are discussed in the

light of these formulae . the second

formula is then utilised in the development of

an approximate method for solving the

momentum equation of the boundary layer

on a cylinder without heat transfer .

the method indicates that with increase of

mach number there is a marked forward

movement of separation from a flat plate in the presence of a constant adverse velocity gradient .

.I 146

.T

supersonic flow past slender bodies with discontinuous

profile slope .

.A

fraenkel and portnoy.

.B

aero.quart. 6, 1955, 114.

.W

supersonic flow past slender bodies with discontinuous

profile slope .

ward's slender-body theory is extended to derive first

approximations to the external forces on slender

bodies of general cross section

with discontinuous profile slope . two

classes of body are considered ..

bodies whose profile (typified by the local

radius) is continuous between the

nose and base, and certain bodies whose

profile is discontinuous, such as

bodies with annular or side air intakes and

wing-bodies on which the wing

has an unswept leading edge . (where air

intakes are concerned, it is

assumed that they are sharp-edged and that

there is no /spillage/ of the

internal flow) .

the following conclusions apply to

the former class of bodies . the

variation of drag with mach number is

found to depend only on the

discontinuities in the longitudinal rate of change

of the cross-sectional area, and is

thus independent of cross-sectional shape .

the drag itself is unchanged if

the direction of the flow is reversed . the

expressions for lift and moment

assume the same forms as for smooth pointed

bodies, the lift depending only

on conditions at the base of the body .

the general theory is applied to

winged bodies of revolution with an

unswept wing leading edge .. the results

bear a marked resemblance to those

obtained by ward . the results for wings

alone are seen to be applicable,

with one modification, to subsonic as well as to supersonic speeds .

.I 147

.T

supersonic flow past slender pointed wings with ?similar?

cross sections at zero lift .

.A

lord,w.t. and brebner,g.g.

.B

aero.quart. 10, 1959, 79.

.W

supersonic flow past slender pointed wings with ?similar?

cross sections at zero lift .

some recent theoretical work on slender pointed wings at zero lift is

co-ordinated and extended . the wings

considered may have any pointed plan

form shape, provided that the trailing

edge is straight and unswept . the root

section profile and cross-section shapes

are arbitrary, provided that, on any

one wing, the latter are /descriptively

similar/ (diamond or parabolic biconvex

for instance), though not necessarily

geometrically similar . the chief aim of

the work is to find wings with simple

geometry, low wave drag and pressure

distributions which are unlikely to be

seriously affected by viscous effects .

wave drag and pressure distributions

are calculated by slender-wing theory .

general formulae, which are both simple

and instructive, are given for the wave

drag and the overall pressure distribution,

with particular emphasis on the root

pressure distribution . results for a number of wings of special interest are

presented and discussed .

.I 148

.T

on displacement thickness .

.A

lighthill,m.j.

.B

j.fluid mech. 4, 1958, 383.

.W

on displacement thickness .

four alternative theoretical treatments of 'displacement

thickness', and, generally, of the influence of boundary layers

and wakes on the flow outside them, are set out, first for

two-dimensional, and then for three-dimensional, laminar or turbulent,

incompressible flow . they may be called the methods of 'flow

reduction', 'equivalent sources', 'velocity comparison' and

the principal expression obtained for the displacement

thickness in three-dimensional flow may be written

if, as orthogonal coordinates (x,y) specifying position on the

surface, we choose x as the velocity potential of the external

flow, and y as a coordinate, constant along the external-flow

streamlines, such that h dy is the distance between (x,y) and

z is the distance from the surface, u and v are the x and y components

of velocity, and u takes the value u just outside the boundary layer .

.I 149

.T

expansions at small reynolds number for the flow past

a sphere and a circular cylinder .

.A

proudman,i. and pearson,j.r.a.

.B

j.fluid mech. 2, 1957, 237.

.W

expansions at small reynolds number for the flow past

a sphere and a circular cylinder .

this paper is concerned with the problem of obtaining higher approximations to the flow past a sphere and a circular cylinder than those represented by the well-known solutions of stokes and oseen . since the perturbation theory arising from the consideration of small non-zero reynolds numbers is a singular one, the problem is largely that of devising suitable techniques for taking this singularity into account when expanding the solution for small reynolds numbers .

the technique adopted is as follows . separate, locally valid the regions close to, and far from, the obstacle . reasons are presented for believing that these 'stokes' and 'oseen' expansions are, respectively, of the forms

where are spherical or cylindrical polar coordinates made dimensionless with the radius of the obstacle, r is the reynolds number, and and vanish with r . substitution of these expansions in the navier-stokes equation then yields a set of differential equations for the coefficients and, but only one set of physical boundary conditions is applicable to each

expansion (the no-slip conditions for the stokes expansion, and
the uniform-stream condition for the oseen expansion) so that
unique solutions cannot be derived immediately . however, the
fact that the two expansions are (in principle) both derived from
the same exact solution leads to a 'matching' procedure which
yields further boundary conditions for each expansion . it is thus
possible to determine alternately successive terms in each
expansion .

the leading terms of the expansions are shown to be closely
related to the original solutions of stokes and oseen, and detailed
results for some further terms are obtained .

.I 150

.T

integration of the boundary layer equations .

.A

meksyn,d.

.B

proc.roy.s.a. 237 1956, 543.

.W

integration of the boundary layer equations .

the equations of the boundary layer

are integrated by an expression of the form

where f(x) is a positive function with x=0

as the stationary point,. (x) is slowly varying,.

the integral contains an unknown parameter

which is found from the condition .

the integral is evaluated by the method of steepest descent . the expressions obtained are

usually divergent, except in few cases which

include blasius's equation,. the divergent

expressions are summed by euler's transformation .

to check the procedure it is applied to falkner

and skan's equation . the results obtained

are very striking,. few terms in the expansions

are sufficient to obtain close agreement with

hartree's laborious numerical computations .

the method is also applied to the general

boundary-layer equation for the case of flow past

an elliptic cylinder, measured by schubauer .

the results obtained are in close agreement

with schubauer's measurements for the velocities,

almost up to separation, for the position of

the separation point,. and in satisfactory agreement downstream of

separation .

.I 151

.T

the generation of noise by isotropic turbulence .

.A

proudman,j.

.B

proc.roy.s.a 214,1952,119.

.W

the generation of noise by isotropic turbulence .

a finite region, with fixed boundaries, of an infinite expanse of compressible fluid is in

turbulent motion . this motion generates noise

and radiates it into the surrounding fluid .

the acoustic properties of the system are studied

in the special case in which the turbulent

region consists of decaying isotropic turbulence .

it is assumed that the reynolds number

of the turbulence is large, and that the mach number is small .

the noise appears to be generated mainly

by those eddies of the turbulence whose

contribution to the rate of dissipation of kinetic

energy by viscosity is negligible .

it is shown that the intensity of sound at large

distances from the turbulence is the same

as that due to a volume distribution of simple acoustic

sources occupying the turbulent region .

in this analogy, the whole fluid is to be regarded

as a stationary and uniform acoustic

medium . the local value of the acoustic power output

p per mass of turbulent fluid is given

approximately by the formula

where a is a numerical constant, u is the

mean-square velocity fluctuation, is the time, and

c is the velocity of sound in the fluid . the

constant a is expressed in terms of the well-known

velocity correlation function f(r) by

assuming the joint probability distribution of the turbulent velocities and their first two

time-derivatives at two points in space to be

gaussian . the numerical value is

then obtained by substituting the form of f(r)

corresponding to heisenberg's theoretical

spectrum of isotropic turbulence .

it is found that the effects of decay make

only a small contribution to the value of a, and

that the order of magnitude of a is not changed

when widely differing forms of the function

f(r) are used .

.I 152

.T

on the flow of compressible fluid past an obstacle .

.A

lord rayleigh, o.m., f.r.s.

.B

.W

on the flow of compressible fluid past an obstacle .

it is well known that according to classical hydrodynamics a steady

stream of frictionless incompressible fluid exercises no resultant

force upon an obstacle, such as a rigid sphere, immersed in it . the

development of a /resistance/ is usually attributed to viscosity, or

when there is a sharp edge to the negative pressure which may accompany

it (helmholtz) . in either case it would seem that resistance involves

something of the nature of a wake, extending behind the obstacle to an

infinite distance . when the system of disturbed velocities, although it may mathematically extend to infinity, remains as it were attached to the obstacle, there can be no resistance .

the absence of resistance is asserted for an incompressible fluid., but it can hardly be supposed that a small degree of compressibility, as in water, would affect the conclusion . on the other hand, high relative velocities, exceeding that of sound in the fluid, must entirely alter the conditions . it seems worth while to examine this question more closely, especially as the first effects of compressibility are amenable to mathematical treatment .

.I 153

.T

on the steady motion of viscous, incompressible fluids, with particular reference to a variation principle .

.A

millikan,c.b.

.B

phil.mag. 7, 1929, 641.

.W

on the steady motion of viscous, incompressible fluids, with particular reference to a variation principle .

except in exceptional cases, it is not possible to represent the motion of a viscous incompressible liquid by means of a variation principle, but all cases of such motion that have yet been discovered belong to this class of /exceptional cases ./ the appropriate functions are given .

.I 154

.T

velocity and temperature distributions in the turbulent

wake behind a heated body of revolution .

.A

hall,a.a. and hislop,g.s.

.B

proc.cam.phil.s. 34, 1938, 345.

.W

velocity and temperature distributions in the turbulent

wake behind a heated body of revolution .

recently (see abstract 954 (1938)) goldstein made calculations based on theories of vorticity transfer, of the distributions of velocity and temperature in the turbulent wake behind a heated body of revolution, and the present authors now record an experimental determination of these distributions in a low-turbulence wind tunnel . difficulty was experienced in obtaining a truly symmetrical wake and observations have been reduced to mean values, curves of which are given .

.I 155

.T

on the solution of the laminar boundary layer equations .

.A

howarth,l.

.B

proc.roy.s.a, 164, 1938, 547.

.W

on the solution of the laminar boundary layer equations .

the problem of the flow along a flat plate

placed edgewise to a steady stream,

when a retarding pressure gradient
varying linearly as the distance x
from the leading edge of the plate
is superposed is discussed . if y denotes
distance measured perpendicular to
the plate, a solution is obtained in the
form of a power series in x where
coefficients are functions of .
differential equations are obtained for these
coefficients . seven of the coefficients
have been obtained with reasonable
accuracy, and the eighth and ninth
roughly . unfortunately it appears
that about eight more terms are
required to carry the solution to
the point of separation,. the work
involved in their determination is
prohibitive . two approximate methods
have been developed for determining
the error when the first seven terms
of the series are used as an approximation .
these methods lead to the
determination of the point of separation
and are in agreement as to its
position . if is the velocity at the
edge of the boundary layer at the
leading edge of the plate and is the

velocity gradient, separation is found

when . a method is

developed for the solution of the

boundary layer equations in any retarded region .

it is obtained by

replacing the velocity distribution at the edge

of the boundary layer by a

circumscribing polygon of infinitesimal sides and

applying the preceding solution

to each of these sides, making the

momentum integral continuous at each

vortex . the problem is thereby

reduced to the solution of a first order

differential equation .

.I 156

.T

the effect of shallow water on wave resistance .

.A

havelock,t.h.

.B

proc.roy.s.a, 100,1922,499.

.W

the effect of shallow water on wave resistance .

the general character of experimental

results dealing with the effect of

shallow water on ship resistance may be

stated briefly as follows ..--at low

velocities the resistance in shallow
water is greater than in deep water, the
speed at which the excess is first
appreciable varying with the type of vessel .
as the speed increases, the excess
resistance increases up to a maximum at a
certain critical velocity, and then
diminishes . with still further increase of
speed, the resistance in shallow
water ultimately becomes, and remains, less
than that in deep water at the same speed .
the maximum effect is the more
pronounced the shallower the water .
for further details and references one
may refer to standard treatises, but one
quotation may be made in regard to
the critical velocity .. /this maximum
appears to be at about a speed such
that a trochoidal wave travelling at this
speed in water of the same depth is
about times as long as the vessel .
it was at one time supposed
that the speed for maximum increase
in resistance was that of the wave of
translation . this, however, holds only
for water whose depth is less than
for greater depths the speed of the

wave of translation rapidly becomes

greater than the speed of maximum

increase of resistance ./ in a recent

analysis of the data, h. m. weitbrecht

expresses a similar conclusion by stating

that for each depth of water there is

a critical velocity, but that the critical

velocity does not vary as the square

root of the corresponding depth .

.I 157

.T

the hodographic transformation in transonic flow .

.A

lighthill,m.j.

.B

proc. roy.s. a, 191, 1947, 323.

.W

the hodographic transformation in transonic flow .

the author studies the problem of finding the shape of a

symmetrical nozzle with the velocity along the axis (x-axis)

specified . the velocity along each streamline is assumed to

increase steadily . the singularity at the sonic velocity and

to the axis of the nozzle) is first studied in the physical

plane by using a power series in . in the hodograph plane,

the two characteristics of the hodograph differential

equation passing through the sonic point and are lines

of branch points . the region between these lines is a region of triple-valuedness for the stream function .

outside this region is single-valued . there are also

singularities at the sonic point and the point corresponding to

the specified condition at the exit of the nozzle . the author

then proposes to construct in the hodograph plane by

at the exit velocity and (3) a finite sum regular throughout ..

sin, where r is the square of the velocity

and the are hypergeometric functions . the a's are

fixed by the required approximation to the specified velocity

distribution along the axis . this solution is single-valued,

convergent and represents except a region near the sonic

point in the nozzle . for this excluded region, the author

inverts the solution to obtain a power series in for 0 . this

is shown to be convergent for the region of interest . the

type of solution considered by the author gives a nozzle

having an infinitely long supersonic part .

.I 158

.T

temperature charts for induction and constant temperature

heating .

.A

heisler,m.p.

.B

a.s.m.e.trans. 14, 1947, 227.

.W

temperature charts for induction and constant temperature

heating .

charts are presented for determining complete temperature historics in spheres, cylinders, and plates . it is shown that for values of the dimensionless time ratio x greater than 0.2 the heating equations reduce to such a simple form that for each shape two charts which give temperatures at any position within the heated or cooled bodies can be plotted . it is also shown that the usual simple heating and cooling charts can also be used for the determination of temperatures and heating times in bodies heated by a constant rate of heat generation at the surface (induction heating) . finally, a two dimensional chart is given for finding heating times in short cylinders, thereby eliminating the trial-and-error solution that is necessary when heating times are found from the present one-dimensional charts .

.I 159

.T

numerical methods for transient heat flow .

.A

dusinberre,g.m.

.B

a.s.m.e.trans. 12, 1945, 703.

.W

numerical methods for transient heat flow .

this paper deals with the application of numerical methods for the solution of heat-conduction problems, their generality being extended in the following ways ..

may proceed most rapidly to a solution, or may proceed

more slowly and with greater precision,. (b) criteria are

developed for the choice of modulus to insure convergence .

this is most important at a convective surface,. (c) a

method is developed for handling k and c when these

properties vary independently with temperature . a

comprehensive appendix gives the derivations, and the use of

equations and charts is demonstrated by typical

examples .

.I 160

.T

approximate analytical solutions for hypersonic flow

past slender power-law bodies .

.A

mirels,h.

.B

nasa r-15, 1959.

.W

approximate analytical solutions for hypersonic flow

past slender power-law bodies .

approximate analytical solutions are presented

for two-dimensional and axisymmetric

hypersonic flow over blunt-nosed slender bodies whose

shapes follow a power law variation . in particular,

the body shape is given by where is the

transverse body ordinate, is the streamwise distance

from the nose, and m is a constant in the range .

both zero-order
solutions and first-order (small but nonvanishing values
of solutions are presented, where m is the
free-stream mach number and is a characteristic
body or streamline slope . the zero-order shock shape
is similar to the body shape for these flows . the
solutions are found within the framework of
hypersonic-slender-body theory .
the limiting case m=1 corresponds to a wedge
or cone flow . the limiting case
corresponds to a constant-energy flow .
the latter cases are included
so that the present study may be applied to all flows
wherein the zero-order shock shape is given by
with m in the range . flow
fields associated with shock shapes having values of m
outside this range are also discussed . for all values
of, except m=1, certain portions of the flow field
riolate the hypersonic-slender-body approximations,
while other portions are consistent with these
approximations . for m=1, all portions of the flow field
are consistent with the approximations .
the approximate solutions are found as follows .
the asymptotic form of the flow in the vicinity of the
body surface is used as a guide to write approximate
expressions for the dependent variables . these
expressions exactly satisfy the continuity and

energy equations and contain arbitrary constants which are evaluated so as to satisfy boundary conditions at the shock . the approximate solutions do not satisfy the lateral momentum equation except at the shock and (for the first-order problem) at the body surface .

the results of the approximate solutions are compared with numerical integrations of the equations of motion for various values of m and (ratio of specific heats) . good agreement is noted, particularly when m and are both near one . the shock is relatively close to the body for the latter cases . sufficient results are presented to evaluate the accuracy of the approximate method for various values of m and .

.I 161

.T

supersonic flow past a family of blunt symmetric bodies .

.A

van dyke,m.d. and gordon,h.d.

.B

nasa r-1, 1959.

.W

supersonic flow past a family of blunt symmetric bodies .

some 100 numerical computations have been carried out for unyawed bodies of revolution with detached bow waves . the gas is assumed perfect with . free-stream mach numbers

are taken as 1.2, 1.5, 2, 3, 4, 6, 10, and . the

results are summarized with emphasis on the sphere

and paraboloid .

.I 162

.T

nearly circular transfer trajectories for descending

satellites .

.A

low,g.m.

.B

nasa r-3, 1959.

.W

nearly circular transfer trajectories for descending

satellites .

  simplified expressions describing the transfer

from a satellite orbit to the point of atmospheric

entry are derived . the expressions are limited to

altitude changes that are small compared with the

earth's radius, and velocity changes small compared

with satellite velocity . they are further restricted

to motion about a spherical, nonrotating earth .

  the transfer orbit resulting from the application

of thrust in any direction at any point in an elliptic

orbit is considered . expressions for the errors in

distance (miss distance) and entry angle due to an

initial misalinement and magnitude error of the deflecting thrust are presented .

the largest potential contributing factor towards

a miss distance stems from the misalinement of the

retrovelocity increment . if this velocity increment is

pointed in direct opposition to the flight path, a 1

misalinement leads to a miss distance of 34.5 miles .

however, it is shown that this error can be avoided

by applying the velocity increment at an angle

between 120 and 150 below the flight-path direction .

the guidance and accuracy requirements to

establish a circular orbit, in addition to the corrections

applied to transform elliptic orbits into circular

ones, are also discussed .

.I 163

.T

an analysis of the corridor and guidance requirements

for supercircular entry planetary atmospheres .

.A

chapman,d.r.

.B

nasa r-55, 1959.

.W

an analysis of the corridor and guidance requirements

for supercircular entry planetary atmospheres .

an analysis is presented of supercircular entry

into a planet's atmosphere giving particular attention

to the corridor through which spacecraft must be

guided in order to accomplish various maneuvers .
a dimensionless parameter based on conditions at the conic perigee altitude is introduced for characterizing supercircular entries and conveniently prescribing corridor widths associated with elliptic, parabolic, or hyperbolic approach trajectories . the analysis applies to vehicles of arbitrary weight, shape, and size . illustrative calculations are made for venus, earth, mars, jupiter, and titan .

for nonlifting vehicles having fixed aerodynamic coefficients, curves are presented of dimensionless parameters from which can be calculated the maximum deceleration, maximum rate of laminar convective heating, and total laminar heat absorbed during single-pass entry at velocities up to twice circular velocity . for lifting vehicles, curves are presented of the maximum deceleration and overshoot boundary of an entry corridor,. equations are presented for estimating laminar aerodynamic heating from the maximum deceleration . it is shown that the corridor width is independent of vehicle weight, dimensions, and drag coefficient, provided these are the same at the overshoot boundary as at undershoot . the corridors of certain planets can be broadened markedly by the application of aerodynamic lift,. for example, the 10-earth-g corridor width for single-pass, nonlifting, parabolic entry is increased from to 52, 51, and 52 miles, respectively, by employing a lift-drag ratio of 1 . the use of aerodynamic lift does not increase appreciably the corridors of mars and titan . all corridor widths decrease rapidly as the entry velocity is increased .

terminal guidance requirements on accuracy of velocity and flight path angle for successfully entering various corridors are compared with analogous requirements for putting a satellite into orbit, for hitting the moon from the earth, and for achieving icbm accuracy . consideration is given to the terminal guidance problem involved in using a planet's atmosphere--rather than rocket fuel--to effect orbital transfers from heliocentric to planeto-centric motion, thereby converting a hyperbolic approach trajectory to an elliptic orbit about the target planet . this fuel saving maneuver appears technologically feasible for certain planetary voyages, and implies the possibility of achieving a large reduction in required earth lift-off weight of chemical propulsion systems .

.I 164

.T

an approximate analytical method for studying entry into planetary atmospheres .

.A

chapman,d.r.

.B

nasa r-11, 1959.

.W

an approximate analytical method for studying entry into planetary atmospheres .

the pair of motion equations for entry into a planetary atmosphere is reduced to a single, ordinary, nonlinear differential equation of second order by disregarding two relatively small terms and by introduring a certain mathematical transformation . the reduced equation includes various terms, certain of which represent the gravity force, the centrifugal acceleration, and the lift force . if these particular terms are disregarded, the differential equation is linear and yields precisely the solution of allen and eggers applicable to ballistic entry at relatively steep angles of descent . if all the other terms in the basic equation are disregarded (corresponding to negligible vertical acceleration and negligible vertical component of drag force), the resulting truncated differential equation yields the solution of sanger for equilibrium flight of glide vehicles with relatively large lift-drag ratios .

a number of solutions for lifting and nonlifting vehicles entering at various initial angles also have been obtained from the complete nonlinear equation . these solutions are universal in the sense that a single solution determines the motion and heating

of a vehicle of arbitrary weight, dimensions, and

shape entering an arbitrary planetary atmosphere .

one solution is required for each lift-drag ratio .

these solutions are used to study the deceleration,

heating rate, and total heat absorbed for entry into

venus, earth, mars, and jupiter . from the

equations developed for heating rates, and from

available information on human tolerance limits

to acceleration stress, approximate conditions for

minimizing the aerodynamic heating of a trimmed

vehicle with constant lift-drag ratio are established

for several types of manned entry . a brief study

is included of the process of atmosphere braking for

slowing a vehicle from near escape velocity to near

satellite velocity .

.I 165

.T

skin-friction measurements in incompressible flow .

.A

smith, d.w. and walker, j. h.

.B

naca report r-26

.W

skin-friction measurements in incompressible flow .

experiments have been conducted to measure in

incompressible flow the local surface-shear stress

and the average skin-friction coefficient for a turbulent
boundary-layer on a smooth flat plate having zero pressure gradient .
the local surface-shear stress was measured by a floating-element
skin-friction balance and also by a calibrated total head
tube located on the surface of the test wall . the
average skin-friction coefficient was obtained from
boundary-layer velocity profiles . the
boundary-layer profiles were also used to determine the location
of the virtual origin of the turbulent boundary layer .
data were obtainec for a range of reynolds numbers
from 1 million to about 45 million with an attendant
change in mach number from 0.11 to 0.32 .

the measured local skin-friction coefficients
obtained with the floating-element balance agree
well with those of schultz-grunow and kempf
for reynolds numbers up to 45 million . the
measured average skin-friction coefficients agree
with those given by the schoenherr curve in the
ranges of reynolds numbers from 1 to 3 million
and 30 to 45 million . in the range of reynolds
numbers from 3 to 30 million the measured values
are less than those predicted by the schoenherr curve .
the results show that the /univeral skin-friction constants/
proposed by coles appraoch asymptotically
a constant value at reynolds numbers exceeding
mentioned constants and the limited reynolds
number range of the present investigation, there is some doubt

as to the validity of any turbulent

skin-friction law written on the basis of the present

results . hence, no new friction law is proposed .

the frictional resistance of a flat plate was

calculated by means of the momentum method and

also the integrated measured local surface shear .

for reynolds numbers from 14 million to 45 million

both methods give about the same result,. whereas

at lower values of reynolds number the momentum

method based on velocity profiles uncorrected for

the effects of turbulence results in a frictional

resistance as much as 4 percent higher than that

of the integrated shear .

the measurement of local surface shear by a

calibrated preston tube appears to be accurate

and inexpensive . the calibration as given by

preston must be modified slighlty, however, to yield the

results obtained from the floating-element

skin-friction balance .

.I 166

.T

flow of chemically reacting gas mixtures .

.A

clarke,j.f.

.B

coa r117, 1961.

.W

flow of chemically reacting gas mixtures .

suitable forms of the equations

for the flow of an inviscid,

non-heat-conducting gas in which chemical

reactions are occurring are derived .

the effects of mass diffusion and

non-equilibrium amongst the internal

modes of the molecules are neglected .

special attention is given to

the speeds of sound in such a gas

mixture and a general expression for

the ratio of frozen to equilibrium

sound speeds is deduced . an example

is given for the ideal dissociating

gas . the significance of the velocity

defined by the ratio of the convective

derivatives of pressure and density is

explained . it is the velocity

which exists at the throat of a

convergent-divergent duct under maximum

mass flow conditions, and it is shown that

this velocity depends on the

nozzle geometry as well as on the 'reservoir' conditions .

as an illustration the phenomena of

sound absorption and dispersion are

discussed for the ideal dissociating gas .

the results can be concisely

expressed in terms of the frozen and

equilibrium sound speeds, the

frequency of the (harmonic) sound

vibration and a characteristic time for

the rate of progress of the reaction .

.I 167

.T

linearized flow of a dissociating gas .

.A

clarke,j.f.

.B

j.fluid mech. 7, 1960, 577.

.W

linearized flow of a dissociating gas .

the equations for planar two-dimensional

steady flow of an ideal dissociating gas

are linearized, assuming small disturbances

to a free stream in chemical

equilibrium .

as an example of their solution, the

flow past a sharp corner in a supersonic

stream is evaluated and the variations

of flow properties in the relaxation zone

are found . numerical illustrations are

provided using an 'oxygen-like' ideal gas

and comparisons made with a characteristics

solution . the flow past a sharp

corner can be studied in a conventional

shock tube and it may be possible to

verify the present theory experimentally .

in particular it may prove feasible to

use the results to obtain a measure of the

reaction rates in the gas mixture .

.I 168

.T

heat conduction through a gas with one inert internal

model .

.A

clarke,j.f.

.B

coa n102, 1960.

.W

heat conduction through a gas with one inert internal

model .

the rate of energy transfer between

parallel flat plates is evaluated

when the (stagnant) gas between them is

polyatomic with one inert internal

mode . deviations of the thermal

conductivity from the complete equilibrium

of the inert mode relaxation time

and the effectiveness of the walls in

exciting or de-exciting this mode .

the results are obtained via a linear

theory consistent with small
temperature differences between the plates .
it is found that the eucken-value
of conductivity could be exceeded
if the relaxation times are non-zero and
the plates very effective in
exciting the inert mode . when relaxation
times are very short the effect
of the walls on the energy transfer rate
is small, but the walls make
their presence felt by distorting the
temperature profiles in /boundary
layers/ adjacent to the walls which are
of order in thickness
time) . this result is
analogous to hirschfelder's (1956) for the
case of chemical reactions .
for experimental measurement of
conductivity in a hot wire cell type
of apparatus it is shown that extrapolation
of measured reciprocal
conductivities to zero reciprocal pressure
should load to the full eucken
value . it is also shown that the slope of
reciprocal apparent (measured)
conductivity versus reciprocal pressure
curves is a function of relaxation

time as well as of the accommodation

coefficients . it is quite possible

that the relaxation effect here is

comparable with the temperature jump

effects, even for rotation in diatomic molecules .

.I 169

.T

on the sudden contact between a hot gas and a cold solid .

.A

j. f. clarke, b.sc., ph.d., a.f.r.ae.s.

.B

.W

on the sudden contact between a hot gas and a cold solid .

the flow induced by the sudden contact between a semi-infinite expanse

of gas and a solid, initially at different temperatures, is examined on

the basis of a linear continuum theory . for times large compared with

the mean time between molecular collisions in the gas, the velocity

and pressure disturbances are found to be concentrated around a wave

front propagating out from the interface at the ambient isentropic sound

speed, whilst, near to the interface, these disturbances are small and

the gas temperatures are nearly equal to those predicted by the

classical constant pressure heat conduction theory .

the possible significance of these results in connection with

reflected shock wave techniques to measure high temperature gas properties is

commented upon .

.I 170

.T

the interaction of a reflected shock wave with the boundary layer in a shock tube .

.A

mark,h.

.B

naca tm.1418.

.W

the interaction of a reflected shock wave with the boundary layer in a shock tube .

ideally, the reflection of a shock from the closed end of a shock tube provides, for laboratory study, a quantity of stationary gas at extremely high temperature . because of the action of viscosity, however, the flow in the real case is not one-dimensional, and a boundary layer grows in the fluid following the initial shock wave .

in this paper simplifying assumptions are made to allow an analysis of the interaction of the shock reflected from the closed end with the boundary layer of the initial shock afterflow . the analysis predicts that interactions of several different types will exist in different ranges of initial shock mach number .

it is shown that the cooling

effect of the wall on the afterflow

boundary layer accounts for the change

in interaction type .

an experiment is carried out which

verifies the existence of the

several interaction regions and shows

that they are satisfactorily

predicted by the theory . along with these

results, sufficient information

is obtained from the experiments to make

possible a model for the

interaction in the most complicated case .

this model is further verified

by measurements made during the experiment .

the case of interaction with a

turbulent boundary layer is also

considered . identifying the type of

interaction with the state of

turbulence of the interacting boundary

layer allows for an estimate of the

state of turbulence of the boundary

layer based on an experimental

investigation of the type of interaction .

.I 171

.T

a low density wind tunnel study of shock wave structure and relaxation phenomena in gases .

.A

sherman,f.s.

.B

naca tn.3298.

.W

a low density wind tunnel study of shock wave structure

and relaxation phenomena in gases .

the profiles and thicknesses

of normal shock waves of moderate

strength have been determined

experimentally in terms of the variation

of the equilibrium temperature

of an insulated transverse cylinder in

free-molecule flow . the shock

waves were produced in a steady state in

the jet of a low-density wind

tunnel, at initial mach numbers of 1.72

and 1.82 in helium and 1.78,

the shock thickness, determined

from the maximum slope of the cylinder

temperature profile, varied from

mean free path in the supersonic

stream . a comparison between the

experimental shock profiles and various

theoretical predictions leads to the

tentative conclusions that .. (1)

the navier-stokes equations are adequate

for the description of the shock

transition for initial mach numbers up

to 2, and (2) the effects of

rotational relaxation times in air can be

accounted for by the introduction

of a /second/ or /bulk/ viscosity

coefficient equal to about two-thirds

of the ordinary shear viscosity .

.I 172

.T

some aerodynamic considerations of nozzle afterbody

combination .

.A

cortright,e.m.

.B

aero. eng. rev. 15, 1956, 59.

.W

some aerodynamic considerations of nozzle afterbody

combination .

the aerodynamic problems associated with

propulsion-system installations have assumed a role

of vital importance in the development of supersonic

aircraft . although air-induction systems have received

moderate attention in the literature, considerably less

information can be found on the design and installation

of turbojet exit nozzles . this condition should not be interpreted to indicate a lack of problems in jet-exit design .

as flight speeds reach supersonic levels, it becomes increasingly difficult to achieve nozzle installations which are efficient over the entire speed range . the difficulties largely stem from the fact that the goals of high jet thrust and low afterbody drag are not always compatible . in many of the compromise solutions, it is generally unsatisfactory to examine isolated nozzle and afterbody performance . rather they must be treated as a unit, and the complex effects of jet interaction with the external stream must be taken into account . to accomplish this, the nozzle and air-frame designers must closely coordinate their efforts .

some of the aerodynamic problems of nozzle afterbody combinations are outlined in this report . particular attention is devoted to the influence of the jet-stream interaction on both nozzle thrust and after-body drag . for this purpose, use is made of shock-boundary-layer-interaction concepts . this approach, although not precise, correctly predicts many trends and is generally enlightening .

.I 173

.T

the effect of a central jet on the base pressure of a cylindrical afterbody in a supersonic stream .

.A

reid,j. and hastings,r.c.

.B

arc r + m.3224, 1962.

.W

the effect of a central jet on the base pressure of

a cylindrical afterbody in a supersonic stream .

this report describes an experimental investigation

of the factors affecting the base flow and

jet structure behind a cylindrical after-body with a central nozzle .

seven interchangeable nozzles were tested .

six of these were convergent-divergent, with a design mach

number of 2.0, jet base diameter ratios ranging

from 0.2 to 0.8 and nozzle divergence angles ranging from

convergent with a jet base diameter ratio of 0.6 .

in the main experimental programme the free-stream

mach number was 2.0 and the boundary layer was

turbulent both on the after-body and in the nozzle .

measurements were made of the base pressure, the surface

pressure distribution inside the nozzle, the overall thrust

and the nozzle mass flow, over a range of jet pressures .

this programme was supplemented by comparative

tests with the jet exhausting into still air (static tests) .

readings were taken of the internal nozzle pressures

and the jet thrust at different jet pressures . schlieren

photography was used extensively throughout .

the results of the tests with external flow are

presented in the form of curves showing the separate effects of jet pressure ratio, jet base diameter ratio, nozzle

design mach number and nozzle divergence angle on the

base pressure and overall thrust . the special case

of base bleed is discussed separately . similar curves are

included for the static tests . these show the effect

of jet pressure ratio and nozzle geometry on the jet thrust .

a general method of correlating data on annular

base pressures is proposed and discussed . essentially, this

method compares the pressure on an annular

base with the calculated pressure on the corresponding

two-dimensional base . it correlates the present

results reasonably well, but is less successful when applied

to more extensive data .

.I 174

.T

investigation at supersonic speeds of the effects of

jet mach number and divergence angle of the nozzle

upon the pressure of the base annulus of a body of

revolution .

.A

bromm,a.f. and o'donnel,r.m.

.B

naca rm l54i16, 1954.

.W

investigation at supersonic speeds of the effects of

jet mach number and divergence angle of the nozzle

upon the pressure of the base annulus of a body of revolution .

an investigation has been conducted in the langley 9-inch supersonic tunnel to determine the jet effects for varying jet mach number and nozzle divergence angle upon the pressure on the base annulus of a model with a cylindrical afterbody . the tests were conducted over a wide range of jet static pressure ratios and at a reynolds number of approximately free-stream mach numbers of 1.62, 1.94, and 2.41 . all testing was conducted with an artificially induced turbulent boundary layer along the model .

in the lower range of jet static pressure ratios, jet flow from a sonic or supersonic nozzle affected the pressure acting on the base annulus in essentially the same manner as shown in naca rm e53h25 which covers jet static pressure ratios up to about present results showed that the base pressure tends to level off with increasing jet static pressure ratio, and at the extreme static pressure ratios reached in tests with sonic nozzles the base pressure began to

decrease . except in the lower range

of jet static pressure ratios,

nozzle divergence angle generally had a

larger effect on the base pressures

than nozzle mach number,. the increase

in base pressure for a change in

divergence angle from 0 to 10 was

small compared to the increase when

the divergence angle was changed from

and other data indicates that the effects

of divergence angle were reduced

when the ratio of jet exit diameter to base

diameter was decreased . jet

mach number effects increased with increase in stream mach number .

.I 175

.T

experiments with static tubes in a supersonic airstream .

.A

holder,d.w., north,r.j. and chinneck,a.

.B

arc r + m 2782, 1953.

.W

experiments with static tubes in a supersonic airstream .

systematic tests have been made at a mach

number of 1.6 on a family of static tubes . the variables

which have been investigated are the shape of the nose, the distance of the holes downstream, and the inclination of

the tube to the flow . pressure measurements have also been

made in the vicinity of a shock wave and close to a wall .

.I 176

.T

base pressure at subsonic speeds in the presence of

a supersonic jet .

.A

craven,a.h.

.B

coa r129, 1960.

.W

base pressure at subsonic speeds in the presence of

a supersonic jet .

this paper presents the results of an experimental investigation

into the effect of supersonic jets upon the base pressure of a bluff

cylinder in a uniform subsonic flow . the ratio of jet diameter to base

diameter was 0.1875 .

jet stagnation pressures giving slight under-expansion of the jet

cause an increase in the base pressure but for larger jet stagnation

pressures the base pressure is again reduced .

a simple theory, based on a momentum integral, shows the dependence

of the base drag upon the jet and free stream speeds and upon the

dimensions of the jet and the base .

.I 177

.T

the mixing of free axially-symmetrical jets of mach number 1.40 .

.A

n. h. johannesen

.B

department of the mechanics of fluids, university of manchester

communicated by the director-general of scientific research (air),

ministry of supply

.W

the mixing of free axially-symmetrical jets of mach number 1.40 .

axially-symmetrical, supersonic, fully-expanded jets of diameter about 0.75 in. and of mach number 1.40 issuing into an atmosphere at rest were investigated by schlieren and shadow photography and by pressure traversing . the development of the jets was found to depend critically on the strength of the shock waves in the core of the jet at the nozzle exit . with strong shock waves present the jet spread very rapidly and was very unsteady . the jet did in some cases break up into large eddies of the same size as the diameter of the jet . when no disturbances were present in the core of the jet the spreading was far more gradual and the jet showed only slight unsteadiness . the turbulent mixing region of the first part of the jet with strong shock waves was investigated in detail by pitot tubes . the first inch was found to correspond to a two-dimensional half-jet . the velocity profiles were similar and well represented by the error integral . the rate of spreading was only half the value for low-speed flow . by integrations across the mixing region the entrainment and the loss of kinetic energy were determined . these quantities were found to agree well with the values estimated by assuming an error-integral velocity profile .

.I 178

.T

on full dispersed shock waves in carbon dioxide .

.A

griffith,w.c. and kenny,a.

.B

j.fluid mech. 3, 1957, 286.

.W

on full dispersed shock waves in carbon dioxide .

it is pointed out that, for shock mach numbers between 1 and

that the adjustments in the energy in all the degrees of freedom

proceed slowly and in parallel and occur over a distance large

compared with the mean free path . theoretical velocity profiles

for such shock waves are given and found to be in excellent

agreement with interferometric shock-tube observations .

.I 179

.T

an analysis of base pressure at supersonic speeds and

comparison with experiment .

.A

chapman,d.

.B

naca tn.2137, 1950.

.W

an analysis of base pressure at supersonic speeds and

comparison with experiment .

in the first part of the

investigation an analysis is made of base
pressure in an inviscid fluid,
both for two-dimensional and
axially-symmetric flow . it is shown that
for two-dimensional flow, and also for
the flow over a body of revolution
with a cylindrical sting attached to
the base, there are an infinite
number of possible solutions satisfying
all necessary boundary conditions
at any given free-stream mach number .
for the particular case of a body
having no sting attached only one
solution is possible in an inviscid
flow, but it corresponds to zero
base drag . accordingly, it is concluded
that a strictly inviscid-fluid
theory cannot be satisfactory for practical applications .
since the exact inviscid-fluid
theory does not adequately describe
the conditions of a real fluid flow,
an approximate semi-empirical theory
for base pressure in a viscous fluid
is developed in a second part of the
investigation . the semi-empirical
theory is based partly on
inviscid-flow calculations, and is restricted

to airfoils and bodies without

boat-tailing . in this theory an attempt

is made to allow for the effects of

mach number, reynolds number, profile

shape, and type of boundary-layer

flow . the results of some recent

experimental measurements of base

pressure in two-dimensional and

axially-symmetric flow are presented for

purposes of comparison . some

experimental results also are presented

concerning the support interference

effect of a cylindrical sting, and

the interference effect of a reflected

bow wave on measurements of base

pressure in a supersonic wind tunnel .

.I 180

.T

boundary layer over a flat plate in presence of shear

flow .

.A

ting,l.

.B

phys. fluids, 13, 1960, 78.

.W

boundary layer over a flat plate in presence of shear

flow .

the governing equations of an incompressible

boundary layer over a flat plate in the presence of a

shear flow with finite vorticity are derived . for large

vorticity, a similarity solution is obtained . for

moderate vorticity, one of the governing equations

is replaced by an approximate one for which

similarity solutions exist .

.I 181

.T

some problems on heat conduction in stratiform bodies .

.A

vodicka,v.

.B

j. phys. soc. japan, 14, 1959, 216.

.W

some problems on heat conduction in stratiform bodies .

problems on heat conduction in multilayer bodies lead usually to

complicated calculations . the present paper gives an idea of specific

difficulties arising in the case of infinite composite solides .

general deductions are applied to a special class of questions .

.I 182

.T

effect of roughness on transition in supersonic flow .

.A

van driest,e.r. and blumer,c.b.

.B

agard r255, 1960.

.W

effect of roughness on transition in supersonic flow .

further experiments carried out in the 12-inch supersonic wind tunnel

of the jet propulsion laboratory of

the california institute of technology

to investigate the effect of

three-dimensional roughness elements (spheres)

on boundary-layer transition on a 10

transfer are reported herein . the

local mach number for these tests was

minimum (effective) size of trip

required to bring transition to its

lowest reynolds number varies as the

one-fourth power of the distance from

the apex of the cone to the trip .

use of available data at other mach

numbers indicates that the mach

number influence for effective tripping

is taken into account by the

simple expression .

.I 183

.T

properties of impact pressure probes in free molecule

flow .

.A

harris,e.l. and patterson,g.n.

.B

utia r52, 1958.

.W

properties of impact pressure probes in free molecule

flow .

an expression has been derived for the mass flow through

a circular tube in free molecule flow when the tube and gas are in

relative motion . the gas entering the tube is assumed to have a

maxwellian distribution function and the molecular reflection process

at the wall is assumed to be diffuse .

the theory has been used to determine the pressure read

by an impact probe in free molecule flow . although the general

expressions derived apply to any value of gas velocity and tube size,

the detailed calculations for the pressure probe are difficult except

for the case of low speeds and long tubes .

an experimental check of the theory has been carried out

using impact probes in a whirling arm apparatus and in the utia low

density wind tunnel . agreement between theory and experiment is

quite satisfactory .

.I 184

.T

scale models for thermo-aeroelastic research .

.A

molyneux,w.g.

.B

rae tn.struct.294, 1961.

.W

scale models for thermo-aeroelastic research .

an investigation is made of the
parameters to be satisfied for
thermo-aeroelastic similarity . it is concluded
that complete similarity obtains
only when aircraft and model are identical
in all respects, including size .

by limiting consideration to
conduction effects, by assuming the major
load carrying parts of the structure
are in regions where the flow is either
entirely laminar, or entirely turbulent,
and by assuming a specific
relationship between reynolds number and nusselt
number, an approach to similarity can
be achieved for small scale models .
experimental and analytical work is
required to check on the validity of these assumptions .

it appears that existing hot wind
tunnels will not be completely
adequate for thermo-aeroelastic work, and
accordingly a possible layout for
the type of tunnel required is described .
automatic programmed control of
the tunnel would appear to be necessary .

.I 185

.T

some possibilities of using gas mixtures other than in aerodynamic research .

.A

dean r. chapman

.B

.W

some possibilities of using gas mixtures other than in

aerodynamic research .

a study is made of the advantages that can be realized in

compressible-flow research by employing a substitute heavy gas in place of air .

most heavy gases considered in previous investigations are either toxic,

chemically active, or (as in the case of the freons) have a ratio of

specific heats greatly different from air . the present report is based

on the idea that by properly mixing a heavy monatomic gas with a

suitable heavy polyatomic gas, it is possible to obtain a heavy gas

mixture which has the correct ratio of specific heats and which is

nontoxic, nonflammable, thermally stable, chemically inert, and

comprised of commercially available components .

calculations were made of wind-tunnel characteristics for 63 gas pairs

comprising 21 different polyatomic gases properly mixed with each of

three monatomic gases (argon, krypton, and xenon) . for a given

mach number, reynolds number, and tunnel pressure, a gas-mixture

wind tunnel having the same specific-heat ratio as air would be

appreciably smaller and would require much less power than a

corresponding air wind tunnel . analogous though different advantages

can be realized in compressor research and in firing-range research .

the most significant applications, perhaps, arise through selecting and

proportioning a gas mixture so as to have at ordinary wind-tunnel temperatures certain dimensionless characteristics which air at

flight temperatures possesses but which air at ordinary wind-tunnel

temperatures does not possess . characteristics which involve the

relaxation time (or bulk viscosity), the variation of viscosity with

temperature, and the variation of specific heat with temperature fall

within this category . other applications arise in heat-transfer

research since certain gas mixtures can be concocted to have any prandtl

number in the range at least between 0.2 and 0.8 .

.I 186

.T

base pressure in supersonic flow .

.A

gadd,g.e., holder,d.w. and regan,j.d.

.B

arc cp271, 1956.

.W

base pressure in supersonic flow .

the problem of accurately predicting the pressure and wake

configuration at the base of bodies in supersonic flow is an

extremely important one inasmuch as a sizeable portion of the total

drag of a given body may be attributable to the low pressure in

this region . although a great deal of theoretical and experimental

work has been done in this field, there does not yet exist a

satisfactory method for accurate predictions .

this paper represents an excellent effort to experimentally

confirm analytically deduced concepts . a large amount of

experimental data on body shapes such as wedges, cones, and cone-cylinders has been obtained over a range of mach numbers up to 4 .

the data are thoroughly discussed with respect to analytical

deductions . on the basis of the evidence accumulated it is

concluded that the boundary-layer thickness has only a small effect

on the base pressure for axisymmetric bodies and for

two-dimensional bodies when the base height-to-chord ratios are of

the order .

reviewer believes this report is a significant contribution in the

field of base pressure and wake flow phenomena .

.I 187

.T

investigation of separated flows in supersonic and subsonic

streams with emphasis on the effect of transition .

.A

chapman, d.r., kuehn, d. m. and larson, h. k.

.B

naca report 1356

.W

investigation of separated flows in supersonic and subsonic

streams with emphasis on the effect of transition .

experimental and theoretical research has been conducted on

flow separation associated with steps, bases, compression corners,

curved surfaces, shock-wave boundary-layer reflections, and

configurations producing leading-edge separation . results were

obtained from pressure-distribution measurements,

shadow-graph observations, high-speed motion pictures, and oil-film

optics . the maximum scope of measurement encompassed

mach numbers between 0.4 and 3.6, and length reynolds
numbers between 4000 and 5000000 .
the principal variable controlling pressure distribution in
the separated flows was found to be the location of transition
relative to the reattachment and separation positions .
classification is made of each separated flow into one of three regimes ..
and /turbulent/ with transition upstream of separation .
by this means of classificaiton it is possible to state rather
literal results regarding the steadiness of flow and the influence
of reynolds number within each regime .
for certain pure laminar separations a theory for calculating
dead-air pressure is advanced which agrees well with subsonic
and supersonic experiments . this theory involves no empirical
information and provides an explanation of why transition
location relative to reattachment is important . a simple analysis
of the equations for interaction of boundary-layer and external
flow near either laminar or turbulent separation indicates the
pressure rise to vary as the square root of the wall shear stress
at the beginning of interaction . various experiments substantiate tnis
variation for most test conditions . an incidental
observation is that the stability of a separated laminar mixing
layer increases markedly with an increase in mach number .
the possible significance of this observation is discussed .

.I 188

.T

an analysis of base pressure at supersonic velocities and
comparison with experiment .

.A

chapman, dean r .

.B

naca report 1051

.W

an analysis of base pressure at supersonic velocities and

comparison with experiment .

in the first part of the investigation an analysis is made of base pressure in an inviscid fluid, both for two-dimensional and axially symmetric flow . it is shown that for two-dimensional flow, and also for the flow over a body of revolution with a cylindrical sting attached to the base, there are an infinite number of possible solutions satisfying all necessary boundary conditions at anh given free-stream mach numger . for the particular case of a body having no sting attached only one solution is possible in an inviscid flow, but it corresponds to zero base drag . accordingly, it is concluded that a strictly inviscid-flow theory cannot be satisfactory for practical applications .

an approximate semi-empirical analysis for base pressure in a viscous fluid is developed in a second part of the investigation . the semi-empirical analysis is based partly on inviscid-flow calculations . in this theory an attempt is made to allow for the effects of mach number, reynolds number, profile shape, and type of boundary-layer flow . some measurements of base pressure in two-dimensional and axially symmetric flow are presented for purposes of comparison . experimental results then are presented concerning the support interference effect

of a cylindrical sting, and the interference effect of a reflected

air wave on measurements of base pressure in a supersonic wind tunnel .

.I 189

.T

experimental investigation of base pressure on blunt-trailing-edge

wings of supersonic velocities .

.A

chapman,d.r., wimbrow,w.r. and kester,r.h.

.B

naca r1109.

.W

experimental investigation of base pressure on blunt-trailing-edge

wings of supersonic velocities .

measurements of base pressure are presented for 29

blunt-trailing-edge wings having an aspect ratio of 3.0 and various

airfoil profiles . the different profiles comprised thickness

ratios between 0.05 and 0.10, boattail angles between --2.9

and 20, and ratios of trailing-edge thickness to airfoil thickness

between 0.2 and 1.0 . the tests were conducted at mach numbers

of 1.25, 1.5, 2.0, and 3.1 . for each mach number, the reynolds

number and angle of attack were varied . the lowest reynolds

number investigated was 0.2 x 10 and the highest was 3.5 x 10 .

measurements on each wing were obtained separately with

turbulent flow and laminar flow in the boundary layer .

span-wise surveys of the base pressure were conducted on several

wings .

the results with turbulent boundary-layer flow showed only small effects on base pressure of variations in reynolds number, airfoil profile shape, boattail angle, and angle of attack . the principal variable affecting the base pressure for turbulent flow was the mach number . at the highest mach number investigated (3.1), the ratio of boundary-layer thickness to trailing-edge thickness also affected the base pressure significantly .

the results obtained with laminar boundary-layer flow to the trailing edge showed that the effect of reynolds number on base pressure was large . in all but a few exceptional cases the effects on base pressure of variations in angle of attack and in profile shape upstream of the base were appreciable though not large . the principal variable affecting the base pressure for laminar flow was the ratio of boundary-layer thickness to trailing-edge thickness .

for a few exceptional cases involving laminar flow to the trailing edge, the effects on base pressure of variations in profile shape, boattail angle, and angle of attack were found to be unusually large . in such cases the variation of base pressure with angle of attack was discontinuous and exhibited a hysteresis . stroboscopic schlieren observations at a mach number of 1.5 indicated that these apparently special phenomena were associated with a vortex trail of relatively high frequency .

.I 190

.T

on magnetohydrodynamic shock waves .

.A

kanwal,r.

.B

j.math.mech. 9, 1960, 681.

.W

on magnetohydrodynamic shock waves .

in the earlier attempts at finding the jump conditions

across a hydromagnetic shock wave (1, 2, 3)

various simplifying assumptions

regarding the shape of the shock and the

dimensions and the character of the

motion are made . from that analysis it

is possible to write down the jump

conditions in a higher degree of generality (4) .

the shock conditions for magnetohydrodynamic

flows can, however, be

derived in their full generality with the help

of the transport equation as used by

thomas (5) in the derivation of shock conditions

in conventional gas dynamics .

the purposes of this paper are ..

cover the present more general case .

that every flow and field quantity

downstream from the shock wave is

expressible separately in terms of

the known values of these quantities

upstream from the shock wave .

in this rearranged form of the equations, various effects of the shock

wave can be easily read off .

the shock conditions along the same

lines as in conventional gas dynamics .

.I 191

.T

a theory for the core of a leading edge vortex .

.A

hall,m.g.

.B

rae r.aero.2644, 1960.

.W

a theory for the core of a leading edge vortex .

in the flow past a slender delta wing

at incidence can be observed a

roughly axially symmetric core of spiralling

fluid, formed by the rolling up

of the shear layer that separates from a

leading edge . the aim in this

report is to predict the flow field within

this vortex core, given

appropriate conditions at its outside edge .

the basic assumptions are

core .

in addition it is assumed that the flow

is axially symmetric and incompressible .

together, these admit outer and inner

solutions for the core from the equations

of motion .

for the outer solution the sub-core

is ignored, and the flow is taken to

be inviscid (but rotational) and conical .

the resulting solution consists of

simple expressions for the velocity components

and pressure . for the inner

solution, which applies to the diffusive

sub-core, the flow is taken to be

laminar, and approximations, some based on

the boundary conditions and some

analogous to those of boundary layer theory,

are made . the solution obtained

in this case is a first approximation, and is

presented in tabular form .

a sample calculation yields results

which are in good qualitative and

fair quantitative agreement with experimental measurements .

.I 192

.T

on the hypersonic viscous flow past slender bodies

of revolution .

.A

yashura,m.

.B

j.phys.soc. japan, 11, 1956, 878.

.W

on the hypersonic viscous flow past slender bodies

of revolution .

a similar solution of the hypersonic viscous flow past slender bodies of revolution is deduced for a special case when the radial coordinate of the body surface at section x is proportional to x, where the radial coordinate have the comparable order value with the thickness of the boundary layer . here, /similar/ is used in the direct meaning that distributions in the boundary layer keep the similar form lengthwise . calculations are accomplished for the region of strong interaction between the boundary layer and the shock wave . from several calculations it may be expected that if the thickness of the body becomes small, the thickness of the layer in which the longitudinal velocity component u is rapidly decreased also becomes small, and in the major part of the boundary layer, only the normal component v is increased . further if the thickness of the body is increased, then, the height of the shock wave, the pressure on the wall, and the shear stress at the wall are also increased while the boundary layer thickness is decreased . the nose region is excluded by the reason that the ordinary boundary layer theory will be invalid there .

.I 193

.T

a study of inviscid flow about air foils at high supersonic

speeds .

.A

eggers, a.j., syvertson, c.a., and krqus, s.

.B

naca report 1123

.W

a study of inviscid flow about air foils at high supersonic speeds .

steady flow about curved airfoils at high supersonic speeds is investigated analyticially . with the assumption that air behaves as a diatomic gas, it is found the the shock-expansion method may be used to predict the flow about curved airfoils up to extremely high mach numbers, provided the flow deflection angles are not too close to those corresponding to shock detachment . this result applies not only to the determination of the surface pressure distribution, but also to the determination of the whole flow field about an airfoil . verification of this observation is obtained with the aid of the method of characteristics by extensive calculations of the pressure gradient and shock-wave curvature at the leading edge, and by calculations of the pressure distribution on a 10-percent-thick biconvex airfoil at 0 angle of attack .

an approximation to the shock-expansion method for thin airfoils at high mach numbers is also investigated and is found to yield pressures in error by less than 10 percent at mach numbers above three and flow deflection angles up to 25 . this slender-airfoil method is relatively simple in form and thus may prove useful for some engineering purposes .

effects of caloric imperfections of air manifest in disturbed flow fields at high mach numbers are investigated, particular attention being given to the reduction of the ratio of specific

heats . so long as this ratio does not decrease appreciably below
to include the effects of these imperfections, should be substantially
as accurate as for ideal-gas flows . this observation is
verfied with the aid of a generalized shock-expansion method and a
generalized method of characteristics employed in forms applicable
for local air temperatures up to about 5000 rankine .
the slender-airfoil method is modified to employ an average
value of the ratio of specific heats for a particular flow field .
this simplified method has essentially the same accuracy for
imperfect-gas flows as its counterpart has for ideal-gas flows .
an approximate flow analysis is made at extremely high mach
numbers where it is indicated that the ratio of specific heats may
approach close to 1 . in this case, it is found that the
shock-expansion method may be in considerable error,. however, the
busemann method for the limit of infinite free-stream mach
number and specific-heat ratio of 1 appears to apply with
reasonable accuracy .

.I 194

.T

general theory of airfoil sections having arbitrary
shape or pressure distribution .

.A

allen,h.j.

.B

naca r833, 1945.

.W

general theory of airfoil sections having arbitrary shape or pressure distribution .

in this report a theory of thin airfoils of small camber is developed which permits either the velocity distribution corresponding to a given airfoil shape, or the airfoil shape corresponding to a given velocity distribution to be calculated . the procedures to be employed in these calculations are outlined and illustrated with suitable examples .

.I 195

.T

correlation of theoretical and photo-thermoelastic results on thermal stresses in idealized wing structure .

.A

tramposch,h. and gerard,g.

.B

j.app.mech. 27, 1960.

.W

correlation of theoretical and photo-thermoelastic results on thermal stresses in idealized wing structure .

after a rather complete exploratory program described in previous papers, the photo-thermoelastic method was applied to the experimental evaluation of the thermal-stress theories . the new technique was correlated with several theories which analyzed the transient thermal stresses in idealized wing structures of high-speed aircraft . various theories were investigated which represented the same idealized wing models and

differed from each other only in the simplifying

assumptions regarding the temperature

distributions in skin and webs . the theories

were evaluated by duplicating the boundary

and initial conditions on plastic models and

then by correlating the theories with the

observed fringe orders in nondimensional form .

a significant general conclusion was

reached after correlating the available theories

and experimental results . owing to

simplifying assumptions concerning the thermal

behavior in the flanges, thermal

stresses predicted by the available theories are all

higher than the experimental

observation . in some cases the discrepancy is as great as 30 per cent .

.I 196

.T

pressure distributions . axially symmetric bodies in

oblique flow .

.A

campbell,i.j. and lewis,r.g.

.B

arc cp213, 1955.

.W

pressure distributions . axially symmetric bodies in

oblique flow .

a simple picture, known from the work of i. lotz, of the flow over the forward part of a body of revolution in oblique flow is derived here from entirely elementary considerations . the pressure at any point of the (forward part of the) body at any angle of incidence depends on three parameters whose values vary along the body . the variation of these parameters along the body can be determined from a relatively small number of wind tunnel or water tunnel measurements . the necessary water tunnel measurements have been made for four axially symmetric head shapes . additional measurements have been made to illustrate the theoretical conclusions . the data for each head shape are adequate for a determination of the pressure coefficient at any point on the head shapes at any angle of incidence (up to 6, say) . in particular they can be used to determine the peak suction at any angle of incidence and so the conditions for the onset of cavitation on the head .

.I 197

.T

pressure distributions on three bodies of revolution to determine the effect of reynolds number up to and including the transonic speed range .

.A

swihart,j.m. and whitcomb,c.f.

.B

naca rm l53h04, 1953.

.W

pressure distributions on three bodies of revolution to determine the effect of reynolds number up to and including the transonic speed range .

this paper presents the results of an investigation conducted in the langley 16-foot transonic tunnel to determine the effects of varying reynolds number on the pressure distribution on a transonic body of revolution at angles of attack through the transonic speed range . the effect of a change in sting cone angle on the pressure distributions and a comparison of experimental incremental pressures with theory is also included .

the models were tested through a mach number range from 0.60 to 1.09 . the reynolds number range based on body length was from 9 x 10 to 39 x diameter was 1.3 x 10 to 4.53 x 10 for the model at 8 angle of attack .

an increase in reynolds number from 9 x 10 to 39 x 10 affected the longitudinal pressure distributions very slightly . these effects were of such a nature as to cause an increase of 0.05 in the normal-force coefficient of the body when tested in the subcritical cross-flow reynolds number range . this increase is in agreement with theoretical approximations .

a comparison between experimental and theoretical values of the incremental pressure coefficient due to angle of attack indicated good agreement except at angles where separated flow areas existed over the body .

the effect of a change in sting-cone angle from 5 to 9 on the pressure distribution of the 120-inch model was negligible up to a mach number of 1.05 . at this mach number the effect was to cause a small increase in the velocity over the rear of the body .

.I 198

.T

investigation of a systematic group of naca 1 - series

cowlings with and without spinners .

.A

nichols,m.r. and keith,a.l.

.B

naca r950, 1949.

.W

investigation of a systematic group of naca 1 - series

cowlings with and without spinners .

an investigation has been conducted in the langley propeller-research tunnel to study cowling-spinner combinations based on the naca 1-series nose inlets and to obtain systematic design data for one family of approximately ellipsoidal spinners . in the main part of the investigation, 11 of the related spinners were tested in various combinations with 9 naca open-nose cowlings, which were also tested without spinners . the effects of location and shape of the spinner, shape of the inner surface of the cowling lip, and operation of a propeller having approximately oval shanks were investigated briefly . in addition, a study was conducted to determine the correct procedure for extrapolating design conditions determined from the low-speed test data to the design conditions at the actual flight mach number .

the design conditions for the naca 1-series cowlings and cowling-spinner combinations are presented in the form of charts from which, for wide ranges of spinner proportions and rates of internal flow, cowlings with near-maximum pressure recovery can be selected for critical mach numbers ranging from

spinners and the effects of the spinners and the propeller on

the cowling design conditions are presented separately to

provide initial quantitative data for use in a general design

procedure through which naca 1-series cowlings can be

selected for use with spinners of other shapes . by use of this

general design procedure, correlation curves established from

the test data, and derived compressible-flow equations relating

the inlet-velocity ratio to the surface pressures on the cowling

and spinner, naca 1-series cowlings and cowling-spinner

combinations can be designed for critical mach numbers as

high as 0.90 .

.I 199

.T

measurement of two dimensional derivatives on a wing-aileron-tab

system .

.A

wight,k.c.

.B

part i, arc r + m 2934, 1955. part ii arc r + m 3029, 1958.

.W

measurement of two dimensional derivatives on a wing-aileron-tab

system .

measurements have been made of the direct

two-dimensional damping and stiffness derivatives for a

in incompressible flow .

corrections arising from the apparatus are discussed and reference is made to an attempt to measure the direct tab derivatives .

the effects are shown of frequency parameter, amplitude of oscillation, reynolds number, aileron angle and position of transition on the wing .

variation with frequency parameter is substantially the same as for vortex-sheet theory and variation of amplitude produces little change in both derivatives . at the lowest reynolds number there is little change in both derivatives with variation of aileron angle for the condition of natural transition, but at higher reynolds numbers the stiffness derivatives increase at .

a forward movement of transition reduces the stiffness derivatives at the smaller aileron angles, but at, at the lowest reynolds number, an increase results .

similar trends are observed for the damping derivatives above .

comparison with vortex-sheet theory shows that the measured values of the stiffness and damping derivatives are approximately 0.6 of the theoretical values .

measurements have been made of the direct tab derivatives and cross aileron-tab derivatives for a per cent aileron and 4 per cent (approx.) tab . in addition some measurements of the direct aileron derivatives have been made for comparison with earlier results together with a number of static derivatives for the wing and controls .

the influence is shown of frequency parameter, reynolds number, position of transition, mean tab angle and sealing of the control hinge gaps . some tests have been made with

the ailcron set at minus 8 deg and the tab at plus 12 deg

for which condition the hinge moment on the aileron was zero .

reasonable agreement with the values given by the /equivalent

profile/ theory is shown for both direct damping

derivatives and for the direct tab stiffness derivative . the direct

aileron stiffness derivative shows some departure from

the theoretical value when .

at and the natural transition, comparison

with the values given by flat-plate theory gives the

following approximate factors, where suffix denotes the

theoretical values ..

.I 200

.T

calculation of derivatives for a cropped delta wing

with subsonic leading edges oscillating in a supersonic

airstream .

.A

watson,j.

.B

arc r + m 3060, 1958.

.W

calculation of derivatives for a cropped delta wing

with subsonic leading edges oscillating in a supersonic

airstream .

the lift, pitching moment and full-span

constant-chord control hinge-moment are derived for a cropped delta wing describing harmonic plunging and pitching

oscillations of small amplitude and low-frequency parameter in

a supersonic air stream . it is assumed that (a) the wing

has subsonic leading edges, (b) the wing is sufficiently thin

and the mach number sufficiently supersonic to permit the

use of linearised theory .

  expressions for the various derivative coefficients are

obtained for a particular delta wing of aspect ratio 1.8 and

taper ratio these are avaluated and tabulated for mach

numbers 1.1, 1.15, 1.2, 1.3, 1.4, 1.5, 1.6 and 1.944 .

.I 201

.T

supersonic flow past oscillating airfoils including

nonlinear thickness effects .

.A

van dyke,m.d.

.B

naca r1183, 1954.

.W

supersonic flow past oscillating airfoils including

nonlinear thickness effects .

  a solution to second order in thickness is derived for

harmonically oscillating two-dimensional airfoils in supersonic

flow . for slow oscillations of an arbitrary profile, the result is

found as a series including the third power of frequency . for

arbitrary frequencies, the method of solution for any specific

profile is indicated, and the explicit solution derived for a single wedge .

nonlinear thickness effects are found generally to reduce the torsional damping, and so to enlarge the range of mach numbers within which torsional instability is possible . this destabilizing effect varies only slightly with frequency in the range involved in dynamic stability analysis, but may reverse to a stabilizing effect at high flutter frequencies . comparison with a previous solution exact in thickness suggests that nonlinear effects of higher than second order are practically negligible .

the analysis utilizes a smoothing technique that replaces the actural problem by one involving no kinked streamlines . this stratagem eliminates all consideration of shock waves from the analysis, yet yields the correct solution for problems that actually contain shock waves .

.I 202

.T

aircraft flutter .

.A

williams,j.

.B

arc r + m 2492, 1951.

.W

aircraft flutter .

the term flutter is used here to denote maintained or violent oscillations of a structure due to aerodynamic forces acting in conjunction with both elastic and inertial forces . attention is restricted to this particular branch of the more general field of aeroelasticity, which embraces buffeting, divergence, and reversal of control, as well as flutter,. airscrew flutter is not specifically considered . the monograph is divided into three main parts, each of which has been made self-contained for the convenience of readers .

in the first part, general methods for the investigation of aircraft flutter, by theoretical analysis and by experiments on flutter models, are set out and discussed . a detailed account of the aerodynamic theory of wings in non-uniform motion is not included, since this has already been provided elsewhere, but methods for the evaluation of the aerodynamic forces required in a theoretical flutter analysis are logically developed, and a bibliography of researches on the aerodynamic theory is given in the appendix . investigations on specific types of aircraft flutter--namely wing flutter, control surface flutter, and tab flutter--are discussed in part these various types of flutter are considered, but the practical details of flutter-prevention devices are omitted . finally, in part 3, methods for the experimental determination of airloads on oscillating aerofoil systems are described, and available airload measurements are analysed and compared with theoretical results .

an attempt has been made to refer in the text to all relevant british work reported by the early part of 1947 . foreign work has been mentioned in parts 1 and 2 only where necessary for the sake of completeness, but in part 3 and the appendix all relevant foreign references known to the author have been included .

matrix notation has been used for the theoretical treatment in

part 1, but otherwise its use has been avoided .

.I 203

.T

calculated velocity distributions and force derivatives

for a series of high-speed aerofoils .

.A

sinott,c.s.

.B

arc r + m 3045.

.W

calculated velocity distributions and force derivatives

for a series of high-speed aerofoils .

the polygon method of woods is used to

calculate the velocity distribution over a number of

two-dimensional aerofoils at low incidence, subcritical flows only

being considered . lift slopes and aerodynamic centres

at zero lift are also calculated .

some comparisons with experimental results are made, and

these show good agreement at zero incidence .

.I 204

.T

a study of the application of airfoil section data

to the estimation of the high subsonic speed characteristics

of swept wings .

.A

hunton,l.w.

.B

naca rm a55c23, 1955.

.W

a study of the application of airfoil section data

to the estimation of the high subsonic speed characteristics

of swept wings .

estimates of the variation with

mach number of the aerodynamic

characteristics of swept wings are made

on the basis of airfoil section

data combined with span-loading theory .

the analysis deals with

examinations of some 26 wings and wing-body

combinations ranging in sweep

angle from 30 to 60 and for mach

numbers between 0.6 and 1.0 .

results of the study indicate

that the two-dimensional section data

afford good qualitative information

for such high-speed aerodynamic

characteristics as the variation with

mach number of drag, zero-lift

pitching-moment coefficient, and lift

coefficient for flow separation .

quantitative estimates of the force

and moment divergence mach numbers

could not be made with any degree of

certainty from the airfoil data

alone . somewhat improved quantitative

estimates for a given

configuration were obtainable by basing the

estimates on the measured

characteristics for a wing of similar plan form

but different section, and

adjusting for the effects of differences in

section on the basis of section data .

.I 205

.T

a correlation of airfoil section data with the aerodynamic

loads measured on a 45 sweptback wing at subsonic mach

numbers .

.A

walker,h.j. and maillard,w.c.

.B

naca rm a55c08, 1955.

.W

a correlation of airfoil section data with the aerodynamic

loads measured on a 45 sweptback wing at subsonic mach

numbers .

an investigation has been made of the possibility of correlating

airfoil section data with measured pressure distributions over a 45

sweptback wing in the mach number range from 0.50 to 0.95 at a

free-stream reynolds number of approximately 2 million .

the wing had an aspect ratio

of 5.5, a taper ratio of 0.53, naca 64a010 sections normal to the

quarterchord line, and was mounted on a slender body of revolution .

at mach numbers of 0.85 and below, and for wing normal-force

coefficients below the maximum normal-force coefficient for an

infinite-aspect-ratio wing yawed 45 to the flow (derived from airfoil section

data by simple sweep relations), good correlation was obtained over most

of the wing between wing-section and two-dimensional-airfoil pressure

distributions . for greater normal-force coefficients lateral

boundary-layer flow permitted the inboard wing sections to rise to high maximum

section normal-force coefficients . the effectiveness of this lateral

boundary-layer flow disappeared towards the tip . for all mach numbers,

the influence of plan-form effects on the pressure distributions limited

the quality of the correlation at the 20- and 95-percent-semispan

stations . above a mach number of about 0.85 the shock waves

originating at the juncture of the body and the wing trailing edge spread over

the span, preventing further application of two-dimensional data .

the spanwise load distributions at moderate normal-force coefficients

could be predicted from span-loading theory for the entire mach number

range of the tests .

.I 206

.T

the applications of the polygon method to the calculation

of the compressible subsonic flow round two-dimensional

profiles .

.A

woods,l.c.

.B

arc cp115, 1953.

.W

the applications of the polygon method to the calculation

of the compressible subsonic flow round two-dimensional

profiles .

this paper sets out the method now used by the author of

applying the polygon method to the calculation of the compressible

subsonic flow round two-dimensional aerofoils . tables have been

constructed which can be used for all aerofoil shapes, putting the

polygon method on the same footing numerically as goldstein's

method has the advantage over approximation 3 that it can be applied in

the following cases which are beyond the scope of goldstein's method ..

conventional aerofoils, (b) the low-speed flow about very thick

aerofoils, e.g., in reference 3 it is applied to circular cylinders, (c)

the flow about symmetric aerofoils between either straight or constant

pressure walls, (d) flow in asymmetric channels,

and (e) more difficult problems

of the flow about aerofoils in the presence

of one or two constraining

walls (to be published) . a method of

calculating lift and moment

coefficients, and their rates of change

with incidence (a) is also

given in the paper .

as an example the velocity distribution and the rates of

change of the lift and moment coefficients with a are calculated for the aerofoil r.a.e.104 at values of m (mach number at infinity) of 0,

and 0.7, for various values of the incidence, a . the velocity

distributions for zero incidence are found to be in fair agreement with

the corresponding experimental results . the results at incidence are

in satisfactory agreement with the experimental results, not for the

same incidence, but for the same lift coefficient . it is found, for

example, that at m = 0.7 the theory for a = 0.8 agrees best

with experiment for a = 1.0, when the lift coefficients are

approximately the same .

.I 207

.T

laminar boundary layer oscillations and transition

on a flat plate .

.A

schubauer,g.b. and skramstad,h.k.

.B

naca r909, 1948.

.W

laminar boundary layer oscillations and transition

on a flat plate .

this is an account of an investigation in which oscillations

were discovered in the laminar boundary layer along a flat plate .

these oscillations were found during the course of an experiment

in which transition from laminar to turbulent flow was being

studied on the plate as the turbulence in the wind stream was

being reduced to unusually low values by means of damping

screens . the first part of the paper deals with experimental

methods and apparatus, measurements of turbulence and
sound, and studies of transition . a description is then given
of the manner in which oscillations were discovered and how
they were found to be related to transition, and then how
controlled oscillations were produced and studied in detail . the
oscillations are shown to be the velocity variations
accompanying a wave motion in the boundary layer, this wave motion having
all the characteristics predicted by a stability theory based on
the exponential growth of small disturbances . a review of this
theory is given . the work is thus experimental confirmation
of a mathematical theory of stability which had been in the
process of development for a period of approximately 40 years,
mainly by german investigators .

.I 208

.T

the hall effect in the viscous flow of ionized gas
between parallel plates under transverse magnetic field .

.A

sato,h.

.B

j.phys.soc. japan, 16, 1961, 1427.

.W

the hall effect in the viscous flow of ionized gas
between parallel plates under transverse magnetic field .
the electrical conductivity of an ionized
gas is anisotropic in the
presence of magnetic field (hall effect) .

the conductivity is expressed by

a tensor in the same form for both fully

and partially ionized gases . by

the use of modified ohm's law and

conventional magnetohydrodynamical

equations the incompressible viscous

flow between parallel plates under

the transverse magnetic field is analyzed

and an exact solution is obtained

when the magnetic reynolds number

is small . the numerical results

reveal a remarkable effect of anisotropy

of conductivity . the acceleration

and deceleration of viscous ionized

gas under combined electric and

magnetic fields are also calculated .

.I 209

.T

boundary layer induced noise in the interior of aircraft .

.A

ribner,h.s.

.B

utia r37, 1956.

.W

boundary layer induced noise in the interior of aircraft .

  at high speeds the turbulent boundary layer washing the

airplane fuselage excites appreciable skin vibration, promoting strong noise in the interior . the fluctuating exciting pressure distribution can be represented as a pattern of moving waves (fourier integral) . a running ripple in the skin follows underneath each wave, and the noise is ultimately due to these ripples .

the acoustic effects of the running ripples are calculated for an infinite sheet,. this is considered the main result of the paper . supersonically moving ripples radiate strong sound in the form of mach waves,. subsonically moving ripples radiate no sound . formulas for the mean square surface pressure and the energy flux are obtained for an assumed idealized turbulent pressure spectrum .

the results are adapted to provide a tentative estimate of the noise generated at subsonic speeds in a practical fuselage . the running ripples are almost noise-free, but multiple reflections at the frames and stringers promote standing waves . an assumption is used to link the two kinds of waves, and this leads to provisional calculations of noise level .. on this basis the noise level is predicted to vary as for thin boundary layers, changing progressively to for thick layers (= external air density, = speed, = layer thickness, = panel thickness) . some comparisons are made with experiment . finally, an idea for minimizing the noise is presented .

.I 210

.T

propeller in yaw .

.A

ribner,h.s.

.B

naca r820, 1945.

.W

propeller in yaw .

it was realized as early as 1909 that a propeller in yaw develops a side force like that of a fin . in 1917, r. g. harris expressed this force in terms of the torque coefficient for the unyawed propeller . of several attempts to express the side force directly in terms of the shape of the blades, however, none has been completely satisfactory . an analysis that incorporates induction effects not adequately covered in previous work and that gives good agreement with experiment over a wide range of operating conditions is presented herein . the present analysis shows that the fin analogy may be extended to the form of the side-force expression and that the effective fin area may be taken as the projected side area of the propeller . the effective aspect ratio is of the order of 8 and the appropriate dynamic pressure is roughly that at the propeller disk as augmented by the inflow . the variation of the inflow velocity, for a fixed-pitch propeller, accounts for most of the variation of side force with advance-diameter ratio v nd .

the propeller forces due to an angular velocity of pitch are also analyzed and are shown to be very small for the pitching velocities that may actually be realized in maneuvers, with the exception of the spin .

further conclusions are .. a dual-rotating propeller in yaw develops up to one-third more side force than a single-rotating propeller . a yawed single-rotating propeller experiences a pitching moment in addition to the side force . the pitching moment is of the order of the moment produced by a force equal to the side force, acting at the end of a lever arm equal to the propeller radius . this cross-coupling between pitch and yaw is small but possibly not negligible .

the formulas for propellers in yaw derived herein (with the exception of the compressibility correction) and a series of charts of the side-force derivative calculated therefrom have been presented without derivation in an earlier report .

.I 211

.T

effect of slight blunting of leading edge of an immersed body on the flow around it at hypersonic speed .

.A

chernyi,g.g.

.B

nasa tt f-35, 1960.

.W

effect of slight blunting of leading edge of an immersed body on the flow around it at hypersonic speed .

manufacturing and maintainance of ideally sharp leading edges and noses is practically impossible, hence a discrepancy arises between the theory established for sharp edges and actual flow around slightly blunted edges, where a detached shock is formed with a subsonic adjacent region . semi-empirical method is worked out showing that the pressure distribution in the vicinity of the leading edge is the same for different thin profiles having the same shape of bluntness on their edges or noses . the data for a flat plate can be used for all of them . for moderate supersonic speed the pressure on the remaining body is practically unaffected by the nose bluntness, and can be computed from a sharp-edge theory . for high supersonic speed a slight blunting of the edge can considerably alter the pattern of flow over a large region . the method consists in replacing blunted edge by action of concentrated forces on the flow,. it is applied to blunted wedge where it shows doubling of the drag computed by classic theory, and to cones, where the drag of a blunted cone may become smaller than that of a sharp one .

.I 212

.T

theory and tunnel tests of rotor blade for supersonic turbines .

.A

stratford,b.s. and sansome,g.e.

.B

arc r + m 3275, 1960.

.W

theory and tunnel tests of rotor blade for supersonic turbines .

in special circumstances where a large work output is required from a turbine in a single stage it is necessary to use high pressure ratios across the nozzle blades, thus producing supersonic velocities at inlet to the rotor . as part of an investigation into such turbines, several designs for the inter-blade passages of the rotor have been tested in a two-dimensional tunnel, a design theory being developed concurrently .

the first design, featuring constant passage width and curvature as in steam-turbine practice, but having thin leading and trailing edges, was found to suffer from focusing of the compression waves from the concave surface, with consequent flow separation from the opposite convex surface . it gave a velocity coefficient of measured at an inlet mach number of 1.90 and turning angle of 140 deg . the measured value compares favourably with values from previous steam tests, where the results have been in the range from 0.65 to 0.92 .

from theoretical reasoning, and from additional test observations, a subsequent passage was designed having an inlet transition length of small curvature, leading to a free-vortex passage of double the transition curvature,. a small amount of contraction was incorporated . schlieren photographs showed the flow in this passage to be almost shock free . a thin region of low-energy air existed close to the convex surface, but liquid-injection tests located only one small bubble of reversed flow . pressure traverses at exit indicated a velocity coefficient of 0.952, based on the area-mean total pressure . when allowance is made for turning angle and reynolds number this result appears to compare quite favourably with previous work .

it would seem that the optimum blade pitching in a turbine
would be about 20 to 30 per cent closer than in a
two-dimensional cascade . however, the resultant pitching
tends to become very close, except at very large
turning angles, with the result that in some applications
difficulties could arise in the practical design and
manufacture .

several uncertainties remain and the present design must be regarded
as still experimental .

.I 213

.T

the performance of supersonic turbine nozzles .

.A

stratford,b.s. and sansome,g.e.

.B

arc r + m 3273, 1959.

.W

the performance of supersonic turbine nozzles .

an investigation has been conducted
at the national gas turbine establishment into the
performance of turbines having high pressure ratios
per stage . the present report discusses the mode of
operation of supersonic nozzles for such turbines,
and describes a cascade experiment . both theory and
experiment demonstrate that the conditions imposed
upon the supersonic flow immediately downstream
of the nozzles (e.g., by a following row of rotor blades)

exert an overriding influence upon the nozzle outlet flow angle, and hence upon the maximum pressure ratio obtainable across the nozzle--providing that the axial component of velocity is subsonic . this is an important difference from the more familiar flow of subsonic turbine nozzles, where, for example, the downstream gas angle is controlled predominantly by the nozzle blade shape and spacing . a suitable test technique using a closed-jet tunnel is demonstrated .

the particular nozzles tested, of convergent-divergent form, had a straight-sided divergent portion of to axial direction) and a design mach number of 2 . the flow was found to be well behaved as regards shock pattern, losses, and starting over the range of pressure ratios tested--between 9 1 and 19 1 . in particular the efficiency at the design pressure ratio of 16.6 1 was high, the velocity coefficient calculated from traverses of pitot and static tubes being 0.98 .

for the conversion of pitot to total pressure at a mach number of 2.5 a high accuracy is important in the measurement of the static pressure,. nevertheless readings from a conventional four-hole instrument appear to be reliable .

.I 214

.T

on the testing of supersonic compressor cascades .

.A

staniforth,r.

.B

ngte r212, 1957.

.W

on the testing of supersonic compressor cascades .

to facilitate the development of high speed axial-flow compressors, an investigation was made into the possibility of measuring blade performance in a stationary cascade at supersonic speeds . a suitable technique was developed and the losses in a variety of cascades were measured, but these losses were too high for the blading to have any possible application . it was concluded that if a useful compressor is to result, it is essential to test the cascades at mach numbers close to the existing technique was suitable only for zero incidence tests, and thus a new approach is necessary .

some of the fundamentals of this cascade testing at low supersonic speeds are discussed in the light of the current understanding of the mode of operation of supersonic compressors at transonic speeds .

.I 215

.T

the test performance of highly loaded turbine stages

designed for high pressure ratio .

.A

johnston,i.h. and dransfield,d.c.

.B

ngte r235, 1959.

.W

the test performance of highly loaded turbine stages designed for high pressure ratio .

a blade design for a highly loaded two-stage

turbine is described and the test performance of

the turbine is presented .

some of the factors affecting the performance and matching

of turbine blade rows operating at supersonic

gas velocity are discussed and investigated by means of tests

on a three-dimensional nozzle cascade tunnel

and on a variety of single-stage turbine builds .

.I 216

.T

the supersonic axial flow compressor .

.A

kantrowitz,a.

.B

naca r974, 1950.

.W

the supersonic axial flow compressor .

an investigation has been made to explore the possibilities of

axial-flow compressors operating with supersonic velocities into

the blade rows . preliminary calculations showed that very

high pressure ratios across a stage, together with somewhat

increased mass flows, were apparently possible with

compressors which decelerated air through the speed of sound in their

blading . the first phase of this investigation, which has been

reported in naca acr l5d20, was the development of efficient

supersonic diffusers to decelerate air through the speed of sound .

the present report is largely a general discussion of some of the essential aerodynamics of single-stage supersonic axial-flow compressors . in the supersonic flow about isolated bodies, large energy losses usually occur due to wave systems which extend far from the bodies . supersonic flow entering a cascade is considered and, in this case, the possibility of entirely eliminating this extended wave system is demonstrated,. thus, no reason for supersonic compressors to be necessarily inefficient is apparent . the conditions that occur as the flow through the compressor is being started are discussed and a hypothesis as to the type of transonic flow which will be encountered is proposed .

as an approach to the study of supersonic compressors, three possible velocity diagrams are discussed briefly . because of the encouraging results of this study, an experimental single-stage supersonic compressor has been constructed and tested in freon-12 . in this compressor, air decelerates through the speed of sound in the rotor blading and enters the stators at subsonic speeds . a pressure ratio of about 1.8 at an efficiency of about 80 percent has been obtained .

.I 217

.T

flow pattern in a converging-diverging nozzle .

.A

oswatitsch,k. and rothstein,w.

.B

naca tm.1215.

.W

flow pattern in a converging-diverging nozzle .

the present report describes a new method for the prediction of the flow pattern of a gas in the two-dimensional and axially symmetrical case . it is assumed that the expansion of the gas is adiabatic and the flow stationary . the several assumptions necessary on the nozzle shape effect, in general, no essential limitation on the conventional nozzles . the method is applicable throughout the entire speed range,. the velocity of sound itself plays no singular part . the principal weight is placed on the treatment of the flow near the throat of a converging-diverging nozzle . for slender nozzles formulas are derived for the calculation of the velocity components as function of the location .

.I 218

.T

intensity, scale and spectra of turbulence in mixing

region of free subsonic jet .

.A

laurence,j.

.B

naca r1292, 1956.

.W

intensity, scale and spectra of turbulence in mixing

region of free subsonic jet .

the intensity of turbulence, the longitudinal and lateral correlation coefficients, and the spectra of turbulence in a 3.5 inch-diameter free jet were measured with hot-wire anemometers at exit mach numbers from 0.2 to 0.7 and reynolds numbers from

the results of these measurements show the following .. (1)

near the nozzle (distances less than 4 or 5 jet diam downstream

of the nozzle) the intensity of turbulence, expressed as percent

of core velocity, is a maximum at a distance of approximately

increasing mach and or reynolds number . at distances greater

than 8 jet diameters downstream of the nozzle, however, the

maximum intensity moves out and decreases in magnitude until

the turbulence-intensity profiles are quite flat and approaching

similarity . (2) the lateral and longitudinal scales of

turbulence are nearly independent of mach and or reynolds number

and in the mixing zone near the jet vary proportionally with

distance from the jet nozzle . (3) farther downstream of the

jet the longitudinal scale reaches a maximum and then decreases

approximately linearly with distance . (4) near the nozzle the

lateral scale is much smaller than the longitudinal and does not

vary with distance from the centerline, while the longitudinal

scale is a maximum at a distance from the centerline of about

mum moves out from the centerline . (6) a statistical analysis

of the correlograms and spectra yields a /scale/ which, although

different in magnitude from the conventional, varies similarly

to the ordinary scale and is easier to evaluate .

.I 219

.T

on the strength distribution of noise sources along

a jet .

.A

ribner,h.s.

.B

utia r51, 1958.

.W

on the strength distribution of noise sources along

a jet .

the spatial distribution of noise sources along a jet is

investigated by application of lighthill's

theory to regions of 'similar'

profiles . the analysis refers to the

noise power emitted by a 'slice' of

jet (section between two adjacent planes

normal to the axis) as a function

of distance x of the slice from the nozzle .

it is found that this power

is essentially constant with x in the initial

mixing region (x law), then

further downstream (say 8 or 10 diameters

from the nozzle) falls off

extremely fast (x law or faster) in the

fully developed jet . because

of this striking attenuation of strength

with distance, it is concluded that

the mixing region produces the bulk of

the noise and must dominate in

muffler behavior,. conversely, the 'fat'

part of the jet must contribute

much less to the total noise power than is commonly supposed .

powell's experiments on the effects of nozzle velocity

profile on total noise power are interpreted

qualitatively . the behavior of

multiple-nozzle or corrugated mufflers,

both as to overall quieting and

frequency-shifting, is also interpreted

in the light of the results . the

possibility emerges that such mufflers

may be improved without serious

thrust loss by the addition of a sound-attenuating shroud .

.I 220

.T

a general purpose analogue correlator for the analysis of

random noise signals .

.A

g. a. allcock, a.m.i.e.e., a.m.brit.i.r.e.

p. l. tanner, m.sc. (eng), grad. i.e.e.

k. r. mclachlan, a.m.brit. i.r.e.

.B

.W

a general purpose analogue correlator for the analysis of

random noise signals .

a large proportion of the current research programme of the

department of aeronautics and astronautics is concerned with the

study of jet noise and boundary layer pressure fluctuations and

their effect on aircraft structures . early in the work it was

decided that for a complete description of the random processes
involved it would be necessary in the experimental programme to
make correlation measurements in addition to the more standard
spectrum and amplitude distribution measurements . it was also
felt that it would be desirable from the university point of view
to construct a general purpose correlator which could later be
used on other types of work . to this end it was decided to give
the correlator a wider bandwidth than might strictly have been
necessary for the problems on hand . subsequent development work
has amply justified this decision .

.I 221

.T

a theoretical study of annular supersonic nozzles .

.A

lord,w.t.

.B

arc r + m 3227, 1961.

.W

a theoretical study of annular supersonic nozzles .

this paper is concerned with the design
of annular supersonic nozzles to produce uniform
flow in supersonic wind tunnels which are axi-symmetrical
and which have an internal coaxial circular cylinder
throughout . symmetrical two-dimensional and conventional
axi-symmetrical nozzles are special cases of
annular nozzles .

proposals are made for design criteria sufficient to

ensure that the flow inside a nozzle is free from limit
lines and shock waves,. the criteria for (symmetrical)
two-dimensional and (conventional) axi-symmetrical
nozzles are new . the two outstanding procedures for
designing two-dimensional and axi-symmetrical nozzles
are generalised to apply to annular nozzles . one of the
design procedures is mainly analytical and the other is
mainly numerical,. the analytical expressions in both
procedures are made much more complicated by the
presence of the internal cylinder but the numerical process
criteria and the mainly numerical design procedure are
successfully applied to the design of a particular
annular nozzle .

.I 222

.T

the flow over delta wings at low speeds with leading
edge separation .

.A

marsden,d.j., simpson,r.w. and rainbird,w.j.

.B

coa r114, 1957.

.W

the flow over delta wings at low speeds with leading
edge separation .

a low speed investigation of the flow over a 40 apex angle delta
wing with sharp leading edges has been made in order to ascertain
details of the flow in the viscous region near the leading edge of the

suction surface of the wing . a physical picture of the flow was

obtained from the surface flow and a smoke

technique of flow visualization,

combined with detailed measurements of total

head, dynamic pressure, flow

directions and vortex core positions in the flow above the wing .

surface pressure distributions were also measured and integrated

to give normal force coefficients .

the results of this investigation were compared with those of other

experimental investigations and also with various theoretical results .

in particular, the normal force coefficients, vortex core positions and

attachment line positions were compared with the theoretical results of

mangler and smith, reference 19 . it was found that ..

exist on the upper surface of the wing outboard of and below

the main vortices . these secondary vortices are formed as a

result of separation of the boundary layers developing outboard

of the top surface attachment lines .

.I 223

.T

a note on the theory of the stanton tube .

.A

gadd,g.e.

.B

arc r + m 3147, 1958.

.W

a note on the theory of the stanton tube .

existing theories for the stanton tube are critically reviewed, and the paper then outlines a

simple method which predicts the calibration function

at high reynolds numbers to the right order of

magnitude .

.I 224

.T

quasi-cylindrical surfaces with prescribed loadings

in the linearised theory of supersonic flow .

.A

jones,j.g.

.B

rae tn.aero.2769.

.W

quasi-cylindrical surfaces with prescribed loadings

in the linearised theory of supersonic flow .

a formula for the velocity field in terms of a given surface

distribution of vorticity is applied to points

lying on the surface . an equation

giving the shape of a quasi

circular-cylindrical surface in terms of a

prescribed loading is derived . as an

example a half ring wing with prescribed

loading is discussed .

.I 225

.T

elliptic cones alone and with wings at supersonic speeds .

.A

jorgensen,l.h.

.B

naca tn4045,1957

.W

elliptic cones alone and with wings at supersonic speeds .

to help fill the gap in the knowledge

of aerodynamics of shapes

intermediate between bodies of revolution

and flat triangular wings, force

and moment characteristics for elliptic

cones have been experimentally

determined for mach numbers of 1.97 and

sectional axis ratios from 1 through 6

and with lengths and base areas

equal to circular cones of fineness

ratios 3.67 and 5 have been studied

for angles of bank of 0 and 90 .

elliptic and circular cones in

combination with triangular wings of aspect

ratios 1 and 1.5 also have been

considered . the angle-of-attack range

was from 0 to about 16, and the

reynolds number was 8x10, based on

model length . in addition to the

forces and moments at angle of attack,

pressure distributions for elliptic

cones at zero angle of attack have been determined .

the results of this investigation indicate that there are distinct aerodynamic advantages to the use of elliptic cones . with their major cross-sectional axes horizontal, they develop greater lift and have higher lift-drag ratios than circular cones of the same fineness ratio and volume . in combination with triangular wings of low aspect ratio, they also develop higher lift-drag ratios than circular cones with the same wings . for winged elliptic cones, this increase in lift-drag ratio results both from lower zero-lift drag and drag due to lift . visual-flow studies indicate that, because of better streamlining in the crossflow plane, vortex flow is inhibited more for an elliptic cone with major axis in the plane of the wing than for a circular cone with the same wing . as a result, vortex drag resulting from lift is reduced . shifts in center of pressure with changes in angle of attack and mach number are small and about the same as for circular cones .

comparisons of theoretical and

experimental force and moment

characteristics for elliptic cones indicate

that simple linearized (flat plate)

wing theory is generally adequate even

for relatively thick cones .

zero-lift pressure distributions and drag

can be computed using van dyke's

second-order slender-body theory .

for winged circular cones, a

modification of the slender-body theory of

naca rep. 962 results in good agreement

of theory with experiment .

.I 226

.T

aerofoil theory of a flat delta wing at supersonic

speeds .

.A

robinson,a.

.B

rae r.aero.2151, 1946.

.W

aerofoil theory of a flat delta wing at supersonic

speeds .

lift, drag, and pressure distribution of a triangular

flat plate moving at a small incidence at supersonic

speeds are given for arbitrary mach number and aspect ratio .

the values obtained for lift and drag are compared with

the corresponding values obtained by strip theory . the

possibility of further applications of the analysis leading up

to the above results is indicated .

.I 227

.T

a technique for improving the predictions of linearised

theory on the drag of straight edge wings .

.A

randall,d.g.

.B

arc cp394, 1957.

.W

a technique for improving the predictions of linearised

theory on the drag of straight edge wings .

  the curve of drag against mach number for straight-edged wings,

calculated by using the linearised theory of supersonic flow, displays

discontinuities in slope at the various mach numbers for which the edges

are sonic . these features, which are not observed in practice, are due

to the fact that linearised theory predicts an infinite pressure along

a subsonic or sonic edge . it is shown that if the linearised equation

of supersonic flow is used to determine the flow over straight-edged

wings, but the linearised boundary condition is replaced by the full

placed by plausible values . on this basis a simple method is derived

for improving the linearised predictions of the drag of straight-edged

wings which exhibits satisfactory agreement with experimental results .

  while the technique is not directly applicable to ridge lines, an artifice renders them amenable to similar treatment .

.I 228

.T

navier-stokes solutions at large distances from a finite body .

.A

i-dee chang

.B

.W

navier-stokes solutions at large distances from a finite body .

this paper is concerned with a theoretical investigation of the flow

field at large distances from an object moving through a viscous fluid .

the discussion will be restricted to the case of two-dimensional

stationary incompressible flow . the object will be assumed to be of

finite size . the domain of the fluid is infinite and it is assumed

that there are no other boundaries for the fluid except that of the

given object . the reynolds number will be assumed to have a fixed

value., thus we shall not consider the limiting cases of the reynolds

number tending to zero or to infinity .

.I 229

.T

interference between the wings and tail surfaces of

a combination of slender body, cruciform wings and

cruciform tail set at both incidence and yaw .

.A

owen,p.r. and anderson,r.g.

.B

rae r.aero.2471, 1952.

.W

interference between the wings and tail surfaces of

a combination of slender body, cruciform wings and

cruciform tail set at both incidence and yaw .

the interference between the wings and the tail surfaces of a combination of circular body, low aspect ratio cruciform wings and cruciform tail in an inviscid flow is analysed using the slender body theory . the system may be subjected to both incidence and yaw and, in general, the tail fins may be staggered angularly with respect to the main wings .

the method is a development of that used by owen and maskell in r.a.e. report no. aero.2441 to analyse similar effects on a system set at zero yaw .

simple expressions to determine the strengths and positions of the trailing vortices (supposed to be rolled-up) downstream of the main wings are given, and from them the forces on the tail are deduced . when the tail surfaces are triangular and of low aspect ratio an exact solution is obtained from slender body theory .. but for rectangular tail surfaces of moderate or high aspect ratio, it is suggested that the changes in lift and sideforce on the tail caused by the wing vortex field can be estimated approximately from the mean upwash and sidewash angles evaluated over the respective tail spans . formulae for these means angles are presented .

.I 230

.T

interference between the wings and tail plane of a

slender wing-body tailplane combination .

.A

owen,p.r. and maskell,e.c.

.B

rae r.aero.2441, 1951.

.W

interference between the wings and tail plane of a

slender wing-body tailplane combination .

an approximate method of predicting the interference between the wings and the tailplane of a slender wing-body-tailplane combination in an inviscid flow is developed, in order to explain the change in centre of pressure position with incidence which has been found to occur in wind tunnel and flight tests on guided weapons . incidence changes in one plane only, normal to the plane containing the wings and the tail surfaces, have been considered .

the method is based on slender body theory and the assumption that the wing trailing vortices roll-up completely before they reach the tailplane,. it is, therefore, applicable to weapons equipped with low aspect ratio wings far separated from the tail surfaces . when the tail surfaces are triangular and of low aspect ratio, an analytical solution is given for the effect of the wing downwash field on the tail lift . for high aspect ratio, rectangular tail surfaces it is suggested by comparison with experimental data, that the tail lift may be estimated approximately from the value of the mean downwash angle across the tail span .

a summary of the method is given in para.5 which, in conjunction with the introduction, may be read independently of the rest of the report .

.I 231

.T

practical calculation of second-order supersonic flow

past non-lifting bodies of revolution .

.A

van dyke,m.d.

.B

naca tn.2744.

.W

practical calculation of second-order supersonic flow

past non-lifting bodies of revolution .

calculation of second-order supersonic flow past bodies of

revolution at zero angle of attack is described in detail, and reduced to

routine computation . use of an approximate tangency condition is shown

to increase the accuracy for bodies with corners . tables of basic

functions and standard computing forms are presented . the procedure is

summarized so that one can apply it without necessarily understanding

the details of the theory . a sample calculation is given, and several

examples are compared with solutions calculated by the method of

characteristics .

.I 232

.T

accuracy of approximate methods for predicting pressure

on pointed non-lifting bodies of revolution in supersonic

flow .

.A

ehret,d.m.

.B

naca tn.2764.

.W

accuracy of approximate methods for predicting pressure

on pointed non-lifting bodies of revolution in supersonic

flow .

the accuracy and range of applicability of the linearized theory,

second-order theory, tangent-cone method, conical-shock-expansion theory

and newtonian theory for predicting pressure distributions on pointed

bodies of revolution at zero angle of

attack are investigated . pressure

distributions and integrated pressure

drag obtained by these methods are

compared with standard values obtained

by the method of characteristics

and the theory of taylor and maccoll .

three shapes, cone, ogive, and a

modified optimum body, are investigated

over a wide range of fineness

ratios and mach numbers .

it is found that the linearized

theory is accurate only at low values

of the hypersonic similarity parameter

number to body fineness ratio) and that

second-order theory appreciably

extends the range of accurate application .

the second-order theory gives

good results on ogives when the ratio of

the tangent of maximum surface

angle to the tangent of the mach angle

is less than 0.9 . tangent-cone

method cannot be widely applied with

good accuracy . in general, the

conical-shock-expansion theory predicts

pressure and drag within

engineering accuracy when the hypersonic similarity

parameter is greater than 1.2 .

although newtonian theory gives good accuracy,

except for cones, at the

highest values of the hypersonic similarity

parameter investigated, it is

less accurate than the conical-shock-expansion theory .

.I 233

.T

the theoretical wave drag of some bodies of revolution .

.A

fraenkel,l.e.

.B

rae r.aero.2420.

.W

the theoretical wave drag of some bodies of revolution .

this report investigates the wave drag of bodies

of revolution with pointed or open-nose forebodies

and pointed or truncated afterbodies . the 'quasi-cylinder' and 'slender-body' theories are reviewed, a reversibility

theorem is established, and the concept of the interference

effect of a forebody on an afterbody is introduced .

the theories are applied to bodies whose profiles are either

straight or parabolic arcs, formulae and curves being

given for forebody and afterbody drag, and for the interference

drag . the results of the two theories are compared

and are seen to agree well in the region of geometries where both

theories are applicable .

.I 234

.T

a second order shock-expansion method applicable to

bodies of revolution near zero lift .

.A

syvertson,c.a. and denis,d.h.

.B

naca tn.3527.

.W

a second order shock-expansion method applicable to

bodies of revolution near zero lift .

a second-order shock-expansion

method applicable to bodies of

revolution near zero lift is developed .

expressions defining the pressures on

noninclined bodies are derived by the

use of characteristics theory in

combination with properties of the flow

predicted by the generalized
shock-expansion method . this result is
extended to inclined bodies to
obtain expressions for the normal-force
and pitching-moment derivatives
at zero angle of attack . the method is
intended for application under
conditions between the ranges of applicability
of the second-order
potential theory and the generalized shock-expansion
mehtod - namely, when the
ratio of free-stream mach number to nose fineness
ratio is in the
neighborhood of 1 .
for noninclined bodies, the pressure
distributions predicted by the
second-order shock-expansion method are
compared with existing experimental
results and with predictions of other
theories . for inclined bodies, the
normal-force derivatives and locations
of the center of pressure at zero
angle of attack predicted by the method
are compared with experimental
results for mach numbers from 3.00 to 6.28 .
fineness ratio 7, 5, and 3
cones and tangent ogives were tested alone

and with cylindrical afterbodies

up to 10 diameters long . in general, the

predictions of the present method

are found to be in good agreement with the

experimental results . for

non-inclined bodies, pressure distributions

predicted with the method are in

good agreement with existing experimental

results and with distributions

obtained with the method of characteristics .

for inclined bodies, the

normal-force derivatives per radian (for

normal-force coefficients

referenced to body base area) are predicted

within 0.2 and the locations of

the center of pressure are predicted

within 0.2 body diameters . on the

basis of these results, the

second-order shock-expansion method appears

applicable for values of the ratio

of free-stream mach number to nose

fineness ratio from 0.4 to 2 .

.I 235

.T

on the minimisation and numerical evaluation of wave

drag .

.A

eminton,e.

.B

rae r.aero.2564.

.W

on the minimisation and numerical evaluation of wave

drag .

a fourier analysis of the linearised theory expression for the

zero-lift wave drag of a smooth, slender body in terms of its

cross-sectional area distribution is used

to derive the area distribution which

minimises the expression for given

length, volume, nose area, base area

and n intermediate areas . another

minimal deduced from this by relaxing

the restriction on volume is used to

evolve a method for the numerical

evaluation of the original expression .

two practical applications of these

results are discussed . the first

is in the design of wing-body combinations

to have small drag rise at

transonic speeds . the second is in the

calculation of the wave drag of

wing-body combinations at zero lift,.

an example is constructed to

illustrate the method and to give an indication of its accuracy .

.I 236

.T

criteria for thermodynamic equilibrium in gas flow .

.A

rudin,m.

.B

phys.fluids, 1, 1958.

.W

criteria for thermodynamic equilibrium in gas flow .

when gases flow at high velocity, the rates
of internal processes may not be fast enough to
maintain thermodynamic equilibrium . by defining
quasi-equilibrium in flow as the condition in which
the temperature, pressure, density, and velocity
deviate by less than a fixed, small percentage from
what they would be if the flowing gas could actually
be in thermodynamic equilibrium, criteria are
derived for determining whether quasi-equilibrium
is a stable condition in the flow . by use of
excitation of molecular vibration as an example, the
general properties of criteria curves are discussed and
interpreted . a discussion is given of how to use
these results to determine definitely whether a flow
is or is not in thermodynamic equilibrium .
applications to dissociating gases, to mixtures, and to
the phenomenon of /choking/ in a laval nozzle
are given special consideration . for cases when
application of the criteria predict nonequilibrium,

equations are provided in a form useful for

numerical forward integration along streamlines .

.I 237

.T

a compressor routine test code .

.A

n. a. dimmock

.B

communicated by the deputy controller aircraft (research and

development), ministry of aviation

.W

a compressor routine test code .

the routine testing of aircraft-type compressors.dash in the main,

axial-flow, multi-stage compressors.dash requires a compromise between

research accuracy and the practical considerations . this test code

is the outcome of a survey of compressor testing techniques and

instrumentation, initiated and subsequently discussed and endorsed

by the aerodynamics sub-committee of the gas turbine collaboration

committee .

the code aims at defining methods of measurement and weighting whereby

compressor performance can be obtained sufficiently accurately

for a realistic and direct comparison to be made between one compressor

and another . the measurement of a quantity at a point in the fluid

flow, and the averaging and weighting of such measurements have been

treated separately as far as is possible .

the recommendations are given in the main text, whilst additional

discussion on these is put into the appendices .

.I 238

.T

on a determination of the pitot-static tube factor

at low reynolds numbers, with special reference to

the measurement of low air speeds .

.A

ower,e. and johansen,f.c.

.B

arc r + m 1437, 1931.

.W

on a determination of the pitot-static tube factor

at low reynolds numbers, with special reference to

the measurement of low air speeds .

reasons for enquiry--to provide a standard instrument for

the calibration of low speed anemometers .

.I 239

.T

design and calibration at low speeds of a static tube

and a pitot-static tube with semi-ellipsoidal nose

shapes .

.A

kettle,d.j.

.B

rae tn.aero.2247, 1953.

.W

design and calibration at low speeds of a static tube

and a pitot-static tube with semi-ellipsoidal nose shapes .

a new static tube and a new pitot-static tube have been designed and calibrated in the no.1 and the no.2 11 ft x 8 ft wind tunnels of the r.a.e., using a long static tube, the error of which is believed to be very small, as a standard for comparison .

the results show that the static pressure measured by these tubes is in error due to the supporting strut and to the nose shape of the tube by an amount which may be calculated for positions of the static slot, or holes, greater than 10 tube diameters ahead of the strut . the readings show no measurable scale effect in the speed range 100-230 ft sec . the static tube is insensitive to yaw in the range 1 with a square-edged slot and is even less sensitive to yaw when the slot edges are rounded . the turbulence of the tunnel has an effect on the static pressure reading .

.I 240

.T

a theoretical analysis of heat transfer in regions of separated flow .

.A

dean r. chapman

.B

naca technote 3792

.W

a theoretical analysis of heat transfer in regions of separated flow .

the flow field analyzed consists of a thin, constant pressure viscous mixing layer separated from a solid surface by an enclosed region of low-velocity air (/dead air/) . the law of conservation of energy is employed to relate calculated conditions within the separated mixing layer to the rate of heat transfer at the solid

surface . this physical speed is app ied to alminar separations in

compressible flow for various prandtl numbers, including consideration

of the case where air is injected into the separated region .

.A

application to turbulent separations is made for a prandtl number of

.B

unity in low-speed flow without injection .

all calculations are for the case of zero boundary-layer thickness at

the position of separation .

for alminar separations the differential equations for viscous flow

at arbitrary mach number are solved for the enthalpy and velocity

profiles within the thin layer where mixing with dead air takes place .

results are presented in tabular form for prandtl numbers between 0.1

and 10 . the rate of heat transfer to a separated laminar region in air

laminar boundary layer having the same constant pressure . injection

of gas into the separated region is calculated to have a powerful effect

in reducing the rate of heat transfn to the wall . it is calculated

that a moderate quantity of gas injection reduces to zero the heat

transfer in a laminar separated flow .

.I 241

.T

laminar mixing of a non-uniform stream with a fluid

at rest .

.A

nash,j.f.

.B

arc 22245, 1960.

.W

laminar mixing of a non-uniform stream with a fluid

at rest .

a theoretical analysis is made of the constant pressure

laminar mixing process between a stream having an initial boundary layer

velocity profile, and a fluid at rest .

the present theory follows the methods of w. tollmien and

s. i. pai with certain modifications .

the results apply to incompressible

flow, but can be extended to the compressible case without difficulty .

.I 242

.T

an approximate theory of base pressure in two dimensional

flow at supersonic speeds .

.A

kirk,f.n.

.B

rae tn.aero.2377, 1954.

.W

an approximate theory of base pressure in two dimensional

flow at supersonic speeds .

an approximate theory of the base pressure in two-dimensional flow

at supersonic speeds is presented using asimplified representation of

the flow and some of the findings of tollmien's work on turbulent mixing

in incompressible flow . good qualitative predictions of the effects of

a boundary layer, of bleed air and of boat-tailing are obtained .

.I 243

.T

investigation with an interferometer of the turbulent

mixing of a free supersonic jet .

.A

gooderum,p.b., wood,g.p. and brevoort,m.

.B

naca r963, 1950.

.W

investigation with an interferometer of the turbulent

mixing of a free supersonic jet .

the free turbulent mixing of a supersonic jet of mach number

of which a description is given, was used for the investigation .

density and velocity distributions through the mixing zone

have been obtained . it was found that there was similarity

in distribution at the cross sections investigated and that, in

the subsonic portion of the mixing zone, the velocity

distribution fitted the theoretical distribution for incompressible flow .

it was found that the rates of spread of the mixing zone both

into the jet and into the ambient air were less than those of

subsonic jets .

.I 244

.T

an improved smoke generator for use in the visualisation

of airflow, particularly boundary layer flow at high

reynolds numbers .

.A

preston,j.h. and sweeting,n.e.

.B

arc r + m 2023, 1943.

.W

an improved smoke generator for use in the visualisation

of airflow, particularly boundary layer flow at high

reynolds numbers .

and rapid method by which boundary

layer flow was rendered visible has been

previously described in the journal of the royal

aeronautical society . it gave promise of

being useful at the highest tunnel speeds provided

a denser smoke could be obtained, which at

the same time was free from the troublesome deposits

associated with the wood smoke .

of the aerodynamics division attempts were made

by the fuel research station to improve the

density of the wood smoke and to reduce the

deposits . these they showed were conflicting

requirements, and whilst some improvement

was effected, it was not sufficient for observation

in the new tunnels at high speeds .

the staff of the director-general of scientific

research and development, ministry of supply,

was then approached and it was decided to

develop an oil smoke generator from a simple

generator of this type which was demonstrated

to us . this has been done successfully .
the final apparatus in contrast to the wood smoke generator is light and compact . it takes only a few minutes to start and can be run as long as desired . improvement on the wood smoke both as regards density and freedom from deposits, which cause premature transition . the density and quality of the smoke are now under control . smokes ranging from a light smoke of bluish white colour to a heavy smoke dense white in appearance can be obtained . the oil smoke retains the advantages of the wood smoke in that it is non-corrosive and non-irritant, and the smell can be tolerated even when it is present in a considerable concentration . a certain amount of condensation is inevitable with oil smokes, but with suitable precautions troubles arising from this can be avoided . a dry solid smoke made by melting a hard wax was successfully generated with the same apparatus . unfortunately because of its flocculent nature this smoke gave rise to solid deposits when passed through bore tubing, leading eventually to complete blockage . this seems to be a feature of solid smokes .

the apparatus has been used to determine transition and laminar separation points on model wings in a number of the national physical laboratory tunnels . smoke filaments have been

maintained in the laminar state up to wind

speeds of 180 ft. sec. in the new tunnels .

there is much to be said for making a standard

practice of visualising boundary layer flow on

models, particularly as the technique is simple and

rapid . it would greatly assist the

interpretation of force measurements and the more detailed

explorations of the boundary layer by total

head tubes and hot wires .

the use of oil smoke is not limited to

boundary layer flow visualisation . the apparatus

described in this report would seem to be

particularly suited for educational work in small

demonstration tunnels .

.I 245

.T

the ground effect on the jet flap in two dimensions .

.A

huggett,d.j.

.B

arc 19,713, 1957.

.W

the ground effect on the jet flap in two dimensions .

this paper presents the results of the first part of an

experimental investigation of the ground effect on simple jet flap

aerofoils . in this part of the work an aerofoil having a 58.1 deg jet flap was tested under two-dimensional conditions .

the pressure lift on the aerofoil was measured, with the ground at fixed positions, for varying jet momentum coefficients . it was found that the effect of the ground on the pressure lift was very small up to a certain critical jet coefficient . on increasing the jet coefficient beyond the critical value a marked loss of pressure lift was observed . this critical value referred to is approximately the same as the jet coefficient at which the jet first hits the ground .

some significant, though highly tentative comments, are made regarding the practical application of this work to the take-off characteristics of a jet flapped aircraft .

.I 246

.T

the design of minimum drag tip fins . with an appendix

- on the conformal transformation of a wing with a

fin .

.A

falkner,v.m.

.B

arc r + m 2279, 1945.

.W

the design of minimum drag tip fins . with an appendix

- on the conformal transformation of a wing with a

fin .

the report describes an investigation into the design of minimum drag tip fins by lifting line theory . the work is based on an exact solution of the conformal transformation which is applicable to this problem following the

method of trefitz . three types of solution are treated,

corresponding to symmetrical upper and lower fins, single

upper or lower fins, and unequal upper and lower fins .

a representative range of solutions for circulation distribution

along wing and fins has been calculated for each of the

three cases by the use of elliptic and theta functions .

a detailed account is given, with examples, of the

procedure for calculating the plan of wing and fins, the lift and

induced drag, and the setting of the fins .

.I 247

.T

the calculation of the pressure distribution on thick

wings of small aspect ratio at zero lift in subsonic

flow .

.A

weber,j.

.B

arc r + m 2993.

.W

the calculation of the pressure distribution on thick

wings of small aspect ratio at zero lift in subsonic

flow .

the method of expressing the velocity increment

over aerofoils directly in terms of the section ordinates

wings of finite aspect ratio . the wings considered are

untapered in plan-form but may be tapered in thickness .

the section can be of any given shape so that in this sense

the analysis is more general than that of refs. 3 to 6 which

deal with wings of biconvex section .

the coefficients required in the calculation are tabulated

for the centre-section of straight and swept-back wings

of aspect ratios 0.5,. 1,. 2,. and 4, the wing of infinite

aspect ratio having been

treated in ref. 1 . the remaining calculations can be made

very quickly .

since wings of very small aspect ratio can be treated also by

the method of slender-body theory, the relations between

linear theory, slender-body theory, and linearised slender-body

theory are discussed . for the special case of ellipsoids,

the results obtained from the various methods are compared

with the exact solution .

.I 248

.T

the application of lighthill formula for numerical

calculation of pressure distributions on bodies of

revolution at supersonic speed and zero angle of attack .

.A

ohman,l.

.B

saab tn.45, 1960.

.W

the application of lighthill formula for numerical

calculation of pressure distributions on bodies of revolution at supersonic speed and zero angle of attack .

an integral expression, given by lighthill and based

on linearized theory, for the external

supersonic flow over the surface of slender pointed

or ducted bodies of revolution at zero

angle of attack is shown to give a good approximation

of the exact flow for a much wider mach

number and thickness range than could be expected from

linearized theory . a numerical method,

based on this expression, is developed and applied for

digital computing . some results from

applying the digital computing procedure for determining

the pressure distribution and wave

drag for various bodies of revolution are given .

.I 249

.T

formulae for the computation of the functions employed

for calculating the velocity distribution about a given

aerofoil .

.A

watson,e.j.

.B

arc r + m 2176.

.W

formulae for the computation of the functions employed

for calculating the velocity distribution about a given

aerofoil .

in order to determine the velocity distribution
about an arbitrary aerofoil, it is necessary to evaluate
the functions and (in the notation of aerofoil theory)
when is given numerically . if the values of are specified
at 2n points equally spaced about the circle into which the
aerofoil is transformed, the formulae obtained here may be
used to calculate these functions at the same points .
formulae are also given for calculating the integrals of or,
since these have application to the design of aerofoils by
thwaites's numerical method .

the simplicity of the formulae for and enables
the effect on the velocity distribution of a local change of shape
readily to be determined by making n large . this is
discussed in 3 .

the formulae are collected in the appendix, and a table
of the coefficients for the case n = 20 is given .

.I 250

.T

pressure distributions at zero lift for delta wings
with rhombic cross sections .

.A

eminton,e.

.B

arc cp.525, 1960.

.W

pressure distributions at zero lift for delta wings
with rhombic cross sections .

the linearised theory of thin wings is used to calculate pressure

distributions over delta wings with rhombic cross sections . a deuce

programme has been written for the calculation and some of the results

are compared with those of slender thin wing theory .

.I 251

.T

a collection of longitudinal stability derivatives

of wings at supersonic speeds .

.A

naysmith,a.

.B

rae tn.aero.2423, 1956.

.W

a collection of longitudinal stability derivatives

of wings at supersonic speeds .

a collection has been made of

theoretical data, for wings alone, on

those stability derivatives that govern

the short-period oscillation of

aircraft travelling at supersonic speeds .

all the derivatives available

have been obtained by means of the linear

theory, and so the information

given is subject to the usual limitations .

the information has been

presented in what is hoped is the most

convenient form to show its extent,

and to expose the parts of the field

where experimental investigation is

most needed .

.I 252

.T

an investigation of interference effects on similar models of

different size in various transonic tunnels in the u.k. .

.A

f. o/hara and l. c. squire, r.a.e.

.B

and a. b. haines, a.r.a.

.W

an investigation of interference effects on similar models of

different size in various transonic tunnels in the u.k. .

details are given of a programme of tests being made on similar

swept-wing models in transonic tunnels of different types . force measurement

results at subsonic speeds in the r.a.e. 3 ft. by 3 ft. slotted tunnel

show only small interference effects for models of moderate blockage

at low incidence., at higher incidences, the interference effect on

lift becomes appreciably greater than estimated by theory, and

significant pitching moment differences occur, apparently due to wall

interference on the wing flow field . comparable but smaller effects

are evident in the results from the a.r.a. 9 ft. by 8 ft. perforated

tunnel . at speeds just above m = 1, the force fluctuates as speed

is increased, because of wave reflection interference . the magnitude

of the fluctuations diminishes as speed is further increased and this

reduction is more marked in the perforated tunnel . pressure measurements along the top of the body at zero incidence show delay in shock movements at high subsonic speeds indicating a blockage effect on speed., the effect is larger in the perforated tunnel though smaller than predicted by theory . above m = 1, both expansion and shock waves are strongly reflected in the slotted tunnel but considerable alleviation, particularly of shock waves, is achieved in the perforated tunnel, for which an analysis of the effects is given, showing for example, the effect of the open-area distribution of the walls .

.I 253

.T

on the ground level disturbance from large aircraft flying at supersonic speeds .

.A

lilley,g.m. and spillman,j.j.

.B

coa n103.

.W

on the ground level disturbance from large aircraft flying at supersonic speeds .

the whitham-walkden theory for the estimation of the strength of shock waves at ground level from aircraft flying at supersonic speeds is applied to the case of a typical projected supersonic civil transport aeroplane .

if a figure of 2 lb sq.ft. (including a factor of 2 for ground reflection) is taken as an upper limit for the acceptable strength of the bow wave from such an aircraft it is shown that restrictions on the climb and flight plan will be involved . the advantage of the

employment of larger engines with or without afterburning is discussed,

with reference also to the penalties involved owing to the increase in

weight of the aircraft and its direct operating costs .

finally it is suggested that an aircraft of given volume could be

designed, by suitable choice of thickness and lift distribution, to

minimise the strength of the shock waves in the far field .

.I 254

.T

boundary layers with suction and injection . a review

of published work on skin friction .

.A

craven,a.h.

.B

coa r136.

.W

boundary layers with suction and injection . a review

of published work on skin friction .

available data on the effects of suction and injection on skin

friction are summarised and compared .

it is shown that injection into a turbulent boundary layer can

produce a skin friction coefficient lower than the laminar value at the

same reynolds number on an impermeable plate .

.I 255

.T

an approximate solution of the turbulent boundary layer

equations in incompressible and compressible .

.A

lilley,g.m.

.B

coa r134, 1960.

.W

an approximate solution of the turbulent boundary layer

equations in incompressible and compressible .

if over the 'outer region' of the boundary layer, where the mean velocity varies but little from its value outside the shear layer, a virtual eddy viscosity is defined, which is constant over the outer region but varies in the direction of the mainstream, a solution of the turbulent boundary layer equations can be found which satisfies the appropriate boundary conditions . the solution leads to a compatibility condition for the virtual eddy viscosity in terms of the wall shear stress, the boundary layer momentum thickness and the mainstream velocity, at least for the case of a constant external velocity . this compatibility condition, which can be expressed as

for moderate to high reynolds numbers, where is the shear velocity, is the boundary layer thickness and is the virtual eddy (kinematic) viscosity, is just the condition townsend (1956) found for the equilibrium of the large eddies . the numerical value of the constant derived by townsend agrees with ours for reynolds numbers (based on x) of about . with this relation for an equation, analoguous to the momentum integral equation solution, can be found for as a function of local freestream velocity, with one disposable parameter .

.I 256

.T

an experimental study of the glancing interaction between

a shock wave and a turbulent boundary layer .

.A

stanbrook,a.

.B

arc cp.555, 1960.

.W

an experimental study of the glancing interaction between

a shock wave and a turbulent boundary layer .

an experimental study has been made at mach numbers from 1.6 to 2.0 of

the interaction between the turbulent

boundary layer on a side wall of a wind

tunnel and the shock wave produced by

a plate mounted on the wall . under

these conditions the shock wave boundary

layer interaction was three

dimensional at least over the region

investigated (up to 10 boundary layer

thicknesses from the plate) . it was

found that the boundary layer was

separated by a shock wave of strength

type occur on the sides of fuselages at

the wing fuselage junction and may

therefore be important with regard to the

design of waisted shapes .

.I 257

.T

on turbulen flow between parallel plates .

.A

pai,s.i.

.B

j.app.mech. 20, 1953, 109.

.W

on turbulen flow between parallel plates .

the reynolds equations of motion of turbulent flow of incompressible fluid have been studied for turbulent flow between parallel plates . the number of these equations is finally reduced to two . one of these consists of mean velocity and correlation between transverse and longitudinal turbulent-velocity fluctuations only . the other consists of the mean pressure and transverse turbulent-velocity intensity . some conclusions about the mean pressure distribution and turbulent fluctuations are drawn . these equations are applied to two special cases .. one is poiseuille flow in which both plates are at rest and the other is couette flow in which one plate is at rest and the other is moving with constant velocity . the mean velocity distribution and the correlation can be expressed in a form of polynomial of the co-ordinate in the direction perpendicular to the plates, with the ratio of shearing stress on the plate to that of the corresponding laminar flow of the same maximum velocity as a parameter . these expressions hold true all the way across the plates, i.e., both the turbulent region and viscous layer including the laminar sublayer . these expressions for

poiseuille flow have been checked with experimental data

of laufer fairly well . it also shows that the logarithmic

mean velocity distribution is not a rigorous solution of

reynolds equations .

.I 258

.T

the effect of turbulence on slider bearing lubrication .

.A

chou,y.t. and saibei,e.

.B

j.app.mech. 25, 1959, 122.

.W

the effect of turbulence on slider bearing lubrication .

based on prandtl's mixing-length mechanism, the pressure

equation for turbulent flow in

slider-bearing lubrication is derived . an analytical

solution is given and compared with

the one for laminar flow . it is found that the turbulent

effect increases the pressure and

consequently, the load-carrying capacity . however, the

power loss also increases .

.I 259

.T

second order theory for unsteady supersonic flow

past slender pointed bodies of revolution .

.A

revell, j. d.

.B

j. ae. scs.1960, 730.

.W

second order theory for unsteady supersonic flow

past slender pointed bodies of revolution .

the thermodynamic properties (z = pv rt, e rt, h rt, s r,

and pressure) are given for equilibrium mixtures of dissociated and

ionized molecules and atoms of the elements nitrogen and oxygen

having the low temperature composition of .78847 n and .21153 o .

the tabulated properties of this mixture (a close approximation to

the properties of air) are given at close intervals from 2000 to

and 10 times the normal density . the results are based on

chemical equilibria between the species o, o, n, n, no, no, no,

no, o, o, o, o, n, n, n and electrons . the method of

presentation permits later corrections for the effect of argon and co

and the contribution of intermolecular forces . the calculations are

based on 9.758 e.v. as the dissociation energy of molecular nitrogen

and 1.45 e.v. as the electron affinity of atomic oxygen .

.I 260

.T

a critical review of skin friction and heat transfer

solutions of the laminar boundary layer of a flat plate .

.A

rubesin,m.w. and johnson,h.a.

.B

asme trans. 1949.

.W

a critical review of skin friction and heat transfer

solutions of the laminar boundary layer of a flat plate .

a review is made of existing literature concerned with

the analytical investigation of the velocity and

temperature distributions in the boundary layers of a heated (or

cooled) flat plate . the plate is postulated infinitely thin

and is parallel to a uniform fluid stream . the more

recent solutions include the combined effects of frictional

dissipation and variable fluid properties . only the results

pertaining to the transfer phenomena occurring at the

plate surface are included, i.e., skin drag and over-all

heat transfer,. the individual temperature and velocity

distributions leading to these results are omitted .

.I 261

.T

experiments on axi-symmetric boundary layers along

a long cylinder in incompressible flow .

.A

yashura,m.

.B

trans. japan soc.ae.sc. 2, 1959.

.W

experiments on axi-symmetric boundary layers along

a long cylinder in incompressible flow .

experiments on axi-symmetric boundary

layers along a long cylinder were made

especially to investigate the effect of transverse

curvature on the velocity profile . laminar

velocity profiles were measured and compared

with theoretical ones with good accuracy .

a representative profile was plotted to see the

effect of transverse curvature, which showed

small, but obvious effect accompanied by

increasing skin friction .

the transition of the flow from laminar to

turbulent was observed, and its reynolds

number was estimated to occur at 1.2 1.8x10

in the present experiment . the turbulent

profile was also measured and plotted by using

the coordinates to express the wall law

deduced by richmond, from which it was

estimated that, as the ratio of the momentum

thickness to body radius increases, the profile

near the outer layer tends to bend down

relative to the line of logarithmic wall law .

.I 262

.T

the formation of a blast wave by a very intense explosion .

.A

taylor,g.i.

.B

proc. roy. soc. a, 201, 1950, 159.

.W

the formation of a blast wave by a very intense explosion .

this paper was written early in 1941

and circulated to the civil defence research

committee of the ministry of home security

in june of that year . the present writer

had been told that it might be possible

to produce a bomb in which a very large

amount of energy would be released by

nuclear fission--the name atomic bomb had

not then been used--and the work here

described represents his first attempt to form

an idea of what mechanical effects might

be expected if such an explosion could occur .

in the then common explosive bomb mechanical

effects were produced by the sudden

generation of a large amount of gas at a high

temperature in a confined space . the

practical question which required an answer

was .. would similar effects be produced

if energy could be released in a highly

concentrated form unaccompanied by the

generation of gas .qm this paper has now

been declassified, and though it has been

superseded by more complete calculations,

it seems appropriate to publish it as it was

first written, without alteration, except for

the omission of a few lines, the addition of

this summary, and a comparison with some
more recent experimental work, so that
the writings of later workers in this field may be appreciated .
  an ideal problem is here discussed . a finite
amount of energy is suddenly released
in an infinitely concentrated form . the motion
and pressure of the surrounding air is
calculated . it is found that a spherical shock
wave is propagated outwards whose
radius r is related to the time t since the explosion
started by the equation
where is the atmospheric density, e is
the energy released and s a calculated
function of, the ratio of the specific heats of air .
  the effect of the explosion is to force most
of the air within the shock front into a
thin shell just inside that front . as the front
expands, the maximum pressure
decreases till, at about 10 atm., the analysis ceases
to be accurate . at 20 atm. 45 of
the energy has been degraded into heat which is
not available for doing work and used
up in expanding against atmospheric pressure .
this leads to the prediction that an
atomic bomb would be only half as efficient, as
a blast-producer, as a high explosive
releasing the same amount of energy .

in the ideal problem the maximum pressure is

proportional to r, and comparison

with the measured pressures near high explosives,

in the range of radii where the two

might be expected to be comparable, shows that

these conclusions are borne out by

experiment .

.I 263

.T

cylindrical shock waves produced by instantaneous energy

release .

.A

lin,s.c.

.B

j.app.phys. 25, 1954, 54.

.W

cylindrical shock waves produced by instantaneous energy

release .

taylor's analysis of the intense spherical explosion

has been extended to the cylindrical case . it is found

that the radius r of a strong cylindrical shock wave

produced by a sudden release of energy e per unit length

grows with time t according to the equation

where is the atmospheric density and

is a calculated function of the specific

heat ratio . for is found to be approximately

unity . for this case, the pressure behind the shock wave decays with radius r according to the relation .

applying the results of this analysis

to the case of hypersonic flight, it can be shown that

the shock envelope behind a meteor or a high-speed

missile is approximately a paraboloid given by

where d and v denote the total

drag and the velocity of the missile, respectively,

and x is the distance behind the missile .

.I 264

.T

asymptotic solution of the two dimensional oscillating

aerofoil problem for high subsonic mach numbers .

.A

eckhaus,w.

.B

proc. 9th int. con. app.mech. 1956.

.W

asymptotic solution of the two dimensional oscillating

aerofoil problem for high subsonic mach numbers .

a new method has been given, for obtaining asymptotic solutions of a

boundary value problem for the wave equation . the method is simpler

than the method previously given by burger, and leads to a result

identical with burger's result .

.I 265

.T

some instabilities arising from the interaction between

shock waves and boundary layer .

.A

lambourne,n.c.

.B

npl aero.348, 1958.

.W

some instabilities arising from the interaction between

shock waves and boundary layer .

a brief review is made of the available information concerning

the flow fluctuations and instabilities arising from shock-induced

separation in the flow over aerofoils and wings . the influence this

phenomenon has on the oscillatory behaviour of aerofoils and control

surfaces is also briefly discussed .

a more detailed consideration is devoted to a recent investigation

at the n.p.l. into the part played by shock-induced separation in the

instability of a control surface .

.I 266

.T

exact solution of the neumann problem . calculation

for non- circulatory plane and axially symmetric flows

about or within arbitrary boundaries .

.A

smith,a.n.c. and pierce,j.

.B

3rd nat. con. app. mech. 1958.

.W

exact solution of the neumann problem . calculation

for non- circulatory plane and axially symmetric flows

about or within arbitrary boundaries .
an exact general method of solving
the neumann or second boundary-value
problem has been developed and has been
applied to the calculation of
low-speed flows about or within bodies of
almost any shape, provided the flow is
either plane or has axial symmetry .
solid-body, inlet, and purely internal
flow problems can be solved . the
method is capable of dealing with several
bodies at once in the presence of
one another, and consequently interference
problems can be treated with ease .
boundaries need not be solid, that is,
flows involving area suction can be
calculated . velocities can be computed
not only for points on the surface of
the body but for the entire flow field .
a surface source distribution is
used as a basis for solution . this
leads to a fredholm integral equation
of the second kind, which is solved as
a set of linear algebraic equations,
usually by a modified seidel method .
at the present time the solution is
programed on the ibm 704 edpm to solve

the flow about any body that has the

previously mentioned characteristics

and whose profile can be defined

satisfactorily by no more than 300

coordinate points . a number of solutions

are presented, to show both the scope

of the method and its accuracy .

computations require from three minutes

to two hours, depending upon the

shape of the body and the number of points

used to define it .

.I 267

.T

steady and transient free convection of an electrically

conducting fluid from a vertical plate in the presence

of a magnetic field .

.A

gupta,s.

.B

app. sc. res. 9, 1959.

.W

steady and transient free convection of an electrically

conducting fluid from a vertical plate in the presence

of a magnetic field .

an analysis is made for the laminar

free convection and heat transfer of

a viscous electrically conducting fluid

from a hot vertical plate in the case

when the induced field is negligible

compared to the imposed magnetic field .

it is found that similar solutions for

velocity and temperature exist when

the imposed magnetic field (acting

perpendicular to the plate) varies inversely

as the fourth root of the distance from

the lowest end of the plate . explicit

expressions for velocity, temperature,

boundary layer thickness and nusselt

number are obtained and the effect

of a magnetic field on them is studied .

it is found that the effect of the

magnetic field is to decrease the rate of

heat transfer from the wall . in the

second part, the method of characteristics

is employed to obtain solutions of

the time-dependent hydromagnetic free

convection equations (hyperbolic) of

momentum and energy put into integral

form . the results yield the time required

for the steady flow to be established,

and the effect of the magnetic field on this time is studied .

.I 268

.T

several magnetohydrodynamic free-convection solutions .

.A

cramer,k.r.

.B

5th nat. heat transfer con. 1962.

.W

several magnetohydrodynamic free-convection solutions .

the influence of transverse magnetic fields on

the laminar free-convection flow of liquid

metals over a vertical flat plate and between

vertical parallel plates is examined for

specific wall temperature variations and

prandtl numbers . the extent of influence on

the flow and temperature fields is determined

by the magnitude of a nondimensional

influence parameter which is the ratio of the

magnetic force to the buoyant force . in

general, increasing the magnetic field strength

decreases the magnitude of the velocity, wall

shear, and surfaces heat transfer and

increases the temperature throughout the fluid .

analytical results demonstrate that magnetic

fields of practical strengths exert

considerable influence on liquid metal free-convection flow fields .

.I 269

.T

on a laminar free-convection flow and heat transfer

of electrically conducting fluid on a vertical flat plate in the presence of a transverse magnetic field .

.A

mori,y.

.B

trans. japan soc. ae. sc. 2, 1959.

.W

on a laminar free-convection flow and heat transfer

of electrically conducting fluid on a vertical flat

plate in the presence of a transverse magnetic field .

the free-convection flow and heat transfer of an

electrically conducting fluid on a

vertical plate in the presence of a transverse magnetic

field is analysed for a magnetic field

fixed to the electrically non-conducting wall . the

boundary layer equations for

self-preserving flows are integrated numerically for the

prandtl number of unity, and the effect

of the transverse magnetic field on the velocity

profile, temperature profile and rate of

heat transfer is discussed . it is concluded that

the heat transfer rate is reduced as the

magnetic field intensity is increased .

.I 270

.T

on combined free and forced convection laminar magnetohydrodynamic

flow and heat transfer in channels with transverse

magnetic field .

.A

mori,y.

.B

int. devel. in heat transfer,1961.

.W

on combined free and forced convection laminar magnetohydrodynamic flow and heat transfer in channels with transverse magnetic field .

combined free and forced convective heat transfer in vertical channels has been studied by many researchers . due to the need for engineering design information there have been many papers concerning cases of fully developed flow with varying wall temperature . forced flows in a channel of electrically conducting fluid with a transverse magnetic field have been studied and the large effects of a magnetic field on the flow pattern have been established .

flows of combined free and forced convection in electrically conducting fluids in vertical channels with a transverse magnetic field are expected to attract attention in future engineering applications, for example, in a magneto-hydrodynamic generator or in plasma studies . however, except for a report by gershuni and zhukhovitskii (1) concerning a particular case, no general study has been published .

this paper is a general treatment of fully developed, free and forced convective, laminar, magneto-hydrodynamic flow in a vertical channel with a

transverse magnetic field . it includes combined free and

forced convective flows in channels without a

magnetic field reported by ostrach (2), tao (3), etc. as

special cases . hartmann flow (4) is included in the

other limit .

.I 271

.T

an experimental test of compressibility transformation

for turbulent boundary layer .

.A

squire,w.

.B

j. ae. sc. 29, 1962.

.W

an experimental test of compressibility transformation

for turbulent boundary layer .

discussion of various turbulent-boundary-layer theories, in

the light of experimental measurements by matting and co-workers .

the application of (1) the mager insulated-wall transformation, and

and illustrated graphically .

.I 272

.T

oscillatory aerodynamic coefficients for a unified supersonic

hypersonic strip theory .

.A

rodden, w. +. and revell, j.d.

.B

j. ae. scs. 1960, 451.

.W

oscillatory aerodynamic coefficients for a unified supersonic hypersonic strip theory .

the shock tube is shown to be a feasible research tool for conducting boundary-layer transition experiments . the use of the shock tube permits the study of transition with highly cooled boundary layers, as may be encountered on hypersonic vehicles .

boundary-layer transition investigations have been made on optically polished pyrex hemisphere-cylinder and ellipse-cylinder models with stagnation-to-wall enthalpy ratios between 4.5 and roughness estimated to be less than 1 microinch (rms) . transition was detected by measurements of the heat-transfer rates on the model surface .

the shock tube experiments indicated that a characteristic feature of transition of a highly cooled boundary layer on a hemisphere was the simultaneous occurrence of transition over the entire supersonic portion of the hemisphere . this implies that transition first occurred in the sonic region . the transition reynolds number (based on local fluid properties at the outer edge of the boundary layer and the momentum thickness) in the sonic region increased from about 225 to 325 as the stagnation-to-wall enthalpy ratio increased from about 9.5 to 29.5 . transition occurred along the cylindrical portion of the hemisphere-cylinder model at a nearly constant momentum thickness reynolds number, increasing from about 400 to 625 as the stagnation-to-wall enthalpy ratio increased from about 9.5 to 29.5 .

the highly cooled boundary layers obtained on the cylindrical portion of the shock tube hemisphere-cylinder model provided an extension of nasa transition results obtained on a cooled hemisphere-cone-cylinder model in a wind tunnel . the transition reynolds numbers obtained from these shock tube data were of the same order of magnitude as the minimum transition reynolds numbers obtained in the wind-tunnel experiments . the results indicate that, for practical purposes, boundary-layer cooling is not a critical transition parameter for blunt bodies with a highly cooled boundary layer resulting from a stagnation-to-wall enthalpy ratio of about 3 to 30 . that is, the transition reynolds number did not vary significantly with boundary-layer cooling in this cooling range, but transition always occurred at a low reynolds number (between about 350,000 and 750,000 based on local external properties and a distance along the body surface from the stagnation point) .

the boundary-layer history (body shape history) appeared to be an important parameter affecting the magnitude of the reynolds number for transition and the amount of increase in the transition reynolds number with increased boundary-layer cooling . that is, transition occurred at a lower reynolds number on the ellipse-cylinder configuration than on the hemisphere-cylinder . also, the increase in transition reynolds number with an increase in boundary-layer cooling was even less significant for the ellipse-cylinder than the hemisphere-cylinder .

.I 273

.T

flow past slender blunt bodies - a review and extension .

.A

capiaux,r. and karchmar,l.

.B

paper 61-201-1904, nat. ias-ars joint meeting, 1961.

.W

flow past slender blunt bodies - a review and extension .

a numerical solution of the inviscid flow field about slender blunt

bodies of revolution has been developed through a combination of two

methods .. the van dyke solution in the subsonic flow

region at the nose, and the method of

characteristics in the supersonic region .

the results are compared with

second-order blast wave theory and with experimental

data,. and the respective merits and

deficiencies of the two theoretical methods

are pointed out . the results of the

numerical solution are further used in a

discussion of the entropy layer, to

propose a possible criterion of entropy layer thickness .

.I 274

.T

analysis of quartz and teflon shields for a particular

re-entry mission .

.A

adams,e.w.

.B

heat transfer and fluid mech. inst. 1961, 222.

.W

analysis of quartz and teflon shields for a particular

re-entry mission .

the transient performance of

ablation type heat protection shields is

treated herein for the surface of

a vehicle returning from outer space to the

earth . the vehicle weighs 8640 kg,

has a ballistic factor of 500 lb ft,

re-enters with a speed of 11 km sec at

ratio of 0.5, and is subjected to a

maximum deceleration of 7.7 times the

gravity constant .

by use of well known equations

for the heat transfer and the mass

transfer at a heated surface, a numerical

calculation method is derived which, for

the investigated ablation processes,

yields exact transient solutions of the

fundamental system of partial differential

equations . the method is applied

to various quartz shields and to one

teflon shield, which all evaporate so

readily under the conditions of the

problem at hand that practically no flow of molten shield material exists .

the solutions also show comparatively small

temperature changes parallel to the surface .

the results show that the nose

of the vehicle is cooled predominantly by

the evaporation of the quartz or the

teflon,. the rest of the vehicle's surface

is cooled by radiation of the quartz

or evaporation of the teflon . the large

mass transfer effects on the nose of

the vehicle are detrimental since the

resulting low surface temperatures prevent

the radiative heat transfer out of

the shield, which does not involve any

mass loss, from being the desirable

governing cooling factor .

.I 275

.T

the effect of lift on entry corridor depth and guidance

requirements for the return lunar flight .

.A

wong,t. and slye,r.

.B

nasa r-80, 1960.

.W

the effect of lift on entry corridor depth and guidance

requirements for the return lunar flight .

corridors for manned vehicles are
defined consistent with requirements
for avoiding radiation exposure and for
limiting values of peak
deceleration . use of lift increases the depth
of the entry corridor . mid-course
guidance requirements appear to be critical
only for the flight-path angle .
increasing the energy of the transfer orbit
increases the required guidance
accuracy for the flight-path angle .
corrective thrust applied essentially
parallel to the local horizontal
produces the maximum change in perigee
altitude for a given increment of
velocity . energy required to effect a
given change in perigee altitude
varies inversely with range measured
from the center of the earth .

.I 276

.T

reaction tests of turbine nozzles for supersonic velocities .

.A

keenan,j.h.

.B

asme trans. 1949, 773.

.W

reaction tests of turbine nozzles for supersonic velocities .

a machine for testing turbine nozzles by the reaction method, which was described in a previous paper, was used to test a series of convergent-divergent turbine nozzles . the results of these tests, along with the test of a convergent turbine nozzle, are compared with each other and with analytical values . two kinds of analytical values are employed, namely, the usual values obtained from an assumed isentropic expansion from inlet state to exhaust pressure, and the values obtained from the assumption that the processes in the nozzle are isentropic except for a normal shock which takes up a position in the nozzle such as to cause the stream to fill the exit area at the exhaust pressure whenever possible . this latter kind of analytical value involves no shock when the exit area can be filled at the exhaust pressure by means of isentropic processes only, or when the exhaust pressure is lowered so far that the shock has passed out of the passage . the agreement of the test results with the calculated results of this latter kind is good, and the disagreement which exists can be attributed largely to separation at the shock and to transmission of exhaust-pressure effects upstream through the boundary layer .

.I 277

.T

study of flow conditions and deflection angle at exit of two-dimensional cascade of turbine rotor blades at critical and supercritical pressure ratios .

.A

hauser,c.h., plohr,h.w. and sonder,g.

.B

naca rm e9k25, 1950.

.W

study of flow conditions and deflection angle at exit

of two-dimensional cascade of turbine rotor blades

at critical and supercritical pressure ratios .

an analysis was made of the flow conditions downstream of a cascade of turbine rotor blades at critical and supercritical pressure ratios . the results of five theoretical methods for determining the deflection angle are compared with those of an experimental method using the conservation-of-momentum principle and static-pressure surveys, and also are compared with an analysis of schlieren photographs of the flow downstream of the blades . a two-dimensional cascade of six blades with an axial width of 1.80 inches was used for the static-pressure surveys and for some of the schlieren photographs . in order to determine the flow conditions several blade chords downstream of the cascade, schlieren photographs were taken of the flow through a cascade of 18 blades having an axial width of 0.60 inch .

for the blade design studied, even at static-to-total pressure ratios considerably lower than that required to give critical velocity at the throat section, the flow was deflected in the tangential direction as predicted for the incompressible case . as the pressure ratio was lowered further, the aerodynamic loading of the rear portion of the blade reached a maximum value and remained constant .

after this condition was attained, the expansion downstream of the

cascade took place with a constant tangential velocity so that no

further increase in the amount of turning across the blade row and

no further increase in the loading of the blade was available .

.I 278

.T

on source and vortex distributions in the linearised

theory of steady supersonic flow .

.A

robinson,a.

.B

q.j.mech.app.math. 1948.

.W

on source and vortex distributions in the linearised

theory of steady supersonic flow .

the hyperbolic character of the differential

equation satisfied by the velocity

potential in linearized supersonic flow entails

the presence of fractional infinities

in the fundamental solutions of the equation .

difficulties arising from this fact can

be overcome by the introduction of hadamard's

finite part of an infinite integral .

together with the definition of certain counterparts

of the familiar vector operators

this leads to a natural development of the analogy between incompressible flow

and linearized supersonic flow . in particular, formulae

are derived for the field of

flow due to an arbitrary distribution of supersonic

sources and vortices .

applications to aerofoil theory, including the

calculation of the downwash in the

wake of an aerofoil, are given in a separate report (ref. 9) .

.I 279

.T

supersonic drag calculations for a cylindrical shell

wing of semicircular cross section combined with a

central body of revolution .

.A

beane,b.j. and ryna,b.m.

.B

douglas sm22627, 1956.

.W

supersonic drag calculations for a cylindrical shell

wing of semicircular cross section combined with a

central body of revolution .

a semi-circular ring wing with a body of revolution on

the axis is studied to find the wave and the vortex drag for

various chordwise lift distributions and for three values of

a parameter describing the wing geometry . using the

wave drag obtained from the chordwise loading that gives the

least drag, together with the vortex and skin friction drags, the maximum lift to drag ratio for each wing geometry is

computed . compared to the estimates made by lomax and

heaslet, somewhat lower drags are found .

.I 280

.T

the surface oil flow technique as used in high speed

wind tunnels in the united kingdom .

.A

stanbrook,a.

.B

rae tn.aero.2712, 1960.

.W

the surface oil flow technique as used in high speed

wind tunnels in the united kingdom .

an examination has been made of the

various versions of the surface oil

flow technique used in different high speed

wind tunnels . to provide

background information for this investigation

some systematic tests were made on

a simple model in a small supersonic tunnel .

the experience gained made it

possible to explain many of the variations

in terms of the different operating

conditions of the tunnels .

the time taken to form a pattern

on a typical model is, to a first

approximation, directly proportional to

the value of the parameter,

the factor being 36,000 12,000 . the

time taken appears to be independent

of the initial thickness of the oil sheet .

a general procedure for the development

of oil mixtures for any purpose

is suggested .

.I 281

.T

higher order approximations for relaxation oscillations .

.A

.B

.W

higher order approximations for relaxation oscillations .

the problem of solving asymptotic developments for all quantities

involved in relaxation oscillations has been solved by haag . this

paper indicates how one can carry out such developments in a case

which is simple enough to be treated explicitly .

.I 282

.T

jet effects on base pressure of conical afterbodies

at mach 1. 91 and 3. 12 .

.A

baughman,l.e. and kochendorfer,f.d.

.B

naca rm e57e06.

.W

jet effects on base pressure of conical afterbodies

at mach 1. 91 and 3. 12 .

data are presented which show the effect of a jet on base pressure for a series of conical afterbody-jet-nozzle combinations having boat-tail angles that varied from 0 to 11 and base-to-jet diameter ratios that varied from 1.11 to 2.67 . the jet nozzles had exit angles from 0 to 20 and were designed for exit mach numbers from 1.0 to 3.2 . pressure ratios up to 30 were tested for both a cold (air) and a hot numbers of 1.91 and 3.12 .

in general, base pressure increased for increasing values of boat-tail angle, nozzle angle, jet temperature, and jet total pressure and for decreasing values of base-to-jet diameter ratio, jet mach number, and free-stream mach number . the addition of tail surfaces produced only small changes in base pressure .

for all variables, base pressure is governed by the maximum pressure rise that can be supported by the wake fluid in the region of the trailing shock . the wake pressure ratio is in turn governed by the jet and free-stream mach numbers adjacent to the wake region and by the state of the boundary layer on the boattail and on the nozzle .

values of wake pressure ratio computed using the theory of korst, page, and childs were in good agreement with experimental values for convergent nozzles .

.I 283

.T

laminar heat transfer around blunt bodies in dissociated

air .

.A

kemp,n.h., rose,p.h. and detra,r.w.

.B

j.ae.sc. 26, 1959.

.W

laminar heat transfer around blunt bodies in dissociated

air .

a method of predicting laminar heat-transfer rates to blunt,

highly cooled bodies with constant wall temperature in dissociated

air flow is developed . attention is restricted to the case of

axisymmetric bodies at zero incidence, although two-dimensional

bodies could be treated the same way . the method is based on

the use of the /local similarity/ concept and an extension of the

ideas used by fay and riddell . a simple formula is given for

predicting the ratio of local heat-transfer rate to stagnation-point

rate . it depends on wall conditions and pressure distribution,

but not on the thermodynamic or transport properties of the hot

external flow, except at the stagnation point .

experimental heat-transfer rates obtained with correct

stagnation-point simulation and high wall cooling in shock tubes are

also presented and compared with the theoretical predictions .

on the whole, the agreement is good, although in regions of

rapidly varying pressure there is evidence that the local similarity

assumption breaks down, and the theory underestimates the

actual heat-transfer rate by up to 25 per cent .

.I 284

.T

the divergence of supersonic wings including chordwise

bending .

.A

biot,m.a.

.B

j. ae. scs. 23, 1956.

.W

the divergence of supersonic wings including chordwise

bending .

the static aeroelastic stability or divergence problem is investigated for thin supersonic wings when not only the spanwise bending and twist are taken into account but also the chordwise bending . the problem is treated in successive phases of increasing complexity from the two-dimensional curling-up of the leading edge to the three-dimensional stability of the cantilever wing . several methods of approach are developed including the nonlinear aspects of the structure and the aerodynamics . results indicate a strong dependence of stability on poisson's ratio and the magnitude of the deformation .

.I 285

.T

on the flutter of panels at high mach numbers .

.A

hedgepeth,j.m.

.B

j.ae.scs. 23, 1956.

.W

on the flutter of panels at high mach numbers .

there have recently arisen some questions as to the

possibility of panel flutter at high dynamic pressures and

mach numbers . in addition, some doubts have been raised

about the convergence of the galerkin method when applied to

such problems . this note is intended to shed light on these

matters .

.I 286

.T

effect of roll on dynamic instability of symmetric

missiles .

.A

murphy,c.h.

.B

j. ae. scs. 21, 1954.

.W

effect of roll on dynamic instability of symmetric

missiles .

this note attempts to extend the discussion by stating a

slightly neater form of generalized stability conditions and

describing certain experimental results on dynamic instability .

.I 287

.T

some theoretical low-speed loading characteristics

of swept wings in roll and sideslip .

.A

bird,j.d.

.B

naca r969, 1950.

.W

some theoretical low-speed loading characteristics

of swept wings in roll and sideslip .

the weissinger method for determining additional span loading for incompressible flow is used to find the damping in roll, the lateral center of pressure of the rolling load, and the span loading coefficients caused by rolling for wing plan forms of various aspect ratios, taper ratios, and sweep angles . in addition, the applicability of the method to the determination of certain other aerodynamic derivatives is investigated, and corrections for the first-order effects of compressibility are indicated .

the agreement obtained between experimentally and theoretically determined values for the aerodynamic coefficients indicates that the method of weissinger is well suited to the calculation of the additional span loading caused by rolling and for the calculation of such resulting aerodynamic derivatives of wings as do not involve considerations of tip suction .

.I 288

.T

the rolling up of the trailing vortex sheet and its

effect on the downwash behind wings .

.A

spreiter,j.r. and sacks,a.h.

.B

j. ae. scs. 18, 1951.

.W

the rolling up of the trailing vortex sheet and its

effect on the downwash behind wings .

the motion of the trailing vortices associated with a lifting wing is investigated by theoretical and visual-flow methods for the purpose of determining the proper vortex distribution to be used for downwash calculations . both subsonic and supersonic speeds are considered in the analysis .

it is found that the degree to which the vortices are rolled up depends upon the distance behind the wing and upon the lift coefficient, span loading, and aspect ratio of the wing . while the rolling up of the trailing vortices associated with high aspect-ratio wings is of little practical importance, it is shown that, with low-aspect-ratio wings, the trailing vortex sheet may become essentially rolled up into two trailing vortex cores within a chord length of the trailing edge .

the downwash fields associated with the two limiting cases of the flat vortex sheet and the fully rolled-up vortices are investigated in detail for both subsonic and supersonic speeds . the intermediate case in which the rolling-up process is only partially completed at the tail position is also discussed .

.I 289

.T

a theoretical study of the aerodynamics of slender

cruciform-wing arrangements and their wakes .

.A

spreiter,j.r. and sacks,a.h.

.B

naca r1296, 1957.

.W

a theoretical study of the aerodynamics of slender

cruciform-wing arrangements and their wakes .

a theoretical study is made of some cruciform-wing

arrangements and their wakes by means of slender-body theory . the

basic ideas of this theory are reviewed and equations are

developed for the pressures, loadings, and forces on slender

cruciform wings and wing-body combinations . the rolling-up of

the vortex sheet behind a slender cruciform wing is considered

at length and a numerical analysis is carried out using 40

vortices to calculate the wake shape at various distances behind

an equal-span cruciform wing at 45 bank . analytical

expressions are developed for the corresponding positions of

the rolled-up vortex sheets using a 4-vortex approximation to

the wake, and these positions are compared with the positions

of the centroids of vorticity resulting from the numerical analysis .

the agreement is found to be remarkably good at all distances

behind the wing .

photographs of the wake as observed in a water tank are

presented for various distances behind a cruciform wing at 0

and 45 bank . for 45 bank, the distance behind the wing

at which the upper two vortices pass between the lower two is

measured experimentally and is found to agree well with the

the calculation of loads on cruciform tails is considered in some detail by the method of reverse flow, and equations are

developed for the tail loads in terms of the vortex positions

calculated in the earlier analyses .

.I 290

.T

dynamic stability of a missile in rolling flight .

.A

bolz,r.e.

.B

j. ae. scs. 19, 1952.

.W

dynamic stability of a missile in rolling flight .

the paper sets down the equations of motion for a symmetric

rolling missile with respect to axes attached to the missile . the

missile may be jet (or rocket) propelled or coasting under

accelerating or decelerating conditions, respectively, wherein the

variable rolling velocity is derived from intentionally or

unintentionally /canted/ fins and or wings .

the equations contain a force and moment system that

includes, in addition to the usual forces and moments, those due to

magnus effects, misaligned surfaces, canted surfaces, jet

misalignment, and the linear accelerations in the plane normal to the

missile axis .

the results present general stability criteria for a rolling missile

which are summarized in the /discussion of stability ./

.I 291

.T

sweepback effects in the turbulent boundary-layer shock-wave

interaction .

.A

stalker,r.j.

.B

j. ae. scs. 27, 1960.

.W

sweepback effects in the turbulent boundary-layer shock-wave

interaction .

experiments are reported on the interaction of turbulent

boundary layers and shock waves with sweptback configurations .

they show that the peak pressure rise at separation, the

upstream influence ahead of separation, and the pressure rise at

reattachment for moderate sweep angles can all be understood by

simple extensions of available two-dimensional theories .

.I 292

.T

rapid laminar boundary layer calculations by piece-wise

application of similar solutions .

.A

smith,a.m.o.

.B

j. ae. scs. 23, 1956.

.W

rapid laminar boundary layer calculations by piece-wise

application of similar solutions .

a method is presented for the rapid calculation of the incompressible laminar boundary layer in an arbitrary flow around

either a two-dimensional or a rotationally-symmetrical body .

the solution is obtained without recourse to von karman's

momentum equation by means of a coarse step-by-step procedure

in which each segment of the velocity distribution is

approximated by one of the falkner-skan family of similar flows .

solutions have at least as much accuracy as those of any other

one-parameter approximate method, and in certain cases the

solutions become exact . in regions of accelerating velocity, the

accuracy appears to be very high . in decelerating flows,

separation is predicted somewhat early compared with exact solutions

that is, the method is conservative in contrast to the von

karman-pohlhausen procedure which sometimes fails to predict

separation that actually exists .

the method is the most rapid hand procedure known to the

author, provided the full history of the boundary layer is

required . if only a thickness such as is needed at one point on a

surface, then it is about equal in speed to the quadrature method .

but, if several values of or other properties along a surface are

required, it is appreciably faster than the quadrature method .

characteristically, only four steps are needed between the forward

stagnation point and the pressure peak . once the

velocity-distribution data are available, each step in a two-dimensional

calculation requires about 5 minutes, using a slide rule .

.I 293

.T

recent studies on the effect of cooling on boundary layer transition at mach 4.

.A

wisniewski,r.j. and jack,j.r.

.B

j. ae. scs. 28, 1961.

.W

recent studies on the effect of cooling on boundary

layer transition at mach 4.

the advent of high-speed flight has necessitated the study of

boundary-layer transition on highly cooled bodies .

investigations such as those of references 1-4 have concentrated

on this problem and have indicated, contrary to the trends

predicted by small-disturbance theory, that premature transition

can be found with cooling . this phenomenon, commonly called

detail in references 2-5 .

the purpose of this note is to report some recent transition

data obtained on a cooled cone in a mach 4 wind tunnel . the

model, a sharp-tip cone (included angle 13.5), was cooled by

liquid nitrogen to a temperature of -340 f . the cooling

method and the data analysis are similar to that described in

reference 3 .

.I 294

.T

an investigation of laminar transitional and turbulent

heat transfer on blunt-nosed bodies in hypersonic flow .

.A

cresci,r.j. and mackenzie,d.a.

.B

j. ae. scs. 27, 1960.

.W

an investigation of laminar transitional and turbulent

heat transfer on blunt-nosed bodies in hypersonic flow .

laminar, transitional, and turbulent heating rates have been measured by means of the shrouded model technique . the reynolds number was varied over a ninefold range,. the enthalpy ratio (stagnation to wall) varied from 2.3 to approximately 1.5 . two different pressure distributions were imposed on the model which consisted of a spherically capped cone .

the experimental data are compared to the laminar hypersonic boundary-layer theory and shown to be in good agreement on the conical portion of the model . on the spherical portion the data are approximately 20 per cent higher than the theoretical prediction . some of this discrepancy can be attributed to radiation to the nose of the model .

the fully developed turbulent heat-transfer data are compared to two theories .. (1) a relatively simple turbulent theory which is based on recent theoretical work and which takes into account the upstream history of the boundary layer, and (2) the flat-plate reference-enthalpy theory, which depends on only /local/ conditions . although both theories are in reasonable agreement with the data, the latter method is simpler and somewhat more accurate .

for transitional flow the theory mentioned first can be readily modified in order to permit reasonable estimates of transitional heat transfer to be obtained . on this basis it is possible to

estimate laminar, transitional, and fully developed turbulent heat

transfer under hypersonic blunt-body conditions .

the behavior of transition reynolds number based on

momentum thickness is also discussed and shown to be in quantitative

agreement with recent shock-tube measurements .

.I 295

.T

a note on transitional heat transfer under hypersonic conditions .

.A

constantino economos and paul a. libby

.B

research assistant and professor of aeronautical engineering,

respectively

polytechnic institute of brooklyn, brooklyn, n.y.

.W

a note on transitional heat transfer under hypersonic conditions .

in references 1 and 2 there were presented experimental data on

transitional heat transfer on a blunt body under hypersonic-flow

conditions obtained by the shroud technique . the data were compared

with a theoretical prediction of transitional heat transfer based

on a suggestion of persh . the agreement between theory and experiment

in the transitional region was found to be 'qualitatively good and

quantitatively fair' .

it is the purpose of this note to present some additional transitional

data obtained in conventional wind-tunnel tests and to indicate a

means for improving somewhat the agreement between transitional theory and experiment .

.I 296

.T

notes on waves through gases at pressures small compared

with the magnetic pressure, with applications to upper

atmosphere aerodynamics .

.A

lighthill,m.j.

.B

j. fluid mech. 9, 1960, 465.

.W

notes on waves through gases at pressures small compared

with the magnetic pressure, with applications to upper

atmosphere aerodynamics .

most treatments of magnetohydrodynamic

waves have confined physical

interpretation to cases when the alfven velocity a

is small compared with the sound

velocity a . here we consider the 'low-beta

situation', in which a is much

larger than a . then, except for two modes with

wave velocity a the only possible

waves are longitudinal ones, propagated

unidirectionally along lines of magnetic

force with velocity a . these can be

interpreted as sound waves, confined to

effectively rigid magnetic tubes of force .

hall-current effects do not alter these
conclusions (in contrast to the high-beta
situation), and finite conductivity
introduces only small dissipation .
 an application is made to the flow pattern
around a body moving through the
f layer of the ionosphere, where, although
neutral particles have a very large
mean free path, charged particles interact
electrostatically and, it is argued, may
be regarded as forming a continuous fluid
whose movement is independent of
that of the neutral particles . a body moving
at satellite speed or below would
then excite the above-mentioned unidirectional
sound waves, but no waves at
much faster alfven velocity . these considerations
suggest that its movement
would be accompanied by a v-shaped pattern of
electron density (figure 2),
which might be in part responsible for some
anomalous radar echoes that have
been reported .

.I 297

.T

compressibility effects in magneto-aerodynamic flows
past thin bodies .

.A

mccune,j.e. and resler,e.l.

.B

j. ae. scs. 27, 1960.

.W

compressibility effects in magneto-aerodynamic flows

past thin bodies .

the effects of compressibility on the steady motion of a highly conducting fluid past thin cylindrical bodies in the presence of a magnetic field are studied . procedures are developed for the solution of this class of magnetoaerodynamic problems over the entire mach number range and for all ratios of magnetic to fluid-dynamic pressure . the results obtained are analogous either to the ackeret theory or the prandtl-glauert rule of conventional aerodynamics, depending on the relative values of the flow speed and the appropriate speed of propagation of magnetoacoustic disturbances . the methods used and the physical interpretation of the solutions obtained vary according to the orientation of the magnetic field with respect to the flow direction .

the results of the theory are explained in terms of the anisotropic propagation of magnetoacoustic pulses studied previously by several authors .

.I 298

.T

incompressible wedge flows of an electrically conducting

viscous fluid in the presence of a magnetic field .

.A

yen,k.t.

.B

j. ae. scs. 27, 1960.

.W

incompressible wedge flows of an electrically conducting

viscous fluid in the presence of a magnetic field .

the purpose of this note is to discuss the two-dimensional

flow of an electrically conducting viscous fluid past a wedge

in the presence of a magnetic field . the governing differential

equations and boundary conditions are given and analyzed .

.I 299

.T

magnetohydrodynamic flow past a semi-infinite plate .

.A

meksyn,d.

.B

j. ae.scs. 29, 1962.

.W

magnetohydrodynamic flow past a semi-infinite plate .

the flow of viscous electrically conducting fluid

past a semi-infinite plate is considered . the

applied constant magnetic field and the constant

on-coming velocity of the fluid are in the direction parallel

to the plate .

in addition to reynolds number the flow in the

boundary layer depends on two parameters

and . the two simultaneous

ordinary nonlinear differential equations are solved by
the asymptotic method for the cases when
and respectively .
  the main results obtained are as follows . the
equations can be solved exactly for and .
the perturbation effect from infinity when k is large
depends on, whereas the perturbation effect from
zero when k is small depends on . for large k,
including there is no solution for . it is
assumed that the fluid is incompressible with constant
physical properties .
.I 300
.T
on a particular class of similar solutions of the equations
of motion and energy of a viscous fluid .
.A
reeves,b.l. and kippenhan,c.j.
.B
j. ae. scs. 29, 1962.
.W
on a particular class of similar solutions of the equations
of motion and energy of a viscous fluid .
  by introducing the similarity concept to the two-dimensional,
incompressible navier-stokes equations and energy equation, a
particular class of solutions is found . two general types of flows
are considered .. (1) laminar free convection--i.e., flows which
take place due to a body force--and (2) laminar forced

convection .

for free convection on vertical plates, similar solutions are obtained for two different power-law surface temperature variations, and it is shown that one of these solutions constitutes a new type of boundary problem . results of numerical integrations of the equations are compared with solutions of the similar boundary-layer equations for free convection, and it is demonstrated that a range of surface temperature variations exists for which the boundary layer equations are no longer valid .

for forced convection, it is shown that the use of similarity transformations provides an alternate method of deriving the ordinary differential equations for some well-known solutions, such as couette and stagnation point flows . solutions are obtained for radial converging or diverging flows between plane surfaces when the temperatures of the surfaces vary as arbitrary powers of the distance from the orgin . results of numerical integrations of the ordinary differential equations are presented for prandtl numbers of 0.01 and 1.0 and for linear surface temperature variations . some rather surprising results are obtained for diverging flows when separation occurs and some revealing comparisons with results from boundary-layer theory are made .

.I 301

.T

approximate design of sharp-cornered supersonic nozzles .

.A

m. a. rahman

.B

nack and sunderland consulting mech. and elec. engineers

.W

approximate design of sharp-cornered supersonic nozzles .

a modified parabolic curve appears to be in close proximity to that

obtained by either the method of characteristics or the wave method .

thus an attempt has been made to use analytic geometry to determine

approximately the contour of a two-dimensional, sharp-cornered

supersonic nozzle in a very short time .

.I 302

.T

approximations for the thermodynamic and transport properties of high

temperature air .

.A

hansen, c. f.

.B

nasa tr r 50, 1959 .

.W

approximations for the thermodynamic and transport properties of high

temperature air .

the thermodynamic and transport properties of high-temperature air are

found in closed form starting from approximate partition functions for

the major components in air and neglecting all minor components . the

compressibility, enthalpy, entropy, the specific heats, the speed of

sound, the coefficients of viscosity and of thermal conductivity, and

the prandtl numbers for air are tabulated from 500degree to 15,000degree

k over a range of pressure from 0.0001 to 100 atmospheres . the energy

of air and the mol fractions of the major components of air can be found from the tabulated values for compressibility and enthalpy . it is

predicted that the prandtl number for fully ionized air, which is in

complete equilibrium, will become small compared to unity, the order of

transparent to heat flux .

.I 303

.T

effect of variable heat recombination on stagnation

point heat transfer .

.A

fenster,s.j. and neyman,r.j.

.B

j. ae. scs. 29, 1962.

.W

effect of variable heat recombination on stagnation

point heat transfer .

earlier studies assume an average heat of formation of atoms

based upon external flow conditions . it is shown that equilibrium

heat transfer decreases by 35 for a typical mach number 24 case

when allowance is made for the proportions of air components .

the variable recombination energy also results in atom mass

fractions which are realistically less for equilibrium than frozen

situations throughout the cold-wall boundary layer .

.I 304

.T

first-order approach to a strong interaction problem

in hypersonic flow over an insulated flat plate .

.A

oguchi,h.

.B

univ. tokyo aero.res r330, 1958.

.W

first-order approach to a strong interaction problem

in hypersonic flow over an insulated flat plate .

the present paper concerns with

the strong interaction phenomenon over an

insulated semi-infinite flat plate with a sharp

leading edge . in particular the main interest

is in the consistent treatment in which the

boundary-layer solution may be joined

continuously with the inviscid solution regarding

flow variables including pressure, normal

velocity, temperature (or streamwise velocity) and density .

it is shown that the behavior of the inviscid

solution may be consistent with that of the

boundary-layer solution to at least first-order

approximation that is correct to the order of,

where m is the mach

number of undisturbed flow, r the reynolds

number based on the distance from leading

edge and the ratio of specific heats . then

the first-order boundary-layer problem is

formulated under such an external circumstance

and an attempt is made for arriving at the solution .

actual calculations are carried out for both cases of air and helium . from the solution

it is found that the region in which the viscous

effect plays a significant role is ranged over

from 0 to a certain finite value of n, say n, in

terms of the similarity coordinate n in the

corresponding incompressible boundary layer .

the numerical results moreover indicate

that the induced pressure is considerably smaller

than the estimate of lees (7) obtained by

his approximate method in which the effect

of the first-order induced pressure on the

boundary layer is ignored and no survey of the

first-order boundary-layer equation is made .

the present results are also found to be in

excellent agreement with experimental data

recently obtained in helium flow by erickson (15) .

.I 305

.T

hypersonic strong viscous interaction on a flat plate

with surface mass transfer .

.A

li,t.y. and gross,j.f.

.B

heat transfer and fluid mech. inst. 1961, 146.

.W

hypersonic strong viscous interaction on a flat plate

with surface mass transfer .

the present report gives an account of

the development of an

approximate theory to the problem of hypersonic

strong viscous interaction

on a flat plate with mass-transfer at the

plate surface . the disturbance

flow region is divided into inviscid and

viscous flow regions . the

hypersonic small perturbation theory is applied

to the solution of the inviscid

flow region . the method of similar solutions

of compressible laminar

boundary layer equations is applied to the

treatment of the viscous flow

region . the law of surface mass-transfer

for similar solutions is derived .

the pressure and the normal velocity are

matched between the inviscid and

viscous flow solutions . formulas for induced

surface pressure, boundary

layer thickness, skin friction coefficient,

and heat transfer coefficient

are obtained . numerical results and their

significance are discussed .

future improvements are indicated .

.I 306

.T

second approximation to laminar compressible boundary

layer on flat plate in slip flow .

.A

maslen,s.h.

.B

naca tn.2818, 1952.

.W

second approximation to laminar compressible boundary

layer on flat plate in slip flow .

the first-order solution for the laminar compressible boundary-layer

flow over a flat plate at constant wall temperature is given . the

effect of slip at the wall as well as the interaction between the

boundary-layer flow and the outer stream flow are taken into

consideration . the solution is obtained explicitly in terms of the known zero

order, or continuum, solution . no

assumptions regarding the prandtl

number or viscosity-temperature law need be made . it is found that the

first-order solution gives a decrease in heat transfer and, for

supersonic flow, an increase in skin friction .

for subsonic flow there is no

first-order shear effect . the change in heat transfer is due to slip

and the change in friction is due to the interaction of the zero- and

first-order velocities at the outer edge of the boundary layer .

.I 307

.T

an approximate solution of hypersonic laminar boundary

layer equations and its application .

.A

nagakura,t. and naruse,h.

.B

j. phys. soc. japan, 12, 1957, 1298.

.W

an approximate solution of hypersonic laminar boundary

layer equations and its application .

approximate formulae of the displacement thickness and the skin

friction of the hypersonic laminar boundary layer are derived by use of

von karman's integral method, assuming the heat-insulated wall, the

prandtl number of unity and chapman and rubesin's formula for the

variation of viscosity with temperature .

the results obtained are

compared with some exact solutions .

because of the good agreement, it

seems that these formulae are very useful .

these formulae, together with the tangent-wedge-approximation, are

applied to the viscous flow over

slender bodies with a sufficiently sharp

leading edge . as an example, the

pressure distribution over a flat plate

is calculated numerically over the

entire region of the surface .

comparison with other author's theoretical

results as well as experimental

values is made .

.I 308

.T

on the hypersonic viscous flow past a flat plate with

suction or injection .

.A

yasuhara,m.

.B

j. phys. soc. japan, 12, 1957, 177.

.W

on the hypersonic viscous flow past a flat plate with

suction or injection .

the hypersonic viscous flow past a flat

plate with suction or injection

is dealt with by karman-pohlhausen's

method in special cases when

suction or injection velocity proportional

to, especially

for the region of strong interaction between

the shock wave and the

boundary layer, were p is the pressure on

the plate and x is the

distance measured along the plate from its leading edge .

several numerical examples are given,

which shows similar effects of

injection to those in the case of incompressible

flow that the injection

makes all the height of the shock wave,

the thickness of the boundary

layer and the pressure on the plate larger

than those in the case of no

injection . on the contrary, in the case of

suction no remarkable change

both in the height of the shock wave and

the pressure on the plate can

be seen and only the velocity profile in

the boundary layer is affected

by the suction .

.I 309

.T

on the motion of a flat plate at high speed in a viscous

compressible fluid, ii, steady motion .

.A

stewartson,k.

.B

j. ae. scs. 22, 1955, 303.

.W

on the motion of a flat plate at high speed in a viscous

compressible fluid, ii, steady motion .

the theory of the steady flow of a viscous compressible fluid

past a flat plate at high mach number due to lees and

probstein is extended by a more complete discussion of the flow in

the inviscid layer between the shock wave and the boundary layer .

it is shown that similar solutions exist in this layer, analogously

to those found by li and nagamatsu in the boundary layer, and

that the two may be joined to give, allowing one minor

assumption, a full account of the flow . it is shown that the
boundary-layer equations may be reduced to those for an incompressible
fluid and that the von karman-pohlhausen method describes
the flow in it with good accuracy . the tangent wedge
approximation for the pressure on the plate, used by lees and his
collaborators, is found to be in deficit
by 10 per cent for air . finally,
it is shown that the theory for weak interaction cannot be
extended further without a complete knowledge of the flow .

.I 310

.T

hypersonic viscous flow over a flat plate .

.A

lees, l. and probstein, r. f.

.B

princeton univ. aero eng. r195, 1952

(abstract by e.m.keen)

.W

hypersonic viscous flow over a flat plate .

in dealing with the steady laminar viscous flow over a
semi-infinite flat plate some of the following topics are discussed .
the streamline in the boundary layer over a leading edge of given
thickness . the rate of growth of the boundary layer in the main stream, and
causes of pressure variations .
asymptotic solutions for thn downstream flow region, including the
joining interaction of shock waves at the leading edge .
pressure variations in the interanl viscous flow layer and in external

inviscid flow considered as prandtl meyer flow . in cases of streamline

deflection, the free stream mach number, zero pressure gradient, and

surface pressure distribution . asymptotic solutions for cases of

fluid injection of a cool gas . prandtl heat transfer . the joining

interaction between the external inviscid flow and the internal viscous

flow layer . steady laminar hpyersonic viscous flow over a flat wedge

and a cone .

.I 311

.T

a method for predicting the onset of buffeting and other separation

effects from wind tunnel tests on rigid models .

.A

pearcey, h. h.

.B

n.p.l. aero. 358, fm 2763. dec. 1958 .

.W

a method for predicting the onset of buffeting and other separation

effects from wind tunnel tests on rigid models .

the method is based on the observation of the divergence that occurs in

the variation of mean static pressure at the trailing edge of an

aircraft wing at the critical stage in the development of boundary-layer

separation when its influence first spreads to the trailing edge and

thereby to the overall flow .

the significance of the trailing-edge pressure variations and their

connection with the effects that separation has on the mean and unsteady

loads is discussed for various types of separation . good prediction

can be obtained from wind-tunnel tests, or warning provided in flight, for low-speed separations and for shock-induced ones up to the stage at

which the shock wave reaches the trailing edge . related divergences in

wake width, lift coefficient, or shock position can also be used .

pressure measurements at other isolated points often indicate the type

of separation .

certain special considerations apply for swept wings .

the various flow changes that are considered are illustrated by

schlieren photographs and described in an appendix .

.I 312

.T

chordwise pressure distributions over several naca

16 series airfoils at transonic mach numbers up to

1.25 .

.A

ladson,c.l.

.B

nasa memo 6-1-59l, 1959.

.W

chordwise pressure distributions over several naca

16 series airfoils at transonic mach numbers up to

1.25 .

a two-dimensional wind-tunnel

investigation of the pressure

distributions over several naca 16-series

airfoils with thicknesses of

and design lift coefficients of

the langley airfoil test apparatus

at transonic mach numbers from 0.7 to

number from 2.4 x 10 to 2.8 x 10 and

in angle of attack from -10 to

and schlieren flow photographs

are presented without analysis .

.I 313

.T

on alternative forms for the basic equations of transonic

flow theory .

.A

spreiter,j.r.

.B

j. ae. scs. 21, 1954.

.W

on alternative forms for the basic equations of transonic

flow theory .

attention has been called by numerous authors to the

possibility of certain alternative forms for the equations for

transonic flow about thin wings . it is the purpose of this note

to contribute to this discussion and to indicate some reasons for

the selection of one form of these in preference to another more

widely used form .

.I 314

.T

simplified method for determination of the critical

height of distributed roughness particles for boundary

layer transition at mach numbers from 0 to 5.

.A

braslow,a.l. and knox,e.c.

.B

naca tn.4363, 1958.

.W

simplified method for determination of the critical

height of distributed roughness particles for boundary

layer transition at mach numbers from 0 to 5.

a simplified method has been devised for determination of the critical height of three-dimensional roughness particles required to promote premature transition of a laminar boundary layer on models of airplanes or airplane components in a wind tunnel with zero heat transfer . a single equation is derived which relates the roughness height to a reynolds number based on the roughness height and on local flow conditions at the height of the roughness, and charts are presented from which the critical roughness height can be easily obtained for mach numbers from 0 to 5 . a discussion of the use of these charts is presented with consideration of various model configurations .

the method has been applied to various types of configurations in several wind-tunnel investigations conducted by the national advisory committee for aeronautics at mach numbers up to 4, and in all cases the calculated roughness height caused premature boundary-layer transition for the range of test conditions .

.I 315

.T

scale effects at high subsonic and transonic speeds

and methods for fixing transition in model experiments .

.A

haines,a.b., holder,d.w. and pearcey,h.h.

.B

arc r + m3012, 1954.

.W

scale effects at high subsonic and transonic speeds

and methods for fixing transition in model experiments .

the major scale effects at high subsonic and transonic

speeds arise from differences between the conditions

under which laminar and turbulent boundary layers separate, and in

how they behave after separation . for turbulent

boundary layers, these conditions and behaviour do not vary greatly

as the reynolds number is changed and in many

examples, it has been shown that they are similar for the turbulent

layers that occur naturally at high reynolds number

and for boundary layers in which transition to turbulent flow is fixed

artificially . the scale effects arising in wind-tunnel

tests made at low reynolds number may, therefore, often be

minimised by fixing transition to turbulent flow by

introducing an artificial disturbance such as that produced by

excrescences attached to the surface . the fact that the

effects of separation are often less severe for laminar layers than

for the turbulent layers that are likely to be encountered

at full scale, makes it all the more important to do this whenever

possible .

several methods which can be used to fix transition are described,

and the results obtained by using them are compared .

in general, in experiments in two-dimensional flow, good agreement is found, and explanations can be advanced for cases in which discrepancies occur . several uncertainties and difficulties that arise in fixing transition are discussed and illustrated by examples . in particular, special care is needed in interpreting the results obtained with transition fixed at very low reynolds numbers (say, less than about r = 1 x 10 based on local chord for wings of about 0.1 thickness chord ratio and possibly higher reynolds numbers for thinner wings) .

the difficulties of fixing transition satisfactorily are increased for three-dimensional wings, particularly if they are swept-back or highly tapered (i.e., small chord and reynolds number near the tip) and if the tests cover a large range of incidence including high incidences for which the flow may separate from very close to the leading edge . under these circumstances, it is frequently necessary to place the excrescences at different chordwise positions for low and high angles of incidence, and this is inconvenient in practice . more research is needed before sound recommendations can be made as to how and where transition should be fixed on such models, particularly since in routine testing, it is often not possible to check the effects of transition-fixing fully .

in the sections dealing with three-dimensional tests, examples are given of the spurious results that have been avoided successfully by fixing transition, of the conditions where even at low reynolds numbers artificial fixing of transition may not be necessary to give a turbulent boundary layer ahead

of the shock, and of the conditions under which there

are some doubts whether the methods used for fixing transition

have been satisfactory .

.I 316

.T

the occurrence and development of boundary layer separations at high

incidences and high speeds .

.A

pearcey, h. h.

.B

r + m 3109, arc 17901, september 1955 .

.W

the occurrence and development of boundary layer separations at high

incidences and high speeds .

this note describes the manner in which the onset of the effects of

boundary-layer separation varies with mach number for two-dimensional

aerofoils, and discusses the influence of section shape as far as it is

known . a brief qualitative description is given of the mechanism

underlying the development of the separated flow and its effects,

followed by a discussion of some of the ways in which this is likely to

differ for swept-back wings at high speeds . finally, the need is

emphasized for continued work in a broadening field .

.I 317

.T

non-equilibrium flow of an ideal dissociating gas .

.A

freeman, n. c.

.B

j. fluid mech. v. 4, 1958 .

.W

non-equilibrium flow of an ideal dissociating gas .

the theory of an'ideal dissociating'gas developed by lighthill/1957/for conditions of thermodynamic equilibrium is extended to non-equilibrium conditions by postulating a simple rate equation for the dissociation process/including the effects of recombination/ . this equation contains the'equilibrium'parameters of the lighthill theory plus a further dissociation phenomena .

the behaviour of this gas is investigated in flow through a strong normal shock wave and past a bluff body . the assumption is made that the gas receives complete excitation of its rotational and vibrational degrees of freedom in an infinitesimally thin region according to the familiar rankine-hugoniot shock wave relations before dissociation begins . the variation of the relevant thermodynamic variables down-stream of this region is then computed in a few particular cases . the method used in the latter case is an extension of the'newtonian'theory of hypersonic inviscid flow . in particular, the case of a sphere is treated in some detail . the variation of the shock shape and the sphere diameter to the length scale of the dissociation process, is exhibited for conditions extending from completely undissociated flow to dissociated flow in thermal equilibrium . results would indicate that significant and observable changes from the undissociated values occur, although values for the non-equilibrium parameter are not, at present, available .

.I 318

.T

inviscid hypersonic flow past blunt bodies .

.A

maslen,s.h. and moeckel,w.e.

.B

j. ae. scs. 24, 1957.

.W

inviscid hypersonic flow past blunt bodies .

two methods are shown for the calculation of the flow field between a blunt body and the shock associated with it for the case of hypersonic flow . real gas effects are included . the solutions consider only symmetric flows--that is, symmetric bodies at zero incidence .

one method consists in tracing successive stream tubes around the body and leads to iterations on the initially assumed position of the shock . the second is an integral method closely analogous to the karman-pohlhausen procedure for boundary layers . a distinction is made between round-nosed and flat-nosed bodies, and both cases are discussed .

a specific example corresponding to a re-entry missile situation is calculated,. the two methods agree within a few per cent . comparison is also made with other known solutions in the stagnation region .

.I 319

.T

propagation of weak disturbances in a gas subject to relaxation effects .

.A

moore,f.k. and gibson,w.e.

.B

j. ae. scs. 27, 1960.

.W

propagation of weak disturbances in a gas subject to

relaxation effects .

a generalized wave equation is derived for sound disturbances

in a gas when relaxation effects connected with, for example,

molecular vibration or dissociation are important . solutions

involving discontinuous wave fronts are presented, and it is shown

that, under certain assumptions, the complete wave equation

reduces to a variant of the telegraph equation . detailed solutions

are presented for disturbance fields produced by a wavy wall in

subsonic and supersonic flow and a simple wedge in supersonic

flow . this study is viewed as a step in the development of a

theory of small disturbances of a high-temperature gas, as is found

behind the shock in hypersonic flight .

.I 320

.T

comment on improved numerical solution of the blasius problem with

three-point boundary conditions .

.A

leigh, d. c.

.B

j. aero. sc. v. 29, may 1962 .

.W

comment on improved numerical solution of the blasius problem with

three-point boundary conditions .

attention is drawn to a previous accurate solution to the problem .

.I 321

.T

improved numerical solution of the blasius problem with three-point

boundary conditions .

.A

christian, w. j.

.B

j. aero. sc. v. 28, november, 1961 .

.W

improved numerical solution of the blasius problem with three-point

boundary conditions .

the blasius equation describes the velocity distribution resulting from

laminar, constant-pressure mixing of a stationary fluid layer and a

moving stream . in connection with a numerical procedure for the univac

based on analytic continuation of the function f' . high-speed computers

now make it feasible to use analytic continuation for numerical

integration of single-point boundary-value problems such that, within

the limits of taylor's expansion, truncation error may be made

arbitrarily small . a brief description of the application of the

routine is given .

.I 322

.T

on the numerical solution of the blasius problem with three-point boundary conditions .

.A

toba, k.

.B

j. aero. sc. v. 29, p480-1, april, 1962 .

.W

on the numerical solution of the blasius problem with three-point

boundary conditions .

relates to a technique for approximate determination of the initial

parameters . the technique is an application of the asymptotic

integration method introduced by meksyn and has been applied to the computation

of the skin friction for shock-generated boundary-layer flow .

.I 323

.T

vorticity interaction at an axisymmetric stagnation

point in a viscous incompressible fluid .

.A

kemp,n.h.

.B

j. ae. scs. 26, 1959.

.W

vorticity interaction at an axisymmetric stagnation

point in a viscous incompressible fluid .

the purpose of the present note is to give an exact solution of

the incompressible navier-stokes equations at an axisymmetric

stagnation point with vorticity in the oncoming flow which varies

linearly with distance from the axis . this solution has application to the hypersonic axisymmetric blunt body problem, for which

lighthill has shown the vorticity in the inviscid shock layer is

very nearly of this form .

.I 324

.T

vorticity effect on the stagnation point flow of a viscous

incompressible fluid .

.A

rott, n. and lenard, m.

.B

j. aero. sc. v. 26, august, 1959 .

.W

vorticity effect on the stagnation point flow of a viscous

incompressible fluid .

the effect of vorticity on axisymmetric stagnation point boundary layer

calculations is investigated by calculating a perturbation to the

stagnation point flow . the shear caused by the vorticity effect is

found to be surprisingly large,.the slope of the shear curve /at zero

vorticity/ as calculated by kemp agrees perfectly with the value deduced

in this note .

.I 325

.T

heat transfer to constant property laminar boundary

layer flows with power function free stream velocity

and wall temperature variation .

.A

levy,s.

.B

j.ae.scs. 19, 1952.

.W

heat transfer to constant property laminar boundary

layer flows with power function free stream velocity

and wall temperature variation .

numerical computations have been performed for the boundary-layer form of the energy equation for incompressible flows with power-function variation of free-stream velocity (u = cx) and of wall temperature (t = ax), the pertinent solutions of the momentum equation in this case being those of hartree . the numerical computations given herein are to some extent a repetition of those given by schuh and by chapman and rubesin, the object of the present computations being the resolution of discrepancies appearing in the previous solutions and an extension of their range . ibm machine calculations were employed in the finite difference calculation presently utilized, the results thereof covering a range of wall-temperature function exponents from values of m(4, 1, 0, -0.0904) . the accuracy of the numerical computations is examined in detail, and the accuracy of the computed functions at the wall, which determine the heat-transfer rate, is estimated to be within 2 per cent .

examination of the results reveals that the results of schuh for the flat plate are in error . for the range of the calculations, it was found that the local heat-transfer coefficient can, with the exception of large negative values, be expressed within 5 per cent as where the exponent of the prandtl number varies from 0.254

to 0.367 for -0.0904 and where the function

can be approximated by the equation

.I 326

.T

forst-order slip effects on the compressible laminar

boundary layer over a slender body of revolution in

axial flow .

.A

shen,s.f. and solomon,j.m.

.B

j.ae.scs. 28, 1961.

.W

forst-order slip effects on the compressible laminar

boundary layer over a slender body of revolution in

axial flow .

analysis of the

compressible boundary layer with transverse curvature in first

order slip flow . no boundary-layer interaction effects are considered

and only the zero pressure-gradient case is examined .

.I 327

.T

on local flat plate similarity in the hypersonic boundary

layer .

.A

moore,f.k.

.B

j. ae. scs. 28, 1961.

.W

on local flat plate similarity in the hypersonic boundary

layer .

a study is made of lees' /local flat-plate similarity/ rule for

the hypersonic laminar boundary layer . it is shown that this

rule is exact under assumptions commonly invoked in the

inviscid theory of hypersonic flow .

beginning from this theoretical basis, a modified local

flat-plate similarity scheme is derived, involving separate rules for

velocity and enthalpy profiles, and is compared with exact

similarity solutions and with the existing theory of hypersonic

leading-edge interaction .

.I 328

.T

the boundary layer near the stagnation point in hypersonic

flow past a sphere .

.A

herring,t.r.

.B

j. fluid mech. 7, 1960, 257.

.W

the boundary layer near the stagnation point in hypersonic

flow past a sphere .

flow properties behind shock waves caused by bluff bodies

traveling at supersonic speeds are of major importance in missile

and high-speed aircraft design . paper presents a mathematical solution for the laminar boundary layer near the stagnation point of a sphere . surface temperature is free-stream static and shock is strong . air is assumed calorically and thermally perfect with a prandtl number of 0.72 and a dynamic viscosity directly proportional to temperature .

based on work of homann (zamm 16, p. 153, 1936) and lighthill simultaneous differential equations for the velocity and temperature profiles . these are solved by numerical integration along a normal to the surface using a digital computer . results are presented as functions of free-stream mach number, reynolds number, and specific heat ratio . as increases, boundary-layer thickness is shown to decrease while shock stand-off distance increases . stand-off distance also decreases with increasing and decreasing specific heat . for constant and specific heat ratio, the product of skin-friction coefficient and the square root of decreases with increasing only approaching a constant value at greater than 10,000 .

reviewer's comment is concerned with the perfect gas assumption for air . author suggests that the effects of dissociation on flow properties are accounted for by a proper choice of specific heat ratio . a consideration of the kinetics of chemical reaction in the cooled boundary layer emphasizes the oversimplification of this approach . the effect on transport properties could have been approximated in present analysis by changing the prandtl number to one more representative of the existing pressures and temperatures .

.I 329

.T

various aerodynamic characteristics in hypersonic rarefied

gas flow .

.A

probstein,r.f. and kemp,n.h.

.B

j. ae. scs. 27, 1960.

.W

various aerodynamic characteristics in hypersonic rarefied

gas flow .

this paper considers the problem of calculating viscous

aerodynamic characteristics of blunt bodies at hypersonic speeds and

at sufficiently high altitudes where the appropriate mean free

path becomes too large for the use of familiar boundary-layer

theory but not so large that free molecule concepts apply .

results of an order-of-magnitude analysis are presented to

define the regimes of rarefied gas flow and the limits of

continuum theory . based on theoretical and experimental

evidence, the complete navier-stokes equations are used as a

model, except /very close/ to the free molecule condition . this

model may not necessarily give the shock wave structure in

detail but satisfies overall conservation laws and should give a

reasonably accurate picture of all mean aerodynamic quantities .

in this /intermediate/ regime there are two fundamental classes

of problems .. a /viscous layer/ class and a /merged layer/

class, the latter corresponding to a larger degree of rarefaction .

for the viscous layer class there is a thin shock wave, but the shock layer region between the shock and the body is fully viscous, although the viscous stresses and conductive heat transfer are small at the shock wave boundary . here, the use of the navier-stokes equations with outer boundary conditions given by the hugoniot relations is justified . for the merged layer class, the shock wave is no longer thin, and the navier-stokes equations can be used to give a solution which includes the shock structure and has free-stream conditions as outer boundary conditions . a simpler procedure is presented for /incipient merged/ conditions where the shock may no longer be considered an infinitesimally thin discontinuity but where it has not thickened sufficiently to entail the /fully merged layer/ analysis . in this case we approximate the shock by a discontinuity obeying conservation laws which include curvature effects, viscous stresses, and heat conduction .

for a sphere and cylinder it is shown that the navier-stokes equations can be reduced to ordinary differential equations for both the viscous and merged layer class of problems . solutions of these equations, when used in connection with hypersonic flow problems, are in general only valid in the stagnation region . to illustrate the viscous layer solutions, numerical calculations have been performed for a sphere and cylinder with the assumption of constant density in the shock layer, which is a useful approximation at hypersonic speeds . to illustrate the merged layer solution, calculations have been carried out for a sphere using the incipient merged layer approximation .

results are presented for detachment distance, surface shear, and heat-transfer rate in the stagnation region of a highly cooled sphere flying at hypersonic speed . with decreasing reynolds number, the shear and heat transfer are shown to increase above the extrapolated boundary-layer values in the viscous layer regime and then to begin falling in the incipient merged regime . as the reynolds number decreases in the incipient merged regime, the density in the shock layer increases, and the static and stagnation enthalpy behind the shock decrease .

calculations performed for an insulated sphere show that, with decreasing reynolds number in the incipient merged regime, the density in the shock layer decreases,. the total enthalpy behind the shock and at the stagnation point increase so that they are higher than the free-stream total enthalpy,. and the stagnation-point pressure behaves like the total enthalpy .

for the highly cooled cylinder in the viscous layer regime, the same quantities are presented as for the sphere . the increase found in shear and heat transfer above extrapolated boundary-layer theory is small, in agreement with vorticity interaction theory .

a discussion is given of the behavior of available experimental data for viscous flow quantities in the intermediate regime and the behavior predicted by the results of the present calculations . qualitative agreement is indicated .

.I 330

.T

taylor instability of finite surface waves .

.A

emmons,h.w., chang,c.r. and watson,b.c.

.B

j. fluid mech. 7, 1960.

.W

taylor instability of finite surface waves .

the instability of the accelerated interface

between a liquid (methanol or carbon

tetrachloride) and air has been investigated

experimentally for approximate

sinusoidal disturbances of wave-number range

from well below to well above the

cut-off . the growth rates are measured and

compared with theoretical results .

a third-order theory shows the phenomena

of overstability which is found in the

experimental results . some measurements

of later stages of growth agree

moderately well with the available theory

and disclose some additional

phenomena of bubble competition, helmholtz

instability with transition to turbulence,

and jet instability with production of drops .

.I 331

.T

effects of surface tension and viscosity on taylor

instability .

.A

bellman,r. and pennington,r.h.

.B

q. app. math. 12, 1954.

.W

effects of surface tension and viscosity on taylor

instability .

the model used is that of two

fluids of infinite depth, with the interface

initially in the form of a sine wave with

amplitude small compared to wave length .

the fluids are considered incompressible,

and only the linear terms in the equations of

hydrodynamics are used . the first four

sections discuss the effects of surface tension

and viscosity . the fifth gives a few numerical

results to illustrate the main points of

the preceding sections .

.I 332

.T

similitude of hypersonic real-gas flows over slender

bodies with blunted noses .

.A

cheng,h.k.

.B

j. ae. scs. 26, 1959, 575.

.W

similitude of hypersonic real-gas flows over slender bodies with blunted noses .

on the basis of the hypersonic small-perturbation theory, the laws of similitude for hypersonic inviscid flow fields over thin or slender bodies are examined, and the restrictions to ideal gases with constant specific heats and to bodies with pointed noses are removed . only steady plane or axisymmetric flows are considered .

inspection of the governing system of equations shows that a similitude law exists for flow fields, under local thermal equilibrium, having the same free-stream atmosphere . for flows of ideal gas with constant specific heats, the requirement of the same free-stream atmosphere--i.e., the same composition, pressure, and density--can be replaced by the requirement of the same ratio of specific heats .

for flows over blunted wedges or cones, special laws of similitude can be obtained .

application of the similarity rules is examined for the case of hypersonic flows of an ideal gas with over flat plates with blunt leading edges, and for the case of equilibrium air flows over wedges . the possibility of simulating nonequilibrium flows over slender or thin bodies is also pointed out .

.I 333

.T

boundary-layer interaction on a yawed infinite wing in hypersonic flow .

.A

robert j. whalen

.B

principal scientist, flight sciences laboratory, inc.,

buffalo, n.y.

.W

boundary-layer interaction on a yawed infinite wing in hypersonic

flow .

the equations are given for the laminar boundary-layer

equations on a yawed infinite wing for constant

wall temperature, under the combined howorth and

mangler transformation . diagrams show the relatively

small influence of yaw, the increase of boundary-layer

secondary flow, and the variation of the local heat

transfer rate with yaw .

.I 334

.T

influence of the leading-edge shock wave on the laminar boundary layer

at hypersonic speeds .

.A

lester lees

.B

california institute of technology

.W

influence of the leading-edge shock wave on the laminar boundary layer

at hypersonic speeds .

in order to bring out the importance of the leading-edge region at

hypersonic speeds, the influence of the leading-edge shock wave on

the laminar boundary layer is investigated in two simple cases of

steady flow over a semi-infinite, insulated flat plate.. (1) sharp leading edge., (2) blunt leading edge, as approximated by a normal shock wave . the streamlines that enter the boundary layer over a large region of the plate surface has previously crossed the shock wave very near the leading-edge, where the shock is strong and highly curved . consequently, the temperature at the outer edge of the boundary layer is appreciably higher than free-stream temperature, and the vorticity there is not zero . the effects of this shock-wave larger than the usual /errors/ made in the boundary-layer theory, and an estimate of these effects can therefore be obtained within the framework of that theory . the numerical magnitude of the shock-wave influence is found to be appreciable . for the case of the blunt leading edge the slope of the curve of induced pressures plotted against the hypersonic interaction parameter closely approaches the experimental data of hammitt and bogdonoff obtained in helium at large values of this parameter . these approximate results show that the influence of the leading-edge region at hypersonic speeds requires careful theoretical and experimental study .

.I 335

.T

the interaction between boundary layer and shock waves in transonic flow .

.A

liepmann, h. w.

.B

j. aero. sc. v. 13, december, 1946 .

.W

the interaction between boundary layer and shock waves in transonic

flow .

experiments of transonic flow past a circular arc profile show that the

shock-wave pattern and the pressure distribution are strongly dependent

upon the state of the boundary layer . a change from laminar to

turbulent boundary layer at a given mach number changes the flow pattern

considerably .

shock waves can interact with the boundary layer in a manner similar to

a reflection from a free jet boundary . these shock waves are not

distinctly discernible from pressure distribution measurements .

.I 336

.T

simplified laminar boundary layer calculations for

bodies of revolution and for yawed wings .

.A

rott,n. and crabtree,l.f.

.B

j. ae. scs. 19, 1962.

.W

simplified laminar boundary layer calculations for

bodies of revolution and for yawed wings .

since the introduction of momentum methods in

boundary-layer calculations by von karman and pohlhausen, many

improvements have been proposed . an especially simple solution

reduces the problem to a quadrature . here, it is proposed to

extend these methods to elementary three-dimensional cases and to

compressible laminar boundary-layer calculations . for comparison, the corresponding problems for the turbulent boundary

layer are also discussed briefly .

.I 337

.T

boundary layer transition with gas injection .

.A

scott,c.j. and anderson,g.e.

.B

j. ae. scs. 25, 1958.

.W

boundary layer transition with gas injection .

the mass-injection process has been proposed as a method

of cooling aerodynamic surfaces, and, since the amount of

coolant required to maintain practical wall temperatures is

considerably larger for turbulent than for laminar boundary layers,

knowledge of the effect of the cooling method on the transition

process is certainly important . exploratory studies reported

here were conducted at mach number 3.7 to ascertain the

effects of gas injection on the stability of the laminar boundary

layer on a conical surface .

.I 338

.T

mass transfer cooling at mach number 4. 8.

.A

leadon,b.m., scott,c.j. and anderson,g.e.

.B

j. ae. scs. 25, 1958.

.W

mass transfer cooling at mach number 4. 8.

mass-transfer experiments on a 5 mil wire porous cone

of 20 total angle have been conducted at using

air and helium injection . details of the experimental technique

are described in references 1 and 2 . in the laminar boundary

layer the recovery factors and heat-transfer coefficients measured

with zero injection agreed within per cent with theory .

transition reynolds numbers observed on the porous cone with

zero injection were half as large as observed on a smooth,

impermeable model of identical geometry in the same channel, but

injection of large amounts of air or helium did not cause

transition to move forward from its zero-injection position on the

porous cone . distributed roughness of this type apparently does

not disturb impermeable wall theory, but it masks whatever

effective roughness may be caused by discrete pore injection .

.I 339

.T

experimental evaluation of heat transfer with transpiration

cooling in a turbulent boundary layer at m=3 .2.

.A

bartle,e.r. and leadon,b.h.

.B

j. ae. scs. 27, 1960.

.W

experimental evaluation of heat transfer with transpiration

cooling in a turbulent boundary layer at m=3 .2.

it is found that for prescribed velocity field, electrical field and conductivity, the current can be calculated by integration . work is related to analytic investigation of the boundary layer in a physically reasonable accelerator .

.I 340

.T

analysis of effects of diffusion of a foreign gas into the laminar boundary layer of a supersonic flow of air in a tube .

.A

radbill,j.r. and kaye,j.

.B

j.ae.scs. 26, 1959.

.W

analysis of effects of diffusion of a foreign gas into the laminar boundary layer of a supersonic flow of air in a tube .

adiabatic wall temperatures and recovery factors are calculated for pipe flows with an entrance mach number of 5 and with uniform injection of helium . predicted values of the recovery factor increase slowly with increasing injection rate and with increasing distance from the tube entrance .

.I 341

.T

the analytical design of an axially symmetric laval nozzle for a parallel and uniform jet .

.A

foelsch,k.

.B

j. ae. scs. 16, 1949.

.W

the analytical design of an axially symmetric laval

nozzle for a parallel and uniform jet .

the equations for the nozzle's contours are derived by integration of the characteristic equations of the axially symmetric flow . since it is not possible to integrate these equations mathematically in an exact form, it was necessary to find a way to approximate the calculations . the approximation offers itself by considering and comparing the conditions of the flow in a cone with those in a nozzle, as a linearization of the characteristic equations .

the first part of the report deals with equations for the transition curve by which the conical source flow is converted into a parallel stream of uniform velocity . the equations are derived by integration along a mach line of the flow in the region where the conversion takes place . a factor f is introduced expressing a relation between the direction and the velocity of the flow along a certain mach line . f remains undetermined and is not involved in the final equations .

in the second part of the report, the spherical sonic flow section is converted into a plane circular section of the throat . the nozzle's contour adjacent to the throat is formed by the arc of a circle connected with the transition curve by a straight line . the gas dynamic properties of the boundary mach line are calculated in table 1, the use of which shortens the calculations considerably .

.I 342

.T

effect of diffusion fields on the laminar boundary

layer .

.A

smith,j.w.

.B

j.ze.scs. 21, 1954.

.W

effect of diffusion fields on the laminar boundary

layer .

a theory is developed which describes the effect of a general

diffusion field on the dynamic and thermal characteristics of a

laminar boundary layer on a flat plate in steady compressible

flow . fluid properties are considered as functions of

temperature and local concentration of the foreign gas . the diffusion

field is described by a differential equation that relates

convective and diffusion transfer and which considers diffusion currents

arising from gradients of concentration and temperature . by

means of the usual transformations the system is reduced to a set

of ordinary differential equations, which in turn are transformed

into a set of integral equations . the latter is amenable to

solution by the method of successive approximations .

the theory and results have bearing on the problem of control

and reduction of aerodynamic heating at hypersonic speeds .

the special feature of this approach lies in the utilization of diffusion fields for the purpose of reducing the detrimental effects

of viscous dissipation . although the theory is adapted to a

fuller investigation of this problem, the numerical examples

considered involve mainly diffusion fields of helium, with which

good results have been achieved at mach numbers 8 and 12 .

whereas at the higher mach number the influx of heat was

practically eliminated, a reversal in the direction of heat flow has

been effected at the lower mach number .

.I 343

.T

transpiration cooling experiments in a turbulent boundary

layer at m=3 .

.A

leadon,b.m. and scott,c.j.

.B

j. ae. scs. 23, 1956.

.W

transpiration cooling experiments in a turbulent boundary

layer at m=3 .

turbulent recovery factor and heat-transfer measurements have been

made on a porous flat wall section at a nominal mach number of 3.0 and a

reynolds number of approximately 4 x 10 using both air and helium as the

transpired gas . measured heat-transfer

coefficients correlate well with the

compressible theory of rubesin for air

and qualitatively with simple film

theory for either coolant, indicating that the heat transfer from a turbulent

boundary layer can be reduced by transpiration

cooling to well below that of

the uncooled boundary layer at the same reynolds number .

.I 344

.T

some experimental techniques in mass transfer cooling .

.A

leadon,b.m.

.B

aero/space eng. 18, 1959.

.W

some experimental techniques in mass transfer cooling .

author introduces his survey by a brief review of the history of investigations dealing with boundary layers on impermeable solid surfaces, and notes that no true theory exists for turbulent boundary layers, the success of studies in this area having been due to the introduction of artificial, if ingenious, assumptions which permitted empirical correlations fd data . the terminology introduced by the author for distinguishing the different situations involving mass transfer from the wall to the stream may give rise to some objections . for instance, /film cooling/ need not refer only to the injection of a liquid, since applications involving gas film cooling exist . also, his restriction of the term /transpiration cooling/ to refer to the injection through a porous surface of a gas only of the same composition as the exterior stream does not enjoy universal usage . the influence of mass transfer on heat transfer through laminar boundary layers and on the transition from laminar to turbulent flow is described, with consideration given to the question of the net effect of the stabilizing influence of surface cooling and the destabilizing influence of injection .

reviewer suggests that author's inaccurate statement to the effect that /thus far the higher energy conditions do not threaten to involve turbulent injection, so turbulent boundary-layer research enjoys a fairly academic serenity broken only by its own frustrations/ be excused on grounds of poetic license, although it ignores the efforts being devoted to the pressing practical problems of erosive burning of solid propellants (possibly the most common example of a complete /aerothermochemical/ problem involving distributed surface heat and mass transfer with chemical reaction in a flow system) and of effusion cooling of rocket nozzles, both of which involve turbulent boundary-layer conditions . author emphasizes the tedious experimental problems involved in research on boundary layers with blowing, and notes the desirability of velocity distribution measurements, especially in turbulent injection layers . the observation that no good data on concentration profiles in the case of the diffusion boundary layer have been published may be an overstatement, since author's bibliography overlooks the work of j. berger (/contribution a l'etude de l'injection parietale,/ doctor's thesis, university of paris, memorial des poudres 38 (annex), p. 1,. paris, imprimerie nationale, 1956) .

.I 345

.T

the interaction of shock waves with boundary layer on a flat surface .

.A

barry,f.w., shapiro,a.h. and neumann,e.p.

.B

j. ae. scs. 18, 1951, 229.

.W

the interaction of shock waves with boundary layer

on a flat surface .

the development of supersonic compressors, supersonic

diffusers, and high-speed aircraft points to the increasing

importance of the interaction between shock waves and boundary

layers .

the experimental work reported here is intended to (1)

provide a better understanding of the nature of the shock

boundary-layer interaction, (2) serve as a guide and stimulus to theoretical

work, and (3) develop an empirical method for predicting the

effects of the interaction .

experiments were performed on the reflection of an oblique

shock from a boundary layer on a flat surface at a mach number

of 2.05 . the effects of shock strength and boundary-layer regime

were explored .

the results are in the form of schlieren photographs,

constant-density contours found from interferometer photographs, and

static pressure distributions at the plate surface .

.I 346

.T

measurements of turbulent friction on a smooth flat plate in supersonic .

.A

.B

.W

measurements of turbulent friction on a smooth flat plate in supersonic .

direct measurements of supersonic local skin friction, using the floating-element technique, are presented for mach numbers from bulent flow and transition are emphasized, although some measurements in the laminar regime are included . the observed effect of compressibility is to reduce the magnitude of turbulent skin friction by a factor of two at a mach number of 4.5 and a reynolds number of about 10 .

the boundary-layer momentum-integral equation for constant pressure is verified within a few per cent by two experimental methods . typical static pressure measurements are presented to show that transition can be detected by observing disturbances in pressure associated with changes in displacement thickness of the boundary layer .

it is found that the turbulent boundary layer cannot be defined experimentally for values of less than about 2,000, where is the momentum thickness . for larger values of there is a unique relationship between local friction coefficient and momentum-thickness reynolds number at a fixed mach number . the appendix compares the present measurements at m = 2.5 with experimental data from other sources .

.I 347

.T

boundary layer measurements in hypersonic flow .

.A

hill,f.k.

.B

j. ae. scs. 21, 1954, 433.

.W

boundary layer measurements in hypersonic flow .

experimental data are presented on boundary-layer formation,

heat transfer, and skin-friction coefficient at mach numbers of

the wall of a conical nozzle in the presence of a favorable pressure

gradient and several rates of heat transfer . the reynolds

number based on momentum thickness varied from 1,500 to 3,500 .

comparison is made with data at lower mach numbers and with

the semiempirical theory of von karman . the existing data up

to mach numbers of nine indicate agreement to within 5 per

cent when compared with a form of the wilson theory, but it is

clear that the effects of heat transfer and pressure gradients

present problems which require extensive study and experiment in

the future .

.I 348

.T

turbulent boundary layer in compressible fluids .

.A

van driest,e.r.

.B

j.ae.scs. 18, 1951, 145.

.W

turbulent boundary layer in compressible fluids .

the continuity, momentum, and energy differential equations
for turbulent flow of a compressible fluid are derived, and the
apparent turbulent stresses and dissipation function are
identified . a general formula for skin friction, including heat
transfer to a flat plate, is developed for a thin turbulent boundary
layer in compressible fluids with zero pressure gradient . curves
are presented giving skin-friction coefficients and heat-transfer
coefficients for air for various wall-to-free-stream temperature
ratios and free-stream mach numbers .

in the special case when the boundary layer is insulated, this
general formula yields skin-friction coefficients higher than those
given by the von karman wall-property compressible-fluid
formula but lower than those given by the von karman
incompressible-fluid formula . heat transfer from the boundary layer
to the plate generally increases the friction and heat-transfer
coefficients .

.I 349

.T

numerical solution of the boundary layer equations
without similarity assumptions .

.A

kramer,r.f. and lieberstein,h.

.B

j. ae. scs. 26, 1959, 508.

.W

numerical solution of the boundary layer equations
without similarity assumptions .

the crocco transformation combined with a mangler transformation is used to carry the boundary-layer problem for axially symmetric blunt bodies into a form suitable for direct numerical computation without introduction of similarity assumptions . conditions which in the original problem appear at infinity now are brought to a finite straight line, and the body is transformed to a parallel line . data can be generated on the stagnation line the equations are a parabolic system of two second-order equations, the boundary-value problem is analogous to the slab problem for the heat equation . an implicit difference equation is used to reduce stability difficulties . special techniques in forming the difference equation result in a linear system of algebraic equations to be solved on any given line of integration, and these solutions are computed from recursion relations generated by back substitution . for bluntnosed bodies with approach flow mach numbers greater than 8 (approximately), large temperature gradients occur across a thin boundary layer of dissociated gas, and it is necessary to use real-gas effects, approximated here by certain fits to the gas tables . a case is computed, however, for a lower mach number approach flow using perfect-gas theory to provide a standard against which similarity solutions may be tested .

.I 350

.T

laminar jet mixing of two compressible fluids with heat release .

.A

s. i. pai

.B

university of maryland

.W

laminar jet mixing of two compressible fluids with heat release .

the laminar jet mixing problems with heat release have been formulated .

a general discussion of the solution of these problems is also given .

the important parameters of these problems are brought out . some

specific cases of the jet mixing problem, such as jet mixing of one

compressible fluid, isothermal jet mixing of two compressible fluids,

and isovel jet mixing of two compressible fluids with heat release, are

discussed in detail.

.I 351

.T

thermal distributions in jeffrey-hamel flows between nonparallel plane

walls .

.A

millsaps, k. and pohlhausen, k.

.B

j. aero. sc. v. 20. march 1953, pp 187-196 .

.W

thermal distributions in jeffrey-hamel flows between nonparallel plane

walls .

the authors give the exact solution for the thermal distributions for

the steady laminar flow of a viscous incompressible fluid between

non-parallel plane walls held at a constant temperature . the velocity

profiles are determined with the aid of jacobian elliptic functions by

using the jeffery-hamel solution of the hydrodynamic problem . it is shown that in this special case the energy equation giving the

temperature profiles can be reduced to an ordinary linear differential

equation with variable coefficients . after the introduction of

dimensionless parameters, numerical solutions are given for diverging and

converging channels with total openings of 10degree for the possible

combinations of three reynolds numbers and five prandtl numbers .

.I 352

.T

on heat transfer over a sweat-cooled surface in laminar

compressible flow with a pressure gradient .

.A

murduchow,m.

.B

j. ae. scs. 19, 1952, 705.

.W

on heat transfer over a sweat-cooled surface in laminar

compressible flow with a pressure gradient .

a simple expression is derived for the normal injection velocity

distribution theoretically required to maintain a given uniform

temperature along a porous surface in the laminar

boundary-layer region of a compressible flow with a given velocity

distribution outside of the boundary layer . this expression is valid for

any given free-stream mach number but is based on a prandtl

number of unity and on the assumption that the viscosity

coefficient varies linearly with the temperature . by using the

dorodnitsyn type of transformation, the variation of fluid

properties even in the case of zero mach number is taken into account . this study is of particular practical interest in

connection with the sweat-cooling of turbine blades and of airfoil surfaces

in high speed flow . the method of analysis consists of applying

the karman-pohlhausen method to both the momentum and

energy boundary-layer equations and of using an additional heat

balance equation, involving the coolant temperature . a

closed-form approximate solution of the equations is then derived .

numerical examples for flow in the immediate vicinity of a

stagnation point and for a typical type of flow over a turbine blade are

given .

.I 353

.T

the effect of helium injection at an axially symmetric stagnation

point .

.A

h. hoshizaki and h. j. smith

.B

missile and space division, lockheed aircraft corporation,

sunnyvale, calif.

.W

the effect of helium injection at an axially symmetric stagnation

point .

an effective means of protecting the surface of a hypersonic re-entry

vehicle is to inject small quantities of a lightweight gas into the

boundary layer through a porous wall . this process, which is known

as mass-transfer cooling, protects the surface in two ways . first

of all, as the injected gas or coolant passes from the reservoir through the wall to the surface, a considerable quantity of heat is absorbed as its temperature is raised from the reservoir temperature to the wall surface temperature . characteristically, lightweight gases have relatively high specific heats .

secondly, the transfer of mass and enthalpy by convection and diffusion normal to the surface alters the characteristics of the boundary layer in such a manner as to reduce the temperature gradient at the wall, and, hence, the conductive heat transfer at the wall . this is sometimes referred to as the blowing effect .

.I 354

.T

laminar heat-transfer and pressure measurements over blunt-nosed cones at large angle of attack .

.A

victor zakkay

.B

research associate, polytechnic institute of brooklyn, freeport, n.y.

.W

laminar heat-transfer and pressure measurements over blunt-nosed cones at large angle of attack .

tests have been conducted at a mach number of 6, in the pibal hypersonic facility, in order to determine the heat-transfer and pressure distributions over a slender blunted cone at angles of attack of erature ratio, stagnation to wall, was approximately 2.3 . the model tested has a sperical nose diameter of 1.0 in., a base diameter of 3.75 in., and a cone half-angle of 20 degrees . the measurements were made at 5 peripheral stations on the model .

in this note the experimental results at a 15 degree angle of attack

are presented . a more detailed analysis of the results for all

angles of attack is presented in reference 1 .

.I 355

.T

the injection of air into the dissociated hypersonic laminar boundary

layer .

.A

sinclaire m. scala

.B

research engineer

missile and ordnance systems department, general electric company,

philadelphia, pa.

.W

the injection of air into the dissociated hypersonic laminar boundary

layer .

in first approximation, dissociated air may be treated as a binary

mixture of air atoms and air molecules . in order to include the

effects of mass transfer into the boundary layer, it becomes necessary

to introduce a third chemical species and hence a second diffusion

equation . we have avoided this complexity by considering the injection

of air molecules into the boundary layer, and hence the theoretical

treatment is accomplished within the framework of a binary mixture gas .

.I 356

.T

on optimum nose curves for missiles in the super-aerodynamic

regime .

.A

tan,h.s.

.B

j. ae. scs. 25, 1958, 56.

.W

on optimum nose curves for missiles in the super-aerodynamic

regime .

author shows that the differential equations defining the

minimum drag body shapes for free molecule flow that were developed

and numerically integrated by w. j. carter (amr 11 (1958), rev.

realized, however, that numerical or analytical integration of the

second-order differential equation is unnecessary since, for the

flow conditions considered, the first integral to the euler equation

can be written prior to the substitution of the expression defining

the pressure coefficient .

.I 357

.T

optimum nose shapes for missiles in the super-aerodynamic

region .

.A

carter,w.j.

.B

j. ae. scs. 24, 1957, 527.

.W

optimum nose shapes for missiles in the super-aerodynamic

region .

the mechanics of the kinetic theory of gases is employed to describe the drag force on the nose of a missile moving in the super-aerodynamic region of the atmosphere . three separate cases are considered--ideal specular reflection, specular-type reflection from a slightly rough surface, and surface absorption followed by random emission of the striking molecules . the calculus of variations is employed to obtain the differential equation of the nose shape which minimizes the drag force for each of the three cases . the resulting differential equations are then solved by a numerical procedure . the drag coefficients for the optimum nose shapes are likewise determined and these are compared with the drag coefficients given by other nose shapes . it is further shown that the drag coefficients arising when specular-type reflections occur are significantly dependent on the nose shape . when surface absorption followed by random emission occurs, the drag coefficient is not strongly dependent on either the missile nose shape or the fineness ratio of the nose .

.I 358

.T

on the model of the free shock separation, turbulent boundary layer .

.A

mager,a.

.B

j. ae. scs. 1956, 181.

.W

on the model of the free shock separation, turbulent boundary layer .

by free shock-separated boundary layers, one means that type of separation where the flow downstream of the separation region is free to adjust to any direction that may result from the shock-boundary-layer interaction process . a detailed model of the free shock-separated turbulent boundary layer is postulated, and the pressure rise following from this model is estimated and compared with experiments . the results are applied to the prediction of separation in an overexpanded nozzle .

.I 359

.T

note on the hypersonic similarity law for an unyawed

cone .

.A

lees,l.

.B

j. ae. scs. 18, 1951.

.W

note on the hypersonic similarity law for an unyawed

cone .

it is now known that the hypersonic similarity law derived for slender cones and ogival bodies under the assumption, is applicable for mach numbers as low as 3 . this note makes use of a series development to infer the hypersonic similarity law for unyawed cones from the taylor-maccoll differential equations and associated boundary conditions . a simple approximate formula for the function of the similarity law is obtained,

and the drag function computed with this formula is compared with kopal's numerical results and, for very slender cones, with

von karman's linearized formula .

.I 360

.T

lift on inclined bodies of revolution in hypersonic flow .

.A

grimminger, g., williams, e. p., and young, g.

.B

j. aero. sc. v. 17, november, 1950 .

.W

lift on inclined bodies of revolution in hypersonic flow .

the importance of body lift lies in the fact that at moderate angles of

attack and high mach number it can constitute an appreciable part of

the total lift of a winged missile . in this paper an attempt has been

made to analyze body lift in hypersonic flow by an approximate method

and, together with a correlation of existing experimental data, to

indicate the probable variation of body lift over a wide range of mach

numbers extending from low supersonic to hypersonic . the method of

analysis of hypersonic flow over inclined bodies of revolution employed

herein has been denoted as the hypersonic approximation . it is an

improvement on the newtonian corpuscular theory of aerodynamics, since it

considers the centrifugal forces resulting from the curved paths of the

air particles in addition to the impact /newtonian/ forces .

.I 361

.T

the flow of a viscous liquid past a flat plate at small

reynolds number .

.A

tomotika,s. and yosinobu,h.

.B

j. math.phys. 35, 1956.

.W

the flow of a viscous liquid past a flat plate at small

reynolds number .

the authors repeat the earlier calculations of piercy

and winny (proc. roy. soc. london. ser. a. 140 (1933),

earlier works were known to be different from each other .

the careful analysis of the present authors shows that the

skin-friction coefficient up to the second approximation

agrees perfectly with that of piercy and winny .

.I 362

.T

three-dimensional effect of flutter in a real fluid .

.A

wen-hwa chu

.B

senior research engineer, department of mechanical sciences,

southwest research institute, 8500 culebra road, san antonio 6, texas

.W

three-dimensional effect of flutter in a real fluid .

in ref. 1, an alternative semi-empirical formulation for flutter in a

real fluid is given . for more accurate determination of the empirical

coefficients, the three-dimensional effect of finite span should be

taken into account . following reissner's approximation for large-aspect-ratio rectangular wings, the boundary-value problem governing the downwash w and the vorticity distribution .

.I 363

.T

an alternative formulation of the problem of flutter in real fluids .

.A

chu,w.h. and abramson,h.n.

.B

j. ae. scs. 26, 1959.

.W

an alternative formulation of the problem of flutter in real fluids .

it is well known, in steady flow, that the actual lift curve slope is somewhat less than that predicted by inviscid flow theory, even at small angles of attack . as the stall angle is approached, the lift curve slope continually decreases and thus deviates even more from the theoretical value . pinkerton employed the measured circulation to determine the pressure distribution and found that the resulting prediction of the moment is considerably improved over that given by the classical theory . this amounts to replacing the conventional kutta-joukowski condition with the condition that the total lift should agree with the measured value, and this, in turn, completely determines the flow pattern . practically, this is accomplished by giving a fictitious camber to the profile . since potential flow theory is valid outside of the boundary layer, once the boundary-layer

thickness is known, the potential flow may be corrected for the
displacement thickness and the viscous wake by appropriate
source distributions . the boundary layer cannot be evaluated,
of course, until the potential flow is known and the circulation is
applied . a criterion to determine the circulation, by
generalizing the kutta-joukowski condition, was proposed by preston
and spence by assuming that the pressure at the trailing edge
shall have the same value when determined from the
potential-flow values above and below the airfoil . this procedure gives
qualitative information concerning viscous effects in steady
flow .

.I 364

.T

a method for analysing the insulating properties of
the laminar compressible boundary layer .

.A

libby,pa. and pallone,a.

.B

j. ae. scs. 21, 1954.

.W

a method for analysing the insulating properties of
the laminar compressible boundary layer .

in some cooling problems associated with high energy flows
it may be convenient to localize strongly the cooling, as for
example by injecting a coolant through an upstream porous strip,
and to depend on the insulating properties of the boundary layer
to reduce, or to eliminate completely the need for further cooling

on the surface downstream of the highly cooled section . this upstream cooling technique may be of interest in connection with optical windows in hypersonic wind tunnels, and on radomes, wings, and bodies of high-speed aircraft and missiles .

in this paper a method for investigating the insulating properties of a laminar compressible boundary layer on a two-dimensional surface with zero heat transfer is presented . the physical situation considered thus corresponds to the case in which the heat transfer downstream of the strongly cooled section is completely eliminated . of practical concern is how the temperature of the uncooled surface varies in the downstream direction from its low initial value and thus how the low energy layer established by the upstream cooling insulates the downstream surface .

the karman integral method extended to both the momentum and energy partial differential equations of the boundary layer has been used . the station, at which cooling and or injection ceases, corresponds to a discontinuity in boundary conditions and thus in solutions . at this point the flux of mass, momentum, and energy within the boundary layer has been made continuous by the introduction of three additional parameters in the velocity and stagnation enthalpy profiles . thus the velocity and stagnation enthalpy profiles have both been taken as sixth degree polynomials . the resulting two integral-differential equations are then solved for two unknown functions of the distance along the wall . these two functions are related to the boundary-layer thickness and to the wall temperature . initial conditions corresponding to a given initial wall temperature and an initial

boundary-layer thickness are prescribed . exact closed-form solutions

for the case of zero axial pressure gradient are obtained . for

flows with significant pressure gradients, numerical solutions

are required in general . several numerical examples of practical

interest are presented .

.I 365

.T

the homogeneous boundary layer at an axisymmetric stagnation

point with large rates of injection .

.A

libby,p.a.

.B

j. ae. scs. 29, 1962.

.W

the homogeneous boundary layer at an axisymmetric stagnation

point with large rates of injection .

this report presents a theoretical analysis of the boundary

layer at an axisymmetric stagnation point with large rates of

air injection . the results of a previous investigation indicated

that for localized mass transfer in the stagnation region, the

rates of injection are considerably greater than those usually

treated . the exact stagnation-point boundary-layer equations

are integrated numerically for an approximate representation

of the gas properties . the two-point boundary conditions are

treated in a new manner which is useful for various

boundary-layer and mixing problems . the exact solutions indicate that for large rates of injection the boundary layer is closely

represented by an inner isothermal shear flow and by and exterior,

relatively thin region, in which the flow variables change to their

free-stream values . an integral method based on profiles

suggested by the exact solutions is developed and shown to lead

to accurate predictions of the integral thicknesses which are

of interest for a study of the downstream influence of the

stagnation-point mass transfer .

.I 366

.T

helium injection into the boundary layer at an axisymmetric

stagnation point .

.A

fox,h. and libby,p.a.

.B

j. ae. scs. 1962, 921.

.W

helium injection into the boundary layer at an axisymmetric

stagnation point .

this report presents a theoretical analysis of the boundary

layer at an axisymmetric stagnation point with large rates of

helium injection . the exact stagnation-point boundary-layer

equations are integrated numerically with approximate

representations of the gas properties . the treatment of the

two-point boundary-value problem employed herein is shown to be

useful for various boundary-layer and mixing problems . the

exact solutions indicate that for large rates of injection the boundary layer can be represented by a thick, inner layer of constant shear, temperature, and composition and by a relatively thin outer region in which the flow variables adjust to their free-stream values . an inviscid-flow model is shown to lead to accurate predictions of this shear layer and will thus provide sufficiently accurate profiles for use in the study of the downstream influence of stagnation-point mass transfer . the heat transfer to the stagnation point is also considered . tabulations of the eigenvalues for a variety of wall conditions and injection rates are given .

.I 367

.T

control system and analysis and design via the second method of lyapunov .

.A

kalman, r. e. and bertram, j. e.

.B

trans. asme. series d, v. 82, june 1960 .

.W

control system and analysis and design via the second method of lyapunov .

the/second method/of lyapunov is the most general approach currently in the theory of stability of dynamic systems . after a rigorous exposition of the fundamental concepts of this theory, applications are made to/a/ stability of linear stationary, linear nonslationary, and nonlinear systems,./b/estimation of transient behavior,./c/control-system optimization,./d/design of relay servos . the discussion is essentially self-contained, with emphasis on the thorough development of the principal ideas and mathematical tools . only systems governed by differential equations are treated here . systems governed by difference equations are the subject of a companion paper .

.I 368

.T

some problems of polar missile control .

.A

best,d.

.B

j.roy.ae.soc., 64, 1960.

.W

some problems of polar missile control .

a polar-controlled missile is one in which manoeuvre is carried out by rotations about roll and pitch axes, that is, in the manner of a conventional aeroplane . this paper discusses some problems in the application of this form of control to homing missiles .

in comparison with the alternative cartesian configuration, this method presents some special design problems . in the former case, it is often possible to resolve the motion into two planes and consider the pitch and yaw control systems as independent two-dimensional problems . this simplification is not possible in the case of polar control and it is usually necessary to consider the whole three-dimensional system . the equations of motion which result are, in general, not susceptible to analysis . because of this,

the design of control systems requires extensive use of

simulators .

.I 369

.T

an approximate solution of the supersonic blunt body

problem for prescribed arbitrary axisymmetric shapes .

.A

traugott,s.

.B

j. ae. scs. 27, 1960, 361.

.W

an approximate solution of the supersonic blunt body

problem for prescribed arbitrary axisymmetric shapes .

the integral method of belotserkovskii has been carried out to

the first approximation for arbitrary blunt axisymmetric bodies

in supersonic or hypersonic flight . this method is direct, in

that it gives the surface-pressure distribution and shock shape for

a prescribed body . results obtained by numerical integration

for several body shapes at several mach numbers are compared

to experimental results with good agreement . it is also shown

that the method can be successfully applied to pointed bodies

with attached shock . in the stagnation region, simple

relationships are found from the equations of the first approximation

which connect the surface-velocity gradient, shock curvature,

shock-detachment distance, and body curvature . these

relations are also correlated with experiment for a variety of shapes as a function of mach number . the correlations permit a rapid estimate of the stagnation-point velocity gradient, important for heat-transfer calculations, for any blunt body from the shock stand-off distance . a method for a higher approximation is described, for which, in contrast to the higher approximations of belotserkovskii, a large number of simultaneous total differential equations with unknown parameters does not occur . one form of this method has been studied numerically . results are given which, though only partially successful, indicate the amount of improvement to be expected from a higher approximation .

.I 370

.T

theoretical pressure distribution on a hemisphere-cylinder combination .

.A

anthony casaccio

.B

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.W

theoretical pressure distribution on a hemisphere-cylinder combination .

in recent years great use has been made of approximate methods for the determination of the pressure distribution on blunt-nosed bodies and afterbodies at high mach numbers . for quasi-spherical bodies it has been suggested that modified newtonian theory in combination with a prandtl-meyer expansion be used on the nose portion, the two laws being matched at the point where the pressure gradients are equal .

no simple approximation, however, has been found for flat-nosed bodies .

as for the pressure distribution on the afterbody, the blast-wave

analogy has been suggested for general nose shapes but particular

afterbody profiles .

the purpose of the present note is to compare these approximate

estimates with a more accurate determination of the flow field about

a hemisphere-cylinder in an ideal gas flow . it was felt that since

experimental investigations in air at this mach number are scarce

and very difficult to obtain, the comparison would be of interest . the

basis of comparison is the flow field as it results from a numerical

integration of the exact equations governing the motion of the ideal

fluid .

.I 371

.T

note on tip-bluntness effects in the supersonic and

hypersonic regimes .

.A

bennett,f.d.

.B

j. ae. scs. 24, 1957, 314.

.W

note on tip-bluntness effects in the supersonic and

hypersonic regimes .

in a recent letter, m. h. bertram presents some data on

flows at m = 6.85 around 10 half-angle cones with blunted

tips . since the demarcation between the supersonic and hypersonic regimes is not sharp and since one expects hypersonic flows

to be generally similar to those at lower mach numbers--

especially where viscous effects do not predominate throughout the

entire field of interest--it is of some value to compare bertram's

results with those obtained by giese and bergdolt for 15

half-angle cones at m = 2.45 . following the observation by charters

and stein that drag coefficient measurements on blunted cones

imply a reynolds number effect, giese and bergdolt study the

convergence to conical flow of the perturbed flow about a cone

with truncated tip . they employ the mach-zehnder

interferometer and the conical flow criterion as analytical tools .

.I 372

.T

an experimental investigation of flow about simple

blunt bodies at a nominal mach number of 5. 8.

.A

oliver,r.e.

.B

j. ae. scs.23, 1956, 177.

.W

an experimental investigation of flow about simple

blunt bodies at a nominal mach number of 5. 8.

an experimental investigation was

conducted in the galcit hypersonic

wind tunnel to determine flow characteristics

for a series of blunt bodies at a

nominal mach number of 5.8 and free-stream reynolds numbers per in. of

measured values for the pressure

coefficient distributions are compared with

a modified newtonian

expression . the agreement is very good for the

three-dimensional bodies and is

fair for the circular cylinder transverse to the

free-stream flow direction . a

complete report of the investigation is given in

a galcit hypersonic wind

tunnel memorandum .

.I 373

.T

the generalized expansion method and its application

to bodies travelling at high supersonic airspeeds .

.A

eggers,a.j., savin,r.c. and syvertson,c.a.

.B

j.ae.scs. 22, 1955, 231.

.W

the generalized expansion method and its application

to bodies travelling at high supersonic airspeeds .

it is demonstrated that the shock-expansion method can be

generalized to treat a large class of hypersonic flows, only one of

which is flow about airfoils . this generalized method predicts

the whole flow field, including shock-wave curvatures and

resulting vorticity, providing that (1) disturbances originating on the surface of an object are largely absorbed in shock waves with which they interact and (2) disturbances associated with the divergence of stream lines in tangent planes to the surface are of secondary importance compared to those associated with the curvature of stream lines in planes normal to the surface . it is shown that these conditions may be met in three-dimensional as well as two-dimensional hypersonic flows . when they are met, surface streamlines may be taken as geodesics, which, in turn, may be related to the geometry of the surface .

the validity of the generalized shock-expansion method for three-dimensional hypersonic flows is checked by comparing predictions of theory with experiment for the surface pressures and bow shock waves of bodies of revolution . the bodies treated are two ogives having fineness ratios of 3 and 5 . tests were conducted at mach numbers from 2.7 to 6.3 and angles of attack up to 15 degrees in the 10- by 14-in. supersonic wind tunnel of the ames aeronautical laboratory . at the lower angles of attack, theory and experiment approach agreement when the ratio of mach number to fineness ratio--that is, the hypersonic similarity parameter--exceeds 1 . at the larger angles of attack, theory tends to break down, as would be expected, on the leeward sides of the bodies .

as a final point, it is inquired if the two-dimensionality of inviscid hypersonic flows has any counterpart in hypersonic boundary-layer flows . the question is answered in the affirmative, and results of experiment are employed to provide a partial check of this conclusion .

.I 374

.T

an investigation of optimum zoom climb techniques .

.A

kelly,h.j.

.B

j. ae.scs. 26, 1959, 794.

.W

an investigation of optimum zoom climb techniques .

the problem of optimal zoom climb maneuvering of a turbojet aircraft has been investigated using the mayer formulation of the calculus of variations . the euler-lagrange equations governing optimum symmetric flight have been integrated numerically by digital computation .

discontinuities in thrust arising from turbojet afterburner blowout have been treated, and conditions which must be satisfied across the interface generated by the discontinuity have been derived .

arbitrary control techniques have been compared with the optimum, and it has been found that performance is relatively insensitive to piloting technique unless a time limitation is imposed which requires high maneuvering load factors .

.I 375

.T

steady flow in the laminar boundary layer of a gas .

.A

illingworth,c.r.

.B

proc. roy. soc. a, 199, 1949.

.W

steady flow in the laminar boundary layer of a gas .

if the boundary-layer equations for a gas are

transformed by mises's transformation, as

was done by karman   tsion for the flow

along a flat plate of a gas with unit prandtl

number, the computation of solutions is

simplified, and use may be made of previously

computed solutions for an incompressible fluid .

for any value of the prandtl number, and

any variation of the viscosity with the temperature

t, after the method has been applied

to flow along a flat plate (a problem otherwise treated

by crocco), the flow near the forward

stagnation point of a cylinder is calculated with

dissipation neglected, both with the effect of

gravity on the flow neglected and with this effect

retained for vertical flow past a horizontal

cylinder . the approximations involved by the neglect

of gravity are considered generally,

and the cross-drift is calculated when a horizontal

stream flows past a vertical surface .

when, and the boundary is heat-insulated,

it is shown that the boundary-layer

equations for a gas may be made identical, whatever be the main stream, with the

boundary-layer equations for an incompressible fluid with a certain,

determinable, main stream . the

method is also applied to free convection at a flat plate

variation with altitude of the state of the surrounding

fluid neglected) and to laminar flow in

plane wakes, but for plane jets the conditions,

previously imposed by howarth,

are also imposed here in order to obtain simple solutions .

.I 376

.T

transformation between compressible and incompressible

boundary layer equations .

.A

van le,n.

.B

j. ae. scs. 20, 1953.

.W

transformation between compressible and incompressible

boundary layer equations .

it is proposed to show that the boundary-layer equation of

compressible flow can be reduced to that of incompressible

flow . such work was initiated by stewartson and by rott and

crabtree . in the following some of the restrictions imposed by

references 1 and 2 will be removed, and it will be shown that the

transformation from compressible boundary layer to

incompressible boundary layer can be applied to the laminar, as well as turbulent, case . a direct method will be used for this purpose .

.I 377

.T

a turbulent analog of the stewartson-illingworth transformation .

.A

culick,f.e. and hill,j.

.B

j. ae. scs. 25, 1958.

.W

a turbulent analog of the stewartson-illingworth transformation .

the stewartson-illingworth transformation is applied to the integral momentum equation for compressible boundary-layer flow, leaving the x-coordinate transformation unspecified, however . it is shown that the transformed equation is the integral momentum equation for incompressible flow if (a) the effect of compressibility on the boundary-layer shape parameter h can be represented by
and (b) the x-coordinate transformation is chosen to be suitably related to the ratio of skin-friction coefficients in compressible and incompressible flows .

experimental evidence is presented which shows that condition (a) is satisfied for turbulent boundary layers up to m = 5 . an x-transformation is chosen according to (b) and an equation is presented which gives the turbulent boundary-layer growth in compressible flow in terms of a simple quadrature . the predictions of this equation are then compared with some measurements on wind-tunnel nozzles .

.I 378

.T

engineering relations for friction and heat transfer to surfaces in high

velocity flow .

.A

eckert, e.

.B

j. aero. sc. v.22, august 1955, pp 585-587

.W

engineering relations for friction and heat transfer to surfaces in high

velocity flow .

in calculations of thermodynamic heating for high speed missiles

parameters have been used based on relationships which hold for

constant-property fluids . the validity of this procedure has been verified

recently in a survey of heat transfer in which a relationship for the

reference temperature was developed . a calculation procedure for

laminar and turbulent boundary layers, based on this relationship, is

given .

.I 379

.T

reverse flow and variational theorems for lifting surfaces

in nonstationary compressible flow .

.A

flax,a.h.

.B

j. ae. scs. 20, 1953.

.W

reverse flow and variational theorems for lifting surfaces

in nonstationary compressible flow .

a reverse-flow theorem for compressible nonsteady flow, valid within the limits of linearized theory, is derived . this theorem gives a general class of relations between linearized solutions for lifting surfaces in direct and reverse flow . based on the same considerations used to establish the theorem, an adjoint variational principle, which may be useful in approximate solutions of non-steady lifting surface problems, is obtained . to illustrate the uses of the reverse-flow theorem, it is applied to the determination of relations between aerodynamic coefficients in direct and reverse flow and to the obtaining of influence functions for total lift, pitching moment, and rolling moment for a wing oscillating with arbitrary motion and surface deformation, in terms of the pressure distributions for simpler cases in reverse flow .

.I 380

.T

effect of quasi-steady air forces on incompressible

bending-torsion flutter .

.A

dugundi,j.

.B

j. ae. scs. 1958.

.W

effect of quasi-steady air forces on incompressible

bending-torsion flutter .

explicit solutions are obtained for the bending-torsion flutter

of a two-dimensional airfoil in incompressible flow under the
assumptions that the theodorsen function, c(k) is set equal to a
real constant, and the diagonal virtual mass terms are negligible .
for the case of small bending to torsion frequency ratio, a
comparison is made of these quasi-steady solutions with an
earlier empirical expression suggested by theodorsen and garrick
for the nonsteady case, and the effect of the c(k) function is
indicated . the importance of the c.g. location for these small
cases is re-emphasized, and the possibility of flutter at zero
air speed is indicated .

.I 381

.T

the axisymmetric boundary layer on a long thin cylinder .

.A

glauert,m.b. and lighthill,m.j.

.B

proc. roy. soc. a, 230, 1955, 188.

.W

the axisymmetric boundary layer on a long thin cylinder .
the laminar boundary layer in axial flow about
a long thin cylinder is investigated by two
methods . one (2) is a pohlhausen method,
based on a velocity profile chosen to represent
conditions near the surface as accurately as
possible . the other (3) is an asymptotic series
solution, valid far enough downstream from
the nose for the boundary-layer thickness to

have become large compared with the cylinder radius . another series solution (due to seban, bond and kelly) is known, valid near enough to the nose for the boundary layer to be thin compared with the cylinder radius . the pohlhausen solution shows good agreement with both series, near and far from the nose, and enables an interpolation to be made (4) between them in the extensive range of distances from the nose for which neither is applicable . the final recommended curves, for the variation along the cylinder of skin friction, boundary-layer displacement area and momentum defect area, are displayed in graphical and tabular form (figure 1 and table 1) and are expected to be correct to within about 2 .

the velocity near the wall is closely proportional to the logarithm of the distance from the axis,. this is the profile used in the pohlhausen method . the analogy with the distribution of mean velocity in turbulent flow over a flat plate is discussed at the end of 2 .

.I 382

.T

a note on the laminar boundary layer on a circular cylinder in axial incompressible flow .

.A

howard r. kelly

.B

u.s. naval ordnance test station, inyokern, china lake, calif.

.W

a note on the laminar boundary layer on a circular cylinder in

axial incompressible flow .

a correction is made for the equation to compute the ratio

of the displacement thickness on a cylinder to the corresponding

thickness on a flat plate .

.I 383

.T

integration of the boundary layer equations for a plane in compressible

flow with heat transfer .

.A

meksyn, d.

.B

proc. roy. soc. series a. v. 231, 1955. pp 274-180 .

.W

integration of the boundary layer equations for a plane in compressible

flow with heat transfer .

the equations of motion of compressible viscous flow with vanishing

pressure gradient past a plane are integrated in semi-convergent

expressions, for the case when the physical constants depend on

temperature and the prandtl number is close to unity .

simple expressions are obtained for the temperature and velocity

distributions in the boundary layer, the drag coefficient, and their

dependence on the physical constants,.they contain the well-known results and several new ones .

for the case when the temperature of the boundary is either above, or

not much below, the temperature of the main flow, the results obtained

closely agree with crocco's numerical computations .

.I 384

.T

application of second-order shock-expansion theory

to several types of bodies of revolution .

.A

lavender,r.e. and deep,r.a.

.B

j. ae. scs. 23, 1956, 1052.

.W

application of second-order shock-expansion theory

to several types of bodies of revolution .

second-order shock-expansion theory is utilized to obtain equations

for the initial normal force curve slope, initial pitching moment curve

slope, and zero-lift wave drag for several type bodies of revolution .

bodies considered are the cone-cylinder, cone-cylinder-frustum,

cone-cylinder-frustum-booster, cone-frustum, and cone-frustum booster .

.I 385

.T

on a generalised porous-wall ?couette type? flow .

.A

lilley,g.m.

.B

j. ae. scs. 26,1959, 685.

.W

on a generalised porous-wall ?couette type? flow .

in a recent paper, the problem of a /couette-type/ flow in which the fixed wall is porous has been considered . the results quoted in the above reference can be obtained rigorously by the method stated below in which a different interpretation to one of the parameters is made .

.I 386

.T

a generalised porous-wall ?couette type? flow .

.A

cramer,k.r.

.B

j. ae. scs. 26, 1959, 121.

.W

a generalised porous-wall ?couette type? flow .

recently, it was observed that the two existing boundary-layer texts (references 1 and 2) did not contain a solution for the case of couette flow with a constant, uniformly distributed suction or blowing . thus, the following analysis considers a /couette-type/ flow between a stationary flat surface and a slightly inclined flat plate moving at a constant velocity . in addition, the flow is subjected to a constant, uniformly distributed suction or blowing at the fixed surface .

.I 387

.T

heat transfer for laminar flow in an annulus with porous wall .

.A

inman,r.m.

.B

j. ae. scs. 26, 1959, 532.

.W

heat transfer for laminar flow in an annulus with porous

wall .

temperature profiles and heat-transfer rates of established

incompressible flow through an annulus channel with porous walls

of constant temperatures are determined at different injection rates .

axial conduction and viscous dissipation are, as usual, neglected .

injecting fluid is tacitly assumed to have the same temperature as

the porous wall .

.I 388

.T

the pressure gradient induced by shear flow past a

flat plate .

.A

glauert,m.b.

.B

j. ae. scs. 1962, 540.

.W

the pressure gradient induced by shear flow past a

flat plate .

article is a continuation of an earlier note on papers by li

on a semiinfinite plate in a uniform shear flow . li had deduced from the form of his equations that stream vorticity caused an

induced pressure gradient in the flow . later papers by li and

murray (amr 15(1962), rev. 7157) support the induced pressure

gradient theory . the author notes, however, that the mathematics used

by li and murray are not acceptable and the problem thus not

resolved . the present note sets up simple models of complete flows

examinable by elementary means . author holds that analyses

demonstrate conclusively that no pressure gradient is induced in

the boundary layer on a flat plate in a limited region of shear flow .

he notes that the original question in the case of unbounded shear

remains obscure--and anyway an unlimited shear layer is not of

great practical importance .

.I 389

.T

simple shear flow past a flat plate in a compressible

viscous fluid .

.A

li,t.y.

.B

j. ae. scs. 22, 1955, 724.

.W

simple shear flow past a flat plate in a compressible

viscous fluid .

by transformation of variables, the problem of a simple shear flow of

a compressible fluid over a flat plate is reduced to the corresponding

problem for an incompressible fluid . the prandtl number of the

compressible fluid is assumed to be unity and its viscosity to be a linear function of temperature .

.I 390

.T

some panel-flutter studies using piston theory .

.A

johns,d.j.

.B

j. ae. scs. 25, 1958.

.W

some panel-flutter studies using piston theory .

the use of piston theory was recently advocated for supersonic aeroelastic analyses, including the problem of panel flutter, and this has stimulated the investigation reported here .

linear piston theory is mainly considered, but some effects of introducing higher order terms are discussed .

flutter of rectangular simply supported panels and of elliptically shaped clamped-edge panels is considered, and some justification is provided for the use of /static/ aerodynamic forces and the neglect of aerodynamic damping . hence, it is concluded that ackeret loading gives more exact results than piston theory .

solution of the flutter equations is made by applying galerkin's method to a rayleigh-type analysis using assumed modes of deformation .

.I 391

.T

flutter of rectangular simply supported panels at high supersonic speeds .

.A

hedgepeth,j.m.

.B

j. ae. scs. 24, 1957.

.W

flutter of rectangular simply supported panels at high

supersonic speeds .

the problem of panel flutter of rectangular simply supported

plates subjected to supersonic flow over one surface is treated

theoretically . the assumption is made, and subsequently

verified, that the /static/ approximation to the aerodynamic flutter

forces yields flutter boundaries with satisfactory accuracy for

mach numbers greater than about 2 . two panel flutter analyses

are performed using this static approximation in conjunction with

thin-plate theory--one employs aerodynamic strip theory, the

other aerodynamic surface theory . the influence of mach

number, dynamic pressure, panel aspect ratio, and midplane

stress on the panel thickness required to prevent flutter is

determined for extensive ranges of these parameters .

.I 392

.T

natural frequencies of rectangular plates with edges

elastically restrained against rotation .

.A

jogarao,v. and lakshmikantham,c.

.B

j. ae. scs. 1957.

.W

natural frequencies of rectangular plates with edges

elastically restrained against rotation .

plates with attachments to heavier members along the

edges can be described as having edges elastically restrained

against rotation, in many cases uniformly along each edge . at

the edges, setting slope, when is the edge bending

moment with always positive, the elastic restraint can be

analytically defined with describing respectively,

clamped and simply supported edges . in this note natural

frequencies of such plates are calculated mainly following the

nomenclature of dana young .

.I 393

.T

the shear flow along a flat plate with uniform suction .

.A

sakurai,t.

.B

j.ae.scs. 24, 1957.

.W

the shear flow along a flat plate with uniform suction .

recently, several authors have investigated the boundary

layer in a shear flow . in this note, an exact solution of the

navier-stokes equations will be presented, which represents the

boundary layer along an infinite flat plate with uniform suction

situated in a shear flow .

.I 394

.T

the viscous flow near a stagnation point when the external flow has

uniform vorticity .

.A

j. t. stuart

.B

national physical laboratory, teddington, middlesex, england

.W

the viscous flow near a stagnation point when the external flow has

uniform vorticity .

in view of the recent controversy between li and glauert on the nature

of the solution of the boundary-layer equations when the external

flow is rotational, it seems worthwhile to draw attention to a certain

exact solution of the navier-stokes equations which lends support to

glauert's point of view .

.I 395

.T

new methods in heat flow analysis with application

to flight structures .

.A

biot,m.a.

.B

j. ae. scs. 24, 1957.

.W

new methods in heat flow analysis with application

to flight structures .

new methods are presented for the analysis of transient heat flow in complex structures, leading to drastic simplifications in the calculation and the possibility of including nonlinear and surface effects . these methods are in part a direct application of some general variational principles developed earlier for linear thermodynamics . they are further developed in the particular case of purely thermal problems to include surface and boundary-layer heat transfer, nonlinear systems with temperature-dependent parameters, and radiation . the concepts of thermal potential, dissipation function, and generalized thermal force are introduced, leading to ordinary differential equations of the lagrangian type for the thermal flow field . because of the particular nature of heat flow phenomena, compared with dynamics, suitable procedures must be developed in order to formulate each problem in the simplest way . this is done by treating a number of examples . the concepts of penetration depth and transit time are introduced and discussed in connection with one-dimensional flow . application of the general method to the heating of a slab, with temperature-dependent heat capacity, shows a substantial difference between the heating and cooling processes . an example of heat flow analysis of a supersonic wing structure by the present method is also given and requires only extremely simple calculations . the results are found to be in good agreement with those obtained by the classical and much more elaborate procedures .

.I 396

.T

variational and lagrangian thermodynamics of thermal convection-fundamental shortcomings of the heat transfer

coefficient .

.A

biot,m.a.

.B

j. ae. scs. 29, 1962.

.W

variational and lagrangian thermodynamics of thermal

convection-fundamental shortcomings of the heat transfer

coefficient .

extension of previous analyses, indicating the possibility of

extending the thermodynamics of irreversible processes to systems

which are not in the vicinity of an equilibrium state and for which

onsager's relations are not verified . this involves generalizations

beyond the narrow field of heat transfer and to principles of wider

range than those of current nonequilibrium thermodynamics .

.I 397

.T

a sublayer for fluid injection into the incompressible

turbulent boundary layer .

.A

turcotte,d.l.

.B

j. ae. scs. 27, 1960.

.W

a sublayer for fluid injection into the incompressible

turbulent boundary layer .

  a sublayer region is introduced in which the intensity of

turbulence grows at a prescribed rate . the decrease in wall shear

stress due to fluid injection into the boundary layer is found under

the hypothesis that the effect of injection is restricted to the

sublayer region . experimental measurements of the velocity

profiles with fluid injection substantiate this hypothesis . the

theoretical decrease in wall shear stress is in good agreement

with experiment,. the solution is particularly simple and for small

values of the injection parameter it contains no arbitrary

parameters . the theory provides a similarity parameter which differs

from the one in general use .

.I 398

.T

heat transfer in turbulent shear flow .

.A

rannie,w.d.

.B

j. ae. scs. 23, 1956.

.W

heat transfer in turbulent shear flow .

  the problems of heat transfer in turbulent shear flow along a

smooth wall are discussed from the point of view of von karman's

well-known 1939 paper on the analogy between fluid friction and

heat transfer . methods for extending the analysis to higher

prandtl numbers are suggested .

.I 399

.T

conduction of heat in composite slabs .

.A

jaeger, j. c.

.B

quart. appl. math. v. 8 july, 1950 . pp 187-198

.W

conduction of heat in composite slabs .

a method of calculating the total quantity of heat that passes through a unit area from zero time to time t is developed . allowance is made for surface resistance by regarding each contact resistance as an additional layer of the appropriate thermal resistance and zero heat capacity

.I 400

.T

buckling stress of clamped rectangular plates in shear .

.A

budiansky,b. and connor,r.w.

.B

naca tn.1559, 1948.

.W

buckling stress of clamped rectangular plates in shear .

by consideration of antisymmetrical, as well as symmetrical, buckling configurations, the theoretical shear buckling stresses of clamped rectangular flat plates are evaluated more correctly than in previous work . the results given, which represent the average of upper and lower-limit solutions obtained by the lagrangian multiplier method, are within percent of the true buckling stresses .

.I 401

.T

inviscid hypersonic airflows with coupled non-equilibrium

processes .

.A

hall,j.g., eschenroeder,a.w. and marrone,p.v.

.B

ias paper 62-67, 1962.

.W

inviscid hypersonic airflows with coupled non-equilibrium

processes .

analyses have been made of the effects of coupled chemical rate processes in external inviscid hypersonic airflows at high enthalpy levels . exact (numerical) solutions have been obtained by the inverse method for inviscid airflow over a near-spherical nose under flight conditions where substantial nonequilibrium prevails through the nose region . typical conditions considered include nose radii of the order of 1 ft at an altitude of 250,000 ft and velocities of 15,000 and 23,000 ft per sec .

the results illustrate the general importance of the coupling among the reactions considered . these included dissociation-recombination, bimolecular-exchange, and ionization reactions . the exact solutions show the bimolecular, no exchange reactions to be important in blunt-nose flow for the kinetics of no and n, as they are in the case of a plane shock wave . an important difference between blunt-nose flow and plane shock flow, however, is the gasdynamic expansion in the curved shock layer of the former . this expansion reduces post-shock reaction rates . as a consequence, in the regime studied the oxygen and nitrogen-atom concentrations tend to freeze in the nose region at levels below those for infinite-rate equilibrium . the reduction below the equilibrium dissociation level can be large, particularly for nitrogen dissociation at higher velocities .

in the regime considered, the chemical kinetics are dominated by two-body collision processes . the inviscid nose flow, including coupled nonequilibrium phenomena, is thus amenable to binary scaling for a given velocity . the binary scaling is demonstrated for a range of altitude and scale by correlation of the exact solutions for given velocity and a constant product of ambient density and nose radius . this similitude, which can also scale viscous nonequilibrium and radiation phenomena in the shock layer, provides a useful flexibility for hypersonic testing where it is applicable .

the afterbody inviscid-flow problem is briefly discussed in the light of the results for the nose flow .

.I 402

.T

magnetohydrodynamics shocks .

.A

de hoffman,f. and teller,e.

.B

phys. rev. 80, 1950, 693.

.W

magnetohydrodynamics shocks .

a mathematical treatment of the coupled motion of
hydrodynamic flow and electromagnetic fields is
given . two simplifying assumptions are introduced .. first,
the conductivity of the medium is infinite, and
second, the motion is described by a plane shock wave .
various orientations of the plane of the shock and
the magnetic field are discussed separately, and the
extreme relativistic and unrelativistic behavior is
examined . special consideration is given to the behavior
of weak shocks, that is, of sound waves . it is
interesting to note that the waves degenerate into common
sound waves and into common electromagnetic
waves in the extreme cases of very weak and very strong
magnetic fields .

.I 403

.T

magnetohydrodynamic shock waves .

.A

helger,l.

.B

astrophys. j. 117, 1953, 177.

.W

magnetohydrodynamic shock waves .

an interpretation of the de hoffman-teller shock-wave
equations for an infinitely conducting
medium is given analogous to the classical interpretation of
the ordinary hydrodynamic shock-wave

equations of rankine and hugoniot . two cases of interest

are considered as a consequence of this theory .

it is shown that weak magnetic fields in interstellar clouds

will be amplified, and, if external mechanisms

are available to reduce the compressional effects of shock

waves, the field will reach a value,

where p is the pressure . also, some aspects of the internal

motions of prominences are considered ,. it is

shown that gauss will yield results in accord with

the observational material .

.I 404

.T

two dimensional transonic flow past airfoils .

.A

kuo,y.a.

.B

naca tn.2356.

.W

two dimensional transonic flow past airfoils .

this report concerns the problem of constructing solutions for

transonic flows over symmetric airfoils . the aspect of the problem

emphasized is, of necessity, not how to form a solution for compressible

flow but how to simplify the initial phase of the problem, namely, the

mapping of the incompressible flow . in the case of the symmetric

joukowski airfoil without circulation, the mapping is relatively simple,

but the coefficients in the power series are difficult to evaluate . as

a result, the problem requires simplification . instead of the exact incompressible flow past the airfoil, an approximate flow is used, which

is derived from a combination of source and sink . this flow differs

only slightly from the exact one when the thickness is small . by the

same method, the flow with circulation is also considered .

after the incompressible-flow functions are approximated in this

fashion, the numerical calculation of the corresponding compressible

flow, by the hodograph theory, does not present any

essential difficulty .

.I 405

.T

tables of thermal properties of gases .

.A

joseph hilsenrath, chalres beckett, william bendict, liila fano, harold

hoge, joseph masi, ralph nuttall, yeram touloukian, harold woolley

.B

nbs circular 564 (1955)

.W

tables of thermal properties of gases .

tables of thermodynamic and transport properties

of air, argon, carbon dioxide, carbon monoxide, hydrogen,

nitrogen, oxygen, and steam .

.I 406

.T

on the behaviour of boundary layers at supersonic speeds .

.A

.B

.W

on the behaviour of boundary layers at supersonic speeds .

this paper considers the implications of recent advances in knowledge of the behaviour of boundary layers in supersonic flow . only the simplest case is considered-dashthat of the two-dimensional boundary layer on a flat plate, with nominal zero longitudinal pressure and temperature gradients .

it is shown that the empirical/intermediate enthalpy/used with success in approximations for skin friction, etc., of laminar boundary layers is closely the same as the mean enthalpy with respect to velocity . furthermore, the mean enthalpics of laminar and turbulent boundary layers may be the same . a nonrigorous approach is made to the problems of self-induced pressure gradients, and the indications are that their effects on laminar skin friction, etc., may become noticeable at mach numbers greater than 5 and they increase as the surface temperature builds up towards zero heat-transfer conditions . the effects with turbulent boundary layers may not be so severe .

finally, the results are applied to give an idea of the magnitude of the drag and aerodynamic heating problems up to m 10, and one result is that, if there is any conflict at the higher mach numbers between surface conditions required for high radiative emissivity and those which may be thought necessary for preserving a laminar boundary layer, then it may be better to choose the former .

.I 407

.T

stationary convection flow of an electrically conducting

liquid between parallel plates in a magnetic field .

.A

gershuni,g.z. and shukhovitskii,e.m.

.B

soviet physics,34, 1958.

.W

stationary convection flow of an electrically conducting

liquid between parallel plates in a magnetic field .

a study is made of the stationary convection

of an electrically conducting liquid in the space

between two parallel plates, heated to different

temperatures, in the presence of a magnetic

field . the distribution of velocity, temperature,

and induced fields are found, and the convective

heat flow is calculated .

.I 408

.T

on convective motion of a conducting fluid between

parallel vertical plates in a magnetic field .

.A

regirer,s.a.

.B

soviet physics, 37, 1960.

.W

on convective motion of a conducting fluid between

parallel vertical plates in a magnetic field .

stationary convective motion of a conducting fluid

between vertical parallel plates in a

magnetic field is considered . an exact solution of the magnetohydrodynamic equations is

obtained for the case of a constant vertical temperature

gradient . the critical value of

grasshof's number is determined for the case when the

temperature of both plates is the same .

.I 409

.T

on the base pressure resulting from the interaction of a supersonic

external stream with a sonic or subsonic jet .

.A

chow, w. l.

.B

j. aero. sc. march, 1959 . p. 176-180 .

.W

on the base pressure resulting from the interaction of a supersonic

external stream with a sonic or subsonic jet .

it is shown that the two-dimensional base pressure problems relating to

base bleed into the wake of blunt-trailing-edge airfoils, or the

interaction between an external supersonic or sonic slipstream with a

sonic or subsonic jet stream of a jet engine, can be calculated by

theoretical considerations . constant-pressure, isoenergetic, turbulent

mixing between the streams and the stagnant fluid in the wake is

assumed . the theoretical calculations are in good agreement with the

experimental results .

.I 410

.T

the supersonic flow about a blunt body of revolution for gases at chemical equilibrium .

.A

gravalos,f.g., edelflet,i.h. and emmons,h.

.B

9th int. astro. fed. 1958.

.W

the supersonic flow about a blunt body of revolution

for gases at chemical equilibrium .

the supersonic flow about a blunt body of

revolution for gases at chemical

equilibrium . a method to determine the shock

wave, and its location, about a body

of revolution moving at supersonic speeds is given .

the method provides also the means

to compute the flow characteristics in the shock layer .

the fluid in which the motion

takes place is assumed to be in chemical equilibrium

within the shock layer ,. its

thermochemical properties must be known . the essential

new features of the method are ..

a) it solves the direct problem, i. e., the initial data

are the conditions upstream and

the body shape ,. b) the integration of the fundamental

equations is done in the physical

plane and the difficulties inherent to other, less direct,

mathematical formulations

of the problem are avoided . a physical interpretation of the method is made which

is in accord with the analytical definition of the problem .

.I 411

.T

data on shape and location of detached shock waves

in cones and sphere .

.A

herberle,j.w., wood,g.p. and gooderum,p.b.

.B

naca tn.2000, 1950.

.W

data on shape and location of detached shock waves

in cones and sphere .

accurate experimental data are given on the shape and the location

of detached shock waves on cones and spheres at mach numbers from 1.17

to 1.81 . the data are correlated to obtain equations that describe the

shock waves . this knowledge of the shock waves should be useful in

calculations of the pressure distribution and the pressure drag of the

fore part of cones and spheres . the experimental data on shock waves

are compared with theory .

.I 412

.T

critical combinations of shear and transverse direct

stress for an infinitely long flat plate with

edges elastically restrained against rotation .

.A

batdorf,s.b. and houbolt,j.c.

.B

.W

critical combinations of shear and transverse direct

stress for an infinitely long flat plate with

edges elastically restrained against rotation .

an exact solution and a closely concurring approximate

energy solution are given for the buckling of an infinitely long

flat plate under combined shear and transverse direct stress

with edges elastically restrained against rotation . it was found

that an appreciable fraction of the critical stress in pure shear

may be applied to the plate without any reduction in the

transverse compressive stress necessary to produce buckling . an

interaction formula in general use was shown to be decidedly

conservative for the range in which it is supposed to apply .

.I 413

.T

turbulent skin friction at high mach numbers and reynolds

numbers in air and helium . nasa r82, 1960 .

.A

matting,f.w., chapman,d.r., nyholm,j.r. and thomas,a.g.

.B

naca r847, 1946.

.W

turbulent skin friction at high mach numbers and reynolds

numbers in air and helium . nasa r82, 1960 .

results are given of local skin-friction measurements in turbulent

boundary layers over an equivalent air mach number range from 0.2 to 9.9

and an over-all reynolds number variation of 2x10 to 100x10 . direct

force measurements were made by means of a floating element . flows

were two-dimensional over a smooth flat surface with essentially zero

pressure gradient and with adiabatic conditions at the wall . air and

helium were used as working fluids . an equivalence parameter for

comparing boundary layers in different working fluids is derived and

the experimental verification of the parameter is demonstrated .

experimental results are compared with the results obtained by several

methods of calculating skin friction in the turbulent boundary layer .

.I 414

.T

the problem of resistance in compressible fluids .

.A

von karman,t.

.B

5th volta cong. 1955,226.

.W

the problem of resistance in compressible fluids .

this report is restricted to the

resistance of bodies of revolution and

of cylindrical bodies of infinite length

moving with uniform velocity in

a compressible fluid . in the case of bodies

of revolution it will be assumed

that the direction of the movement is

parallel to the axis of symmetry .

it will be assumed that the fluid satisfies the equation of state of perfect

gases, i. e. const., where p denotes

the pressure, the density and

t the absolute temperature . in addition

to obeying this equation the

fluid is characterized by the statement

that the intrinsic energy of the

unit mass amounts to where for

simplicity's sake the specific heat

will be expressed in work rather

than heat units . the ratio between

the specific heat at constant pressure

and the specific heat at constant

volume will be denoted by . it is

known that the value of x depends

upon the number of degrees of freedom

of the molecules,. if this number

is denoted by n . for air

the value x = 1.4 will be used .

the limiting case x = 1 will

be referred to as that of a

assumed that in the range

considered and are independent of

the temperature .

.I 415

.T

the aerodynamic design of section shapes for swept wings .

.A

pearcey, h.h.

.B

2nd inter. congr. of int. council of aero. sc. september, 1960 .

.W

the aerodynamic design of section shapes for swept wings .

an extension of work of lock and rogers and the result of cooperation by

n.p.l., r.a.e. and members of the british aircraft industry to achieve a

satisfactory design for an aircraft cruising at low supersonic speeds .

knowledge of shock-wave prediction, onset of wave drag and

shock-induced separation allows the basic design to be generalized for a wide

range of parameters . unpublished work by bagley on the relation of

aerodynamic coefficients and geometry is used . the role of upper

surface velocity distribution is noted and methods for predicting pressure

distributions with shock waves are reviewed for both subsonic and

transonic flows .

.I 416

.T

methods of boundary-layer control for postponing and alleviating

buffeting and other effects of shock-induced separation .

.A

pearcey, h.h. and stuart, c.m.

.B

smf fund paper, no. f.f. -dash 22, 1959 .

.W

methods of boundary-layer control for postponing and alleviating

buffeting and other effects of shock-induced separation .

the use of boundary-layer control to increase the separation-free margins of mach number and lift coefficient beyond the cruise point of high-speed aircraft may often be preferred to design changes that impair the cruising performance or the landing and take-off characteristics . the factors that influence the choice of method and details of its application are discussed, emphasising particularly the need to maintain effectiveness over most of the chord to cover the wide range of separation positions encountered as the shock moves over the wing with changing flight conditions .

research at the national physical laboratory that has embraced high-velocity blowing, vane and air-jet vortex generators, and, in a preliminary way, distributed suction, is briefly described . the relative merits of the various methods are discussed, and some results achieved in their application are given .

for vortex generators, the importance is stressed of the vortex paths determined by the interactions of neighbouring vortices and their images . thus, systems of counter-rotating vortices always leave the surface in pairs and lose their effectiveness . co-rotating systems are therefore preferred for many applications . blowing, which in wind-tunnel tests gives results as good as or better than vortex generators and does not have the disadvantage of a drag penalty at cruise, has not yet been assessed in flight . air-jet vortex generators, which would also avoid the drag penalty, show promise of producing significant effects with relatively small blowing pressures and quantities .

.I 417

.T

on the stability of two dimensional parallel flows .

.A

lin,c.c.

.B

pt.iii - stability in a viscous fluid. q. app. math. 3, 1945, 273.

.W

on the stability of two dimensional parallel flows .

this is the last part of the author's theory of the stability of plane laminar motion . (for parts 1 and 2, cf. the same quart. 3, 117-142, 218-234 (1945),. these rev. 7,225,226 .) the stability character of a viscous fluid is considered in detail . the author proceeds first to give a proof of a criterion of stability due to heisenberg .. if a velocity profile has an number and phase velocity, the disturbance with the same wave number is unstable in the real fluid when the reynolds number is sufficiently large . this destabilizing effect of viscosity is one of the most interesting phenomena in the general stability theory,. its physical and mathematical significance is carefully discussed .

the author then discusses the behavior of the so-called neutral curve for the two characteristic types of velocity distribution, the boundary layer type profile and the symmetrical profile . the asymptotic behavior of the neutral curve is discussed first . the main difference between profiles with and without a point of inflection is that the two branches of the neutral curve approach and for profiles with a flex, but both converge to for the profile without a flex . the most important results are as follows .

for sufficiently large reynolds number r . (2) there always exists a minimum r below which the motion is stable . a similar result was obtained by synge from energy considerations . synge found a limiting curve below which the motion is necessarily stable . the author's discussion of the asymptotic behavior of the curves shows further that there always exists a maximum value of a beyond which the motion is stable for all reynolds numbers . hence the qualitative shape of the curve is determined .

the author proceeds to show that simple approximate expressions for the stability limit can be obtained from his general analysis for a given velocity profile . these approximate stability limits for plane poiseuille flow and blasius flow are found to be r=5906 and r=502 . the reynolds numbers are based on the width of the channel and the displacement thickness, respectively . finally, the method for computing the complete instability curve is presented and the plane poiseuille case and the blasius problem worked out in detail . the stability limit for blasius flow had been given before by tollmien and schlichting . the present more exact computations agree well with tollmien's result as far as the minimum critical reynolds number is concerned . the value found here is r=420 . the neutral curve for poiseuille motion had not been obtained before . the minimum critical number here is found to be r=5314 . the agreement with the estimate from the simple criterion mentioned above is thus very good .

a discussion of the physical significance of the viscous

effects and of future developments concludes the paper .

.I 418

.T

transition form laminar to turbulent shear flow .

.A

morkovin,m.v.

.B

asme trans, 80, 1958,1121.

.W

transition form laminar to turbulent shear flow .

recent experimental studies of transition from laminar to

turbulent shear flows are reviewed . certain common features

are emphasized and related to the stability theories of viscous

shear layers . the three-dimensional character, the

unsteadiness, and the nonlinear and random behavior of the latter stages

of the transition process are also examined .

.I 419

.T

the design of intermediate vertical stiffeners on web

plates subjected to shear .

.A

rockey,k.c.

.B

aero. quart. 1956, 275.

.W

the design of intermediate vertical stiffeners on web plates subjected to shear .

the correct design of intermediate vertical stiffeners on web plates subjected to shear becomes very important when the web plates are designed to operate at loads close to their buckling loads . this paper presents details of an extensive series of tests conducted on stiffened web plates subjected to shear . from the analysis of the results obtained from these tests, new empirical relationships between the flexural rigidity and spacing of the intermediate stiffeners and the buckling stress of the stiffened web plate have been obtained .

one interesting and important feature of these new relationships is that they define more clearly than hitherto the difference in the behaviour of single-and double-sided stiffeners .

.I 420

.T

an experimental study of the flow field about swept and delta wings with sharp leading edges .

.A

jaszlics, i. and trilling, l.

.B

j. aero. sc. august 1959 . p 487 - 494, 544 .

.W

an experimental study of the flow field about swept and delta wings

with sharp leading edges .

a series of experiments was performed to define the flow field on the

upper surface of high aspect ratio swept wings and narrow delta wings at

high angles of attack .

it was found that near the root section of either type of wing the flow

is conical . the edge of the vortex sheet which originates at the

leading edge is a straight line whose position relative to the leading edge

depends only on incidence . on swept wings, the vortex edge turns

down-stream as soon as the vortex sheet covers the front half of the wing

chord, and the flow under the vortex sheet outboard of that turning

point is uniform and parallel to the leading edge of the wing . on

narrow delta wings, the conical symmetry persists almost to the trailing

edge .

.I 421

.T

analytic study of induced pressure on long bodies of

revolution with varying nose bluntness at hypersonic

speeds .

.A

van hise,v.

.B

nasa r78, 1960.

.W

analytic study of induced pressure on long bodies of

revolution with varying nose bluntness at hypersonic

speeds .

a systematic study of induced pressures on a series

of bodies of revolution with varying nose bluntness

has been made by using the method of characteristics

for a perfect gas . the fluid mediums investigated

were air and helium and the mach number range

was from 5 to 40 . a study of representative shock

shapes was also made . flow parameters obtained

from the blast-wave analogy gave good correlations of

induced pressures and shock shapes . the

induced-pressure correlations yielded empirical equations for

air and helium which cover the complete range of

nose bluntness considered . (nose fineness ratios

varied from 0.4 to 4.) available experimental

results were in good agreement with the characteristics

solutions . properties connected with the concept of

hypersonic similitude enabled correlations of the

calculations to be made with respect to nose shape,

mach number, and ratio of specific heats .

.I 422

.T

bending of a square plate with two adjacent edges free

and the others clamped or simply supported .

.A

leissa,a.w. and niedenfuhr,f.w.

.B

a.i.a.a. jnl. 1, 1963, 116.

.W

bending of a square plate with two adjacent edges free

and the others clamped or simply supported .

the title problems were solved for the two cases .. (1) uniform

transverse loading, (2) a concentrated force at the free corner . a

function is chosen to exactly satisfy the biharmonic equation

while the boundary conditions are enforced at a number of points

plied at discrete points around the boundary for each of the four

problems and the resulting 35 simultaneous equations were solved

on an ibm 704 . tables listing the values of deflection and

bending moments are presented . this paper provides useful information

on the solution of these problems which are intractable by

analytical methods .

.I 423

.T

an experimental investigation of the flow over blunt-nosed

cones at a mach number of 5. 8.

.A

machell,r.m. and o'bryant,w.t.

.B

guggenheim aero. lab. memo 32, 1956.

.W

an experimental investigation of the flow over blunt-nosed

cones at a mach number of 5. 8.

shock shapes were observed and static pressures were measured

on spherically-blunted cones at a nominal mach number of 5.8 over a

range of reynolds numbers per inch from 97,000 to 238,000, for angles of yaw from 0 to 8 . six combinations of the bluntness ratios 0.4, 0.8, and 1.064 with the cone half angles 10, 20, and 40 were used in determining the significant parameters governing pressure distribution .

the pressure distribution on the spherical nose for both yawed and unyawed bodies is predicted quite accurately by the modified newtonian theory given by, where is the angle between the normal to a surface element and the flow direction ahead of the bow shock . cone half angle was found to be the significant parameter in determining the pressure distribution near the nose-cone junction and over the conical afterbody . on the 40 spherical nosed cone models the flow overexpanded with respect to the taylor-maccoll pressure in the region of the spherical-conical juncture, after which the pressure returned rapidly to the taylor-maccoll value . for models with smaller cone angles the region of minimum pressure occurred farther back on the conical portion of the model, and the taylor-maccoll pressure was approached more gradually . the shape of the pressure distributions as described in nondimensional coordinates was independent of the radius of the spherical nose and of the reynolds number over the range of reynolds number per inch between .97 x 10 and 2.38 x 10 .

integrated results for the pressure foredrag of the models at zero yaw compared very closely with the predictions of the modified newtonian approximation, except for models with large cone angles and small nose radii, where the drag approaches the value given by the taylor-maccoll theory for sharp cones .

.I 424

.T

cantilever plate with concentrated edge load .

.A

holl,d.l.

.B

j.app.mech. 1937, 8.

.W

cantilever plate with concentrated edge load .

the author gives, by the method of finite differences, an

approximate solution of the problem of a finite length

of a cantilever plate which bears a concentrated load at

the longitudinal free edge . all the boundary conditions

are taken into account, and the plate action is

determined approximately at all points of the plate . the

author points out that a secondary maximum transverse

stress occurs at the clamped edge nearest the loading

point, and that the longitudinal stress is greatest directly

under the loading point .

.I 425

.T

the solution of elastic plate problems by electrical analogies .

.A

r. h. macneal,

.B

pasadena, calif.

.W

the solution of elastic plate problems by electrical analogies .

a dynamic-analogy method for the solution of elastic plate problems is

described in this paper . the electrical circuits developed here can be

set-up and studied on an electric-analog computer . problems involving
deflections under constant load, transient vibrations, or normal
modes can be solved in this way . the method of applying boundary
conditions to plates with irregular edges is given, together with
a detailed description of the representation of the boundary conditions
for a rectangular variable-thickness plate . solutions that have been
obtained on the cal tech electric-analog computer are presented for
the static deflections and normal modes of a rectangular cantilever
plate .

.I 426

.T

preliminary analysis of axial flow compressors having
supersonic velocity at the entrance of the stator .

.A

ferri,a.

.B

naca rm l9g06, 1949.

.W

preliminary analysis of axial flow compressors having
supersonic velocity at the entrance of the stator .

a supersonic compressor design having supersonic velocity at the
entrance of the stator is analyzed on the assumption of two-dimensional
flow . the rotor and stator losses assumed in the analysis are based on
the results of preliminary supersonic cascade tests . the results of
the analysis show that compression ratios per stage of 6 to 10 can be
obtained with adiabatic efficiency between 70 and 80 percent .

consideration is also given in the analysis to the starting, stability, and range of efficient performance of this type of

compressor . the desirability of employing variable-geometry stators

and adjustable inlet guide vanes is indicated . although either

supersonic or subsonic axial component of velocity at the stator

entrance can be used, the cascade test results suggest that higher

pressure recovery can be obtained if the axial component is supersonic .

.I 427

.T

flow of gas through turbine lattices .

.A

deich,m.e.

.B

naca tm.1393, 1956.

.W

flow of gas through turbine lattices .

  paper is a translation of chap. 7 of the book /technical

gas-dynamics/ (see amr 9, rev 1869) . the topics treated are best

shown by the list of paragraph headings . they are .. 7-1 .

geometrical and gasdynamical parameters of the lattices,.

fundamentals of flow through lattices,. 7-2 . theoretical methods of

investigation or plane potential flow of incompressible fluid through a

lattice,. 7-3 . electro-hydrodynamic analogy,. 7-4 . forces acting

on an airfoil in a lattice,. theorem of joukowsky for lattices,. 7-5 .

fundamental characteristics of lattices,. 7-6 . friction losses in

plane lattice at subsonic velocities,. 7-7 . edge losses in plane

lattice at subsonic velocities,. 7-8 . several results of

experimental investigations of plane lattices at small subsonic velocities,. 7-9 . flow of gas through lattice at large subsonic

velocities,. critical mach number for lattice,. 7-10 . profile losses in

lattices at large subsonic velocities,. 7-11 . flow of a gas through

reaction lattices at supersonic pressure drops,. 7-12 . impulse

lattices in supersonic flow,. 7-13 . losses in lattices at near sonic

and supersonic velocities,. 7-14 . computation of angle of

deflection of flow in overhang section of a reaction lattice at supersonic

pressure drops,. 7-15 . characteristic features of three-dimensional

flow in lattices .

.I 428

.T

the quasi-cylinder of specified thickness and shell

loading in supersonic flow .

.A

portnoy,j.

.B

aero. quart. 11, 1960, 387.

.W

the quasi-cylinder of specified thickness and shell

loading in supersonic flow .

the methods of the operational

calculus are used to obtain a linear

approximation to the shape of the mean camber

surface of a quasi-cylinder in a

supersonic flow in terms of its shell thickness

and loading distributions . the

analysis deals with a generalised quasi-cylinder ,.

that is one which, although lying

close to a mean cylinder, need not possess

axial symmetry . the quasi-cylinder

is also permitted to be within the small

disturbance field of other separate

components, e.g. a centre-body . because

the linearised theory is inadmissable

for internal duct flows close to and beyond

the first reflected characteristic cone,

the present solution is likewise invalid close

to and beyond the position where

this characteristic meets the mean cylinder .

the work given here enables the

camber shapes of /ring-wings/, which have

been used theoretically to reduce or

even nullify the wave-drag of a central slender-body,

to be found . an example

illustrates the general method .

.I 429

.T

a description of the r. a. e. high speed supersonic

tunnel .

.A

poole,j.a.

.B

rae tn.aero.2678,1960

.W

a description of the r. a. e. high speed supersonic

tunnel .

an account is given of the high supersonic speed tunnel now nearing

completion . the design philosophy is reviewed, the principal features

are described and some of the more interesting development problems

are noted .

.I 430

.T

calibration of the flow in the mach 4 working section

of the 4ft . x 3ft . high supersonic speed wind tunnel

at rae bedford .

.A

andrews,d.r. and brown,c.s.

.B

rae tn.aero.2820, 1962.

.W

calibration of the flow in the mach 4 working section

of the 4ft . x 3ft . high supersonic speed wind tunnel

at rae bedford .

mach number and flow angle distributions in the working section of

the mach 4 nozzle of the 4 ft x 3 ft high-supersonic-speed wind tunnel

are presented for a range of total pressure and humidity .

.I 431

.T

free-flight measurements of the zero-lift drag and

base pressure on a wind tunnel interference model (m=0 .

8 - 1. 5) .

.A

greenwood,g.h.

.B

rae tn.aero.2725, 1960.

.W

free-flight measurements of the zero-lift drag and

base pressure on a wind tunnel interference model (m=0 .

8 - 1. 5) .

five free-flight models were flown to measure the zero-lift drag and

body base pressure on a standard wind tunnel interference model over a

mach number range of 0.84 to 1.48 .

roughness bands on the wings and body of the model are shown to

produce a small but definite increase in the zero-lift drag at all mach

numbers .

the measured drag is in fair agreement with corresponding measurements

made in various transonic tunnels with differences that could plausibly

be explained as the effects of tunnel interference .

the effect of a simulated wind tunnel support sting is shown to

increase the base pressure . the discrepancy between models with and

without a sting is greatest at subsonic speeds and progressively

decreases with increasing mach number until at m = 1.4 the sting has no

effect on base pressure .

.I 432

.T

theoretical damping in roll and rolling moment due

to differential wing incidence for slender cruciform

wings and wing-body combinations .

.A

adams,g.j. and dugan,d.w.

.B

naca r1088, 1952.

.W

theoretical damping in roll and rolling moment due

to differential wing incidence for slender cruciform

wings and wing-body combinations .

a method of analysis based on slender-wing theory is developed

to investigate the characteristics in roll of slender cruciform wings

and wing-body combinations . the method makes use of the

conformal mapping processes of classical hydrodynamics which

transform the region outside a circle and the region outside

an arbitrary arrangement of line segments intersecting at the

origin . the method of analysis may be utilized to solve other

slender cruciform wing-body problems involving arbitrarily

assigned boundary conditions .

in the present report, the application of the method has

shown ..

differential incidence of both pairs of opposite surfaces of the

cruciform wing-body combinations are practically independent

of the body-diameter-maximum-span ratio up to a value of this

ratio of 0.3 .

arrangement is only 62 percent greater than that for a

corresponding planar wing-body combination .

dence of both pairs of the opposing surfaces of the cruciform

wing-body arrangement, is only 52 percent greater than that for a corresponding planar wing-body combination .

unit surface deflection) of the cruciform wing-body arrangement

having four equally deflected panels is therefore 94 percent of

the corresponding planar wing-body combination .

.I 433

.T

application of two dimensional vortex theory to the

prediction of flow fields behind wings of wing-body

combinations at subsonic and supersonic speeds .

.A

rogers,a.w.

.B

naca tn.3227, 1954.

.W

application of two dimensional vortex theory to the

prediction of flow fields behind wings of wing-body

combinations at subsonic and supersonic speeds .

a theoretical investigation has been made of a general method for

predicting the flow field behind the wings of plane and cruciform wing

and body combinations at transonic or supersonic speeds and slender

configurations at subsonic speeds . the wing trailing-vortex wake is

represented initially by line vortices distributed to approximate the

spanwise distribution of circulation along the trailing edge of the

exposed wing panels . the afterbody is represented by corresponding

image vortices within the body . two-dimensional line-vortex theory is

then used to compute the induced velocities at each vortex and the

resulting displacement of each vortex is determined by means of a numerical stepwise integration procedure . the method was applied to the calculation of the position of the vortex wake and the estimation of downwash at chosen tail locations behind triangular-wing and cylindrical-body combinations at supersonic speeds . the effects of such geometric parameters as aspect ratio, angle of attack and incidence, ratio of body radius to wing semi-span, and angle of bank on the vortex wake behind wings of wing-body combinations were studied . the relative importance of wing vortices, the corresponding image vortices within the body, and body crossflow indetermining the the total downwash was assessed at a possible tail location .

it was found that the line-vortex method of this report permitted the calculation of vortex paths behind wings of wing-body combinations with reasonable facility and accuracy . a calculated sample wake shape agreed qualitatively with one observed experimentally, and sample results of the line-vortex method compared well with an available exact crossflow-plane solution . an empirical formula was derived to estimate the number of vortices required per wing panel for a satisfactory computation of downwash at tail locations . it was found that the shape of the vortex wake and the ultimate number of rolled-up vortices behind a wing depend on the circulation distribution along the wing trailing edge . for the low-aspect-ratio plane wing and body combinations considered, it appeared that downwash at horizontal tail locations is largely determined except near the tail-body juncture by the wing vortices alone for small ratios of body radius to wing semispan, and by the body upwash alone for large values of that ratio .

.I 434

.T

contributions of the wing panels to the forces and
moments of supersonic wing-body combinations at combined
angles .

.A

spahr,j.r.

.B

naca tn.4146, 1958.

.W

contributions of the wing panels to the forces and
moments of supersonic wing-body combinations at combined
angles .

a wind-tunnel investigation was
conducted at a mach number of 1.96
and at reynolds numbers (based on the
mean aerodynamic chord of the
exposed wing) of 0.36 and 1.03 million
to determine the normal forces,
pitching moments, and rolling moments
contributed by each wing panel of
a cruciform-wing and body combination
over a wide range of combined
angles of pitch and roll . the wings
were triangular of aspect ratio 2,
and the body was an ogive-cylinder
combination . the effects of forebody
length and roughness and of the
presence of the adjacent panels on these

panel contributions were determined .

the results of the investigation

show that large changes in the panel

forces and moments can occur as the

result of combined angles . a general

theoretical method based on slender-body

and strip theories was found to

yield results in good agreement with the

wind-tunnel measurements . these

comparisons indicate that the changes in

the panel characteristics due to

combined angles are caused primarily by

a cross coupling between the

side-wash velocities due to angle of attack

and sideslip and by the presence

of forebody vortices due to crossflow

separation . it was found that an

increase in forebody length increases

the effect of the forebody vortices

because of the dependence of the strength

of these vortices on the forebody

length .

.I 435

.T

application of similar solutions to calculations of

laminar heat transfer on bodies with yaw and large

pressure gradients in high speed flow .

.A

beckwith,i.e. and cohen,n.b.

.B

nada tn.d625, 1961.

.W

application of similar solutions to calculations of

laminar heat transfer on bodies with yaw and large

pressure gradients in high speed flow .

an integral method for the rapid calculation of heat-transfer distributions on yawed cylinders of arbitrary cross-sectional shape and on bodies of revolution in high-speed flows is developed for laminar boundary layers . the method involves the quadrature of a function of the pressure distribution (assumed given) and satisfies the integral energy equation with the assumption of local similarity, wherein the actual boundary-layer profiles at every station are replaced by corresponding profiles from a family of similar solutions .

the method is compared with other local similarity methods and with experimental heat-transfer data on a circular cylinder and on a body of revolution designed for large axial pressure gradients . good agreement between theory and data is obtained and it is shown that the present integral method, in both its complete and simplified form, gives generally better agreement with the data than certain other local similarity methods .

numerical examples are presented showing that the effect of sweep and gas properties on heat-transfer distribution is small .

.I 436

.T

heat transfer in planetary atmospheres at super-satellite speeds .

.A

hoshizaki, h.

.B

ars prep. 2173-61, august 1961 .

.W

heat transfer in planetary atmospheres at super-satellite speeds .

the main purpose of this investigation is to examine the dependence of heat transfer in planetary atmospheres on the total enthalpy up to flight velocities of 50,000 ft/sec where a large proportion of the atoms are ionized . the /total thermodynamic and transport property/ concept discussed by hirshfelder /j.chem.phys.,26/2/,feb.,1957/ is used .

.I 437

.T

hypervelocity stagnation point heat transfer .

.A

scala, s. m. and warren, w. r.

.B

arsj. jan. 1962 .

.W

hypervelocity stagnation point heat transfer .

this analysis includes the specific contributions of atoms, molecules, tions are .. /i/ partially ionized air can be approximated as a four-component gas including n2, n, n and e,. /ii/ the gas is in local thermochemical equilibrium,. /iii/ there is no charge separation,. /iv/ thermal diffusion is neglected,. /v/ no electrical or magnetic fields,. low re effects are neglected .

.I 438

.T

stagnation point heat transfer measurements at super

satellite speeds .

.A

offenhartz,e., wisblatt,h. and flagg, r.f.

.B

j. roy. aero. soc., 66, 1962.

.W

stagnation point heat transfer measurements at super

satellite speeds .

brief description of experiments performed by using shock tube

techniques for measurement of the stagnation point heating of a blunt

body over a stagnation enthalpy range of 650 to 900,

corresponding to velocities between 32,000 ft. per sec. and 39,000

ft per sec., respectively . data thus provided are used for

comparison with theory .

.I 439

.T

a factor affecting transonic leading edge flow separation .

.A

wood,g.p. and gooderum,p.b.

.B

naca tn.3804, 1956.

.W

a factor affecting transonic leading edge flow separation .

a change in flow pattern that was observed as the free-stream mach number was increased in the vicinity of 0.8 was described in naca technical note 1211 by lindsey, daley, and humphreys . the flow on the upper surface behind the leading edge of an airfoil at an angle of attack changed abruptly from detached flow with an extensive region of separation to attached supersonic flow terminated by a shock wave . in the present paper, the consequences of shock-wave--boundary-layer interaction are proposed as a factor that may be important in determining the conditions under which the change in flow pattern occurs . when the mach number is high enough, the attached-flow pattern exists because then the shock wave is far enough behind the leading edge to keep the influence of the high pressure behind the shock wave from extending through the boundary layer to the immediate vicinity of the leading edge and affecting the flow there . some experimental evidence in support of the importance of shock-wave--boundary-layer interaction is presented .

.I 440

.T

compilation of information on the transonic attachment
of flows at the leading edge of airfoils .

.A

lindsey,w.f. and landrum,e.

.B

naca tn.4204, 1958.

.W

compilation of information on the transonic attachment
of flows at the leading edge of airfoils .

schlieren photographs have been compiled of the two-dimensional flow at transonic speeds past 37 airfoils having variously shaped profiles, some of which are related and vary in thickness and camber . the data

for these airfoils were analyzed to provide basic information on the

flow changes involved and to determine factors affecting transonic-flow

attachment, which is a transition from separated to unseparated flow at

the leading edges of two-dimensional airfoils at fixed angles of attack

as the subsonic mach number is increased .

.I 441

.T

evaluation of high angle-of-attack aerodynamic derivative

data and stall-flutter prediction techniques .

.A

halfman,r.l., johnson,h.c. and haley,s.m.

.B

naca tn.2533, 1951.

.W

evaluation of high angle-of-attack aerodynamic derivative

data and stall-flutter prediction techniques .

the problem of stall flutter is approached in two ways . first,

using the m.i.t.-naca airfoil oscillator, the aerodynamic reactions on

wings oscillating harmonically in pitch and translation in the stall

range have been measured, evaluated, and correlated where possible with

available published data, with the purpose of providing empirical

information where no aerodynamic theory exists . the major effects of

reynolds number, airfoil shape, and reduced frequency on the aerodynamic

reactions have been reaffirmed . no instances of negative damping were

observed in pure translatory motion and the ranges of negative damping

occurring in pure pitch had the same general trends noted by other

experimenters . data on the time-average values in the stall range of
both lift and moment are presented for the first time .
  second, the results of numerous experimental observations of stall
flutter have been reviewed and the various known attempts at its
prediction have been examined, compared, and extended . the sharp drop in
critical speed and change to a predominantly torsional oscillation
usually associated with the transition from classical to stall flutter
is apparently primarily but not entirely caused by the marked changes
in moment due to pitch . fairly good stall-flutter predictions have
been reported only when adequate empirical data for this aerodynamic
reaction happened to be available for the desired airfoil shape,
reynolds number range, and reduced-frequency range . a semiempirical
method of predicting the variations of moment in pitch with airfoil
shape, reduced frequency, initial angle of attack, and amplitude of
oscillation has been presented .

.I 442

.T

some effects of variations in several parameters including
fluid density on the flutter speed of light uniform
cantilever wings .

.A

woolston,d.c. and castile,g.e.

.B

naca tn.2558, 1951.

.W

some effects of variations in several parameters including
fluid density on the flutter speed of light uniform

cantilever wings .

an experimental investigation has been made of some effects of variations in several parameters, including fluid density, on the flutter characteristics of light uniform cantilever wings . the assortment of wings tested covered a variety of positions of the elastic axis and center of gravity and values of the aspect ratio of 8, 6, and 4 . the relative-density parameter (where k is representative of the ratio of fluid density to wing mass) was varied over a range of values from 1.2 to nearly 14 . special emphasis has been placed on the lower values .

the experimental investigation has been supplemented by an analytical investigation based on the two-dimensional aerodynamic theory for incompressible flow . in a few instances corrections for the effects of finite span have been made . in general, the theoretical results followed the trends indicated by experiment except at very low values of the relative-density parameter . for these low values the analytical considerations employed indicated a freedom from flutter not found experimentally . at higher values of the flutter-speed coefficient is shown to decrease with decreasing values of and to be nearly proportional to the inverse of the square root of the air density .

.I 443

.T

calculated and measured pressure distributions over the midspan section of the naca 4412 airfoil .

.A

pinkerton, r. m.

.B

naca r. 563, 1936 .

.W

calculated and measured pressure distributions over the midspan section

of the naca 4412 airfoil .

pressures were simultaneously measured in the variable-density tunnel

at 54 orifices distributed over the midspan section of a 5 by 30 inch

rectangular model of the n.a.c.a. 4412 airfoil at 17 angles of attack

ranging from -dash 20degree to 30degree at a reynolds number of

approximately 3,000,000 . accurate data were thus obtained for studying the

deviations of the results of potential-flow theory from measured results

technique are presented .

it is shown that theoretical calculations made either at the effective

angle of attack or at a given actual lift do not accurately describe the

observed pressure distribution over an airfoil section . there is

therefore developed a modified theoretical calculation that agrees

reasonably well with the measured results of the tests of the n.a.c.a. 4412

section and that consists of making the calculations and evaluating the

circulation by means of the experimentally obtained lift at the

effective angle of attack,. i.e., the angle that the chord of the model makes

with the direction of the flow in the region of the section under

consideration . in the course of the computations the shape parameter

is modified, thus leading to a modified or an effective profile shape

that differs slightly from the specified shape .

.I 444

.T

an approach to the flutter problem in real fluids .

.A

rott,n. and george,m.b.t.

.B

inst. aero. scs. perp.509, 1955.

.W

an approach to the flutter problem in real fluids .

an approximate theory of airfoils in unsteady motion in a viscous fluid is proposed, in which viscous effects are accounted for by relaxing the kutta condition and replacing it by a relation derived from experiments in steady flow . applications here, are limited to moderate viscous effects below the stall . the possibility of one-degree-of-freedom flutter is discussed under this assumption . the discussion is partly extrapolated to the domain of stall flutter . some possibilities of further development of this theory for the stalled case are indicated .

.I 445

.T

on the application of mathieu functions in the theory of subsonic compressible flow past oscillating airfoils .

.A

reissner,e.

.B

naca tn.1961.

.W

on the application of mathieu functions in the theory of subsonic compressible flow past oscillating airfoils .

an account is given of explicit solutions in terms of mathieu function functions of the problem of two-dimensional subsonic compressible

flow past oscillating airfoils . the results are applied to the

calculation of three-dimensional corrections for the two-dimensional

theory and the effect of the incorporation of the three-dimensional

effects on the mathieu function solution of the two-dimensional

problem is shown . the developments are formal and must be

supplemented by an appreciable amount of numerical calculations before the

theory can be applied to specific problems .

.I 446

.T

wake of a satellite traversing the ionosphere .

.A

rand,s.

.B

phys. fluids, 3, 1960.

.W

wake of a satellite traversing the ionosphere .

the particle treatment is applied to a study of the

structure of the wake behind a charged body

moving supersonically through a low-density plasma .

for the case of a body whose dimensions are

considerably smaller than a debye length, a solution is obtained

which is very similar in structure to the

solution obtained by using the linearized fluid dynamics equation .

for the case of a disk whose radial

dimensions are much larger than a debye length, two

conical regions are found in the wake . at the

surface of each of these cones, over thicknesses of
the order of a debye length, the ion and electron
densities are increased over their ambient values .
formulae for the electrohydrodynamic drag on a
wire, and on a large disk are obtained .

.I 447

.T

motion of thin bodies in a highly rarefied plasma .

.A

yoshira,h.

.B

phys.fluids,4, 1961.

.W

motion of thin bodies in a highly rarefied plasma .
magnetic effects are considered negligible,
and the velocity of the body is in a range between the
electron and positive ion thermal speeds .
the self-consistent field approach is used in which the
electron distribution is assumed to be maxwellian,
while the positive ion distribution function is given
by the /collision-free/ boltzmann equation .
it is assumed that the ion reflection at the body surface
is specular, and the body is sufficiently thin so
that the ion distribution function is a small perturbation
of a maxwellian distribution . the solution for
the simple case of a dielectric body with a given surface
charge, as well as some general properties to

be expected for a conducting body are given .

.I 448

.T

induction drag on a large negatively charged satellite

moving in a magnetic-field-free ionosphere .

.A

wyatt,p.j.

.B

j. geophys. rev. 65, 1960.

.W

induction drag on a large negatively charged satellite

moving in a magnetic-field-free ionosphere .

an induction drag, experienced by a

charged satellite during its traversal of the

ionosphere, has been theoretically postulated by

several authors . previous 'exact' treatments

of the problem are inapplicable to large systems, and

the semiempirical approach of jastrow

and pearse may yield somewhat questionable results .

the present description initially

considers the satellite as a completely permeable spherical

shell of charge, thus avoiding the

difficult boundary conditions introduced by the 'exact'

linearized treatment . the effects of

permeability are then shown to be approximately removable

by means of an iterative process . a

final result, apparently valid to within an order of magnitude, is obtained for the drag force

arising solely from electrical effects . its magnitude

is considerably less than that obtained by

jastrow and pearse .

.I 449

.T

interaction of a charged satellite with the ionosphere .

.A

davis,a.h. and harris,i.

.B

nasa tn. d704, 1961.

.W

interaction of a charged satellite with the ionosphere .

the problem of the ion density distribution around a

charged satellite has been treated by a numerical method

which does not require linearization of the equations or

restriction to infinitesimal objects . however, magnetic field

effects were not considered, and a number of other

simplifying assumptions were required . some sample calculations

for spherical satellites are presented, illustrating the

general character of the satellite wake . calculations of the

so-called /charge drag/ were also made, yielding results

qualitatively similar to those previously obtained by jastrow and pearse .

.I 450

.T

some physical interpretations of magnetohydrodynamic duct flows .

.A

fujihiko sakao

.B

university of tokyo, tokyo, japan

.W

some physical interpretations of magnetohydrodynamic duct flows .

this note presents some physical interpretations of magnetohydrodynamic

duct flows with various boundary conditions viewed in the light of the

effects of conducting walls on the pattern of electric current, taking

examples from published results on rectangular ducts . the current

patterns are illustrated in fig. 1 for rectangular ducts having

various combinations of conducting and nonconducting walls, a uniform

magnetic field being applied in the horizontal direction .

.I 451

.T

liapunov's methods in automatic control theory .

.A

parks, p. c.

.B

control, november and december 1962 .

.W

liapunov's methods in automatic control theory .

the work of a. m. liapunov and his theory of stability is discussed .

the second method of liapunov is shown to have applications for linear

equations with real constant coefficients, for a proof of the

routh-hurwitz criterion, and linear equations with periodic coefficients .

practical examples include non-linear stability problems of control, and

the functions have uses in other areas of control systems .

.I 452

.T

symmetric joukowsky airfoils in shear flow .

.A

tsien,h.s.

.B

q. app. math. 1, 1943, 130.

.W

symmetric joukowsky airfoils in shear flow .

the velocity components of the fluid far from the airfoil

are given by where c is the chord of

the airfoil, and k are constants, u and v are velocity

components in the directions of the coordinates x and y .

the solution is sought in the form of the stream function

and satisfies laplace's equation . a general expression for

for vanishing disturbance velocities at points far from

the origin is written, and the flows due to a source, a vortex

and a solid circular cylinder in shear flow are considered

as examples . typical streamline patterns are shown for

these cases . from the eulerian equations of motion the

author obtains the expression for in terms of the

parameter and derivatives of . the general form

of is introduced and the appropriate solution for the

pressure p is obtained . by integration around a contour

enclosing the body, expressions are obtained, analogous to

the blasius formulae, for the force and couple on any

cylinder in this type of flow . these formulae are applied to the case of a symmetrical joukowsky airfoil . the method of

conformal transformation is employed in the determination

of . the boundary condition of tangential flow at the

airfoil surface must be satisfied by the total flow in the airfoil

plane, but this condition leads to a boundary condition for

in the transformed plane . the kutta-joukowsky

condition of finite velocity at the trailing edge also leads to a

condition on in this plane . from these conditions and the

general expression for the circulation and the strengths

of the doublets and quadruplets required for the force and

moment are determined . hence, the formulae for lift and

moment coefficient are obtained . these involve, in addition

to the usual (potential-flow) terms, terms proportional to .

the ten functions that appear in the expressions for the

lift and moment coefficients are tabulated for values of the

thickness ratio between 0 and 1 . the aerodynamic-center

position and the coefficient of the moment about the

aerodynamic center are also calculated and are presented

graphically as functions of .

.I 453

.T

the influence of two-dimensional stream shear on airfoil maximum lift .

.A

.B

.W

the influence of two-dimensional stream shear on airfoil maximum lift .

the cornell aeronautical laboratory is conducting a program of theoretical and experimental research on low-speed aerodynamics as applied to stol and vtol aircraft . the objective of this program is to re-examine certain aspects of classical aerodynamic information, in the light of low-speed flight requirements, with the aim of seeking aerodynamic processes which might be exploited to enhance law-speed performance .

one aspect of propeller-driven aircraft which has recently received increasing attention is the existence of strong gradients of longitudinal velocity, or shear, in the propeller slipstream . this slipstream shear interacts with a wing surface and can alter the wing characteristics . in theoretical treatments of a wing interacting with a propeller slipstream, the first important simplification is the replacement of the slipstream with an ideal uniform

jet, free of all velocity

gradients . the application of these theories requires that one equate the actual slipstream to an effective uniform jet . one method employed is to assume the uniform jet has a momentum flux equal to the average in the propeller slipstream . these and similar procedures are well founded on momentum considerations., however, the implicit assumption is that the flow nonuniformity, the shear, does not influence the wing characteristics .

.I 454

.T

several approximate analyses of the bending of a rectangular cantilever plate by uniform normal pressure .

.A

nash,w.a.

.B

j.app.mech. 1952, 33.

.W

several approximate analyses of the bending of a rectangular cantilever plate by uniform normal pressure .

three methods of approximating the deflections and moments occurring in a rectangular cantilever plate subjected to uniform normal pressure over its entire surface are presented in this paper . the first is the application of the well-known finite-difference procedure . the second and third are collocation methods, one based upon polynomial solutions of the lagrange equation, the other employing /mixed/ hyperbolic-trigonometric terms satisfying this equation . in the last two methods the boundary conditions are satisfied exactly along the clamped edge and at a finite number of points along the free edges of the plate . the results obtained for the particular case of a cantilever plate with uniform normal load indicate that the use of a relatively small number of points in the collocation method yields values of deflections and moments that are in substantial agreement with those given by the finite-difference procedure . it cannot be concluded from these results that the collocation method using the assumed functions will give satisfactory results with fewer points than the finite-difference method for cantilever plates with loading different from the one investigated .

.I 455

.T

modified cross-lees mixing theory for supersonic separated

and reattaching flows .

.A

glick,h.s.

.B

galcit hyp. res. proj. memo 53, 1960.

.W

modified cross-lees mixing theory for supersonic separated

and reattaching flows .

re-examination of the crocco-lees method has shown that the previous quantitative disagreement between theory and experiment in the region of flow up to separation was caused primarily by the improper c(k) relation assumed . a new c(k) correlation, based on low-speed theoretical and experimental data and on supersonic experimental results has been developed and found to be satisfactory for accurate calculation of two-dimensional, laminar, supersonic flows up to separation .

a physical model which incorporates the concept of the /dividing/ streamline and the results of experiment . according to this physical model, viscous momentum transport is the essential mechanism in the zone between separation and the beginning of reattachment, while the reattachment process is, on the contrary, an essentially inviscid process . this physical model has been translated into crocco-lees languages using a semiempirical approach, and approximate c(k) and f(k) relations have been determined for the separated and reattaching regions . the results of this analysis have been applied to the problem of

shockwave, laminar-boundary-layer interaction, and satisfactory

a study of separated and reattaching regions of flow has led to

quantitative agreement with experiment has been achieved .

.I 456

.T

a study of flow fields about some typical blunt-nosed

slender bodies .

.A

vaglio-laurin,r. and trella,m.

.B

pibal r.623, 1960.

.W

a study of flow fields about some typical blunt-nosed

slender bodies .

complete inviscid flow fields about three model axisymmetric

configurations have been determined numerically . configurations

decreasing bluntness) and flight conditions have been selected so

as to indicate separately effects of nose shape, drag coefficient,

flight mach number, and thermodynamic behavior of the gas (either

ideal calorically perfect gas or air in equilibrium dissociation) .

results are presented for thirteen cases . particular attention is

devoted to interpretation and, when possible, correlation of

pressure distributions on, and shock shapes about, the cylindrical

afterbodies . it is found that .. (a) the correlation of pressure

distributions on bodies having nonspherical noses involves interpretive

modifications of the law suggested by blast wave analogy . also shocks about these bodies are not described by parabolae,. (b) for

all configurations there is substantial influence of gas behavior

on shock shape,. this, however, can be correlated in terms of the

gas conditions along a generally defined streamline,. (c) the shock

layer can generally be divided into two regions (the first bound by

the body and the aforementioned streamline, the second delimited

by this streamline and the shock) wherein flow properties can

either be approximated by simple laws or correlated .. (d) for each

configuration knowledge of the complete flow field in one flight

condition (even pertaining to ideal gas flow) can be used to

estimate features of flows under general flight conditions including

those where equilibrium dissociation is encountered .

.I 457

.T

on laminar boundary-layer flow near a position of separation .

.A

goldstein,s.

.B

q. j. mech.app. mech. 1, 1948, 43.

.W

on laminar boundary-layer flow near a position of separation .

singularities are considered in the solution of

the laminar boundary-layer

equations at a position of separation . a singularity of

the type here considered occurred

in a careful numerical computation by hartree

for a linearly decreasing velocity

distribution outside the boundary layer ,. it may
occur generally . whenever it does
occur, the boundary-layer equations cease to be
valid at and near separation on the
upstream side, and also downstream of separation .
the work suggests that
singularities may arise in the solution of non-linear
parabolic equations due to their
non-linearity . the formulae found may help
computers of laminar boundary layers,
who desire more than a rough solution, to have
an end-point at which to aim .

.I 458

.T

a new series for calculation of steady laminar boundary
layer flows .

.A

gortler,h.

.B

j. math. mech. 6, 1957, 1.

.W

a new series for calculation of steady laminar boundary
layer flows .
a new and general method for solving
problems of plane and steady laminar
boundary layer flows in incompressible
fluids with arbitrary outer pressure

distribution is developed . this method

is based on the introduction of the

dimensionless quantities

as new independent spatial variables .

ordinates, u(x) the given outer velocity

distribution, v the kinematic viscosity .)

the solution of the boundary layer problem

is then given as a power series in e

with coefficient functions depending on n .

this series is a formally exact solution

of the boundary layer problem .

the new series solution has the following qualities ..

have the significance only of cartesian

coordinates, the influence of wall

curvature being neglected in boundary

layer theory, the new coordinates

are adjusted to the data of the special

problem in any case of application .

the new variables represent a logical

development of former efforts in

the field of boundary-layer flow calculation .

with other series solutions known for

some special cases is that the leading

term of the new series satisfies

exactly the outer boundary condition

at all cross-sections along the wall .

therefore, the succeeding terms give

corrections only in the inner part of
the boundary layer . accordingly,
taking also no. 1 into account, the zero
order term by itself gives a good
approximation for the boundary layer flow .

.I 459

.T

on the solution of the laminar boundary layer equations .

.A

tani,i.

.B

j.phys.soc.japan, 4, 1949, 149.

.W

on the solution of the laminar boundary layer equations .

the theory of the laminar boundary layer
offers a means of determining the skin friction
under the assumption of a given velocity
distribution outside the boundary layer . owing to the
mathematical difficulties, however, exact solutions
are possible only when the velocity distribution
is expressed as a simple function of the distance
along the surface . more complicated velocity
distributions necessitate recourse to the method of
expansion in series or that of step-by-step
calculations, but the labor involved is too great for the
methods to be of practical use . approximate
method due to pohlhausen (1921), which had long

been recommended for general use, gives a reasonably accurate solution in a region of accelerated flow, but recently its adequacy in a region of retarded flow has been questioned . separation of flow may actually occur where the solution of pohlhausen fails to give it . more recently howarth solution, which gives fairly reasonable results in a region of retarded flow .

howarth's solution essentially consists in solving the boundary layer equations for the particular case in which the velocity u outside the boundary layer decreases linearly with the distance x measured along the surface, and utilizing the solution by replacing the actual distribution of u by a circumscribing polygon of infinitesimal sides . therefore, it is assumed that the velocity distribution at any section depends on the velocity gradient du/dx at that section only, being affected by the conditions upstream only in so far as this affects the momentum thickness 0 . in other words, the velocity distribution across the boundary layer is determined by a parameter .

.I 460

.T

correlated incompressible and compressible boundary layers .

.A

stewartson,k.

.B

proc. roy. soc. a, 200, 1949, 84.

.W

correlated incompressible and compressible boundary

layers .

the boundary-layer equations for a compressible

fluid are transformed into those for an

incompressible fluid, assuming that the boundary

is thermally insulating, that the viscosity is

proportional to the absolute temperature, and

that the prandtl number is unity . various

results in the theory of incompressible boundary

layers are then taken over into the

compressible theory . in particular, the existence of

method for retarded flows is applied to determine

the point of separation for a uniformly

retarded main stream velocity . a comparison with

an exact solution is used to show that this

method gives a closer approximation than does pohlhausen's .

.I 461

.T

approximate methods fore predicting separation properties

of laminar boundary layers .

.A

curle,n. and skan,s.w.

.B

aero. quart. 8, 1957, 257.

.W

approximate methods fore predicting separation properties

of laminar boundary layers .

some new solutions for steady incompressible laminar boundary

layer flow, obtained by gortler, have been used to test the accuracy of

two methods which are commonly used to predict separation . a

modification of stratford's criterion for separation is given in this

paper and is probably the most accurate and the simplest of all methods

at present in use . modified numerical functions are also given for

thwaites's method of predicting the main characteristics of the boundary

layer over the whole surface, which improve the accuracy of the method .

.I 462

.T

photo-thermoelasticity .

.A

gerard,g and gilbert,a.c.

.B

j.app.mech. 24, 1957.

.W

photo-thermoelasticity .

this paper summarizes the optical and physical

properties of the photoelastic model material paraplex p-43 over

the temperature range from room temperature to -40 f .

descriptions are presented of techniques and equipment

developed to obtain the modulus of elasticity, the material

fringe value, and the thermal-expansion coefficient as a function of temperature . experimental investigations

were conducted into the plane-stress problems of a disk

contracting upon an elastic inclusion and the transient

thermal-stress field produced by a temperature differential

suddenly applied to the upper edge of a long beam . the

data are correlated with theory using the material

properties obtained in the calibration phase . also included are

photographic results of an exploratory investigation of the

thermal-shock phenomenon produced by the sudden

application of a temperature differential upon plastic beams

of various length-depth ratios .

.I 463

.T

physical properties of plastics for photo-thermoelastic

investigation .

.A

tramposch,h. and gerard,g.

.B

j.app. mech. 25, 1958.

.W

physical properties of plastics for photo-thermoelastic

investigation .

the optical and physical properties of

paraplex p43, castolite, and epoxy resin

hysol 6000-op, which are potentially of

interest in photothermoelastic investigations,

were investigated over a temperature range from +100 to -60 f . results on the

thermal-expansion coefficient, the material

fringe value, and the modulus of elasticity

as functions of temperature are presented .

also evaluated were thermal properties of

importance in heat conduction . photothermoelastic

figures of merit, which rate the

optical sensitivity of materials in photothermoelastic

applications, as well as a new

method to determine this figure in a relative manner are presented .

.I 464

.T

flow studies on flat plate delta wings at supersonic

speeds .

.A

michael,w.h.

.B

naca tn.3472, 1955.

.W

flow studies on flat plate delta wings at supersonic

speeds .

an experimental study has been made to investigate some aspects

of the nature of the flow around delta wings . vapor-screen,

pressure-distribution, and ink-flow studies were made at a mach number of 1.9 on

a series of semispan delta-wing models with slender wedge airfoil

sections and very sharp leading edges . the models had semiapex angles

ranging from 5 to 31.75 .

separated regions of vorticity existed along the chords of all the wings in the series tested . concentrated vortex cores were found only on wings of very small semiapex angles . for wings with medium and large semiapex angles, the separated vorticity was concentrated in a region extending over the outboard part of the span and lying close to the wing upper surface .

the results show that theoretical aerodynamic calculations, such as those in naca tn 3430, utilizing a single, separated vortex pair above the wing upper surface to represent the separated vorticity can be applied at supersonic speeds for very slender wings .

.I 465

.T

slender delta wings with sharp edges at zero lift .

.A

weber,j.

.B

rae tn.aero.2508.

.W

slender delta wings with sharp edges at zero lift .

several slender wings of delta planform with sharp edges have been investigated theoretically at zero lift at subsonic and at supersonic speeds . most of the wings have diamond-shaped cross sections and are intended to lead to a type of flow with leading-edge separation in the lifting condition . the pressure distributions and overall normal-pressure drags resulting from various theoretical methods are compared with one another and some discussion is included concerning the possibility of achieving the results, calculated for an inviscid stream, in a real flow in the presence of a viscous layer around the body .

.I 466

.T

development of the vapour screen method of flow

visualization in the 3ft tunnel at rae bedford.

.A

mcgregor,i.

.B

.W

development of the vapour screen method of flow

visualization in the 3ft tunnel at rae bedford.

the vapour screen method of flow visualisation in supersonic wind tunnels is outlined, and the development of a suitable technique for use in the 3 ft tunnel described, together with the associated optical and photographic equipment .

the results of tests to determine the humidity required to produce an optimum density of fog in the working section over the mach number range temperature discussed . numerous vapour screen photographs of the flow over and behind delta wings are included and some comparisons made with the corresponding surface oil-flow patterns .

the process of condensation, the physical and optical properties of the resulting fog, and the formation of the vapour screen picture are all considered in some detail .

the effects of humidity on the mach number and static pressure in the working section were investigated and the results are compared with theoretical estimates at a nominal mach number of 2.0 . it is shown that the adverse effects of condensation on the flow at high mach

numbers may be alleviated by the use of liquids with a lower latent heat
of evaporation than water, and some results obtained at a mach number of
the possibility of extending the vapour screen technique to transonic
and subsonic speeds is also considered, and some results obtained at a
mach number of 0.85 are included .

.I 467

.T

thin airfoil theory based on approximate solution of the transonic flow
equation .

.A

spreiter, j. r. and alksne, a. y.

.B

naca tn 3970, may, 1957 .

.W

thin airfoil theory based on approximate solution of the transonic flow
equation .

the present paper describes a method for the approximate solution of the
nonlinear equations of transonic small disturbance theory . although
the solutions are nonlinear, the analysis is sufficiently simple that
results are obtained in closed analytic form for a large and significant
class of nonlifting airfoils . application to two-dimensional flows
with free-stream mach number near 1 leads, for instance, to general
expressions for the determination of the pressure distribution on an
airfoil of specified geometry and for the shape of an airfoil having a
prescribed pressure distribution and gives, furthermore, the correct
variation of pressure with mach number at mach number 1 . for flows that
are subsonic everywhere, the method yields a pressure-correction

formula that is more accurate than the prandtl-glauert rule and compares favorably with existing higher approximations . for flows that are supersonic everywhere, the method yields the equivalent, in transonic approximation, of simple wave theory . results obtained by application of these general expressions are shown to correspond closely to existing solutions and to experimental data for a wide variety of airfoils .

.I 468

.T

a refinement of the linearised transonic flow theory .

.A

hosowaka,a.

.B

j.phys.soc. japan, 15, 1960.

.W

a refinement of the linearised transonic flow theory .

a new method is proposed to calculate the velocity and pressure distributions around a thin symmetrical aerofoil or a slender body of revolution flying at transonic speed . it is essentially a refinement of the linearized transonic flow theory due to oswatitsch and maeder, such that a correction term is introduced to take account of the nonlinear character of the transonic flow . as examples of application, a symmetrical circular-arc aerofoil and a circular-arc body of revolution in the sonic flow are dealt with, and the results are found to be in good agreement with experiments, except for the rear portion in the latter case .

.I 469

.T

linearised transonic flow about slender bodies at zero

angle of attack .

.A

maeder,p.f. and thommen,h.u.

.B

asme trans. j.app.mech. 28, 1961.

.W

linearised transonic flow about slender bodies at zero

angle of attack .

the simple linearized transonic flow theory

as originally proposed by oswatitsch and

keune(1) and by the present authors (2)

is improved by considering and partially

correcting its error . in this manner a theory

which is easy to apply and which should

be valid for a great number of smooth bodies

is obtained . this improved theory

predicts shock waves in the lower transonic regions .

it is applied to a number of significant

body and airfoil shapes and its predictions are

compared with experiments and results

of other theoretical investigations .

.I 470

.T

some notes for the small disturbance linear theory

of the method of local linearisation of the flow over

an airfoil at mach number of unity .

.A

miyai,y.

.B

proc. 10th japan nat. cong. app. mech. iii-4, 1960, 207.

.W

some notes for the small disturbance linear theory

of the method of local linearisation of the flow over

an airfoil at mach number of unity .

in this paper, the pressure

distribution at the surface of a symmetrical

non-lifting aerofoil with free stream

mach number of unity has been

investigated by means of the small-disturbance

linear theory or the method of local

linearization . and by comparing with

the calculated results based on an

hodograph method, the accuracy of these

approximate methods has been

evaluated . moreover, when these approximate

methods are used for the calculation

of the pressure coefficient, some notes

necessary to obtain more correct

results have been discussed .

.I 471

.T

.A

.B

.W

.I 472

.T

waves in supersonic flow .

.A

.B

.W

waves in supersonic flow .

in this chapter we shall mainly consider problems of steady, two-

dimensional (plane) supersonic flow . using the fact that in this case

there is a steady wave system, we shall find solutions by an indirect

approach . that is, we shall first study the conditions under which

simple stationary waves may exist in the flow, and then find the flow

boundaries to which they correspond or which may be fitted to

them . in this procedure the limited upstream influence in a supersonic

field is very helpful, for it allows flows to be analyzed or

constructed step by step, which is a method that is not possible in

the subsonic case .

.I 473

.T

freeman method .

.A

.B

.W

freeman method .

the freeman method (ref. 26) is similar to

chester's method in that the newtonian-plus-centrifugal solution (eq.

with the von mises transformation . a method of successive
approximations is applied to both plane and axially symmetric blunt-nosed bodies
for small and infinite free-stream mach number .
formulas for the streamlines, shock shape, and pressure distribution are
determined to this approximation . a number of special shapes are
treated in ref. 26, and in certain cases the theory has a singular point
where the first approximation to the pressure vanishes., that is,
for a sphere (see eq. 7-113) . as in chester's method, the theory is
not applicable where the pressure becomes too small .

.I 474

.T

laminar mixing of a compressible fluid .

.A

dean r. chapman

.B

.W

laminar mixing of a compressible fluid .
a theoretical investigation of the velocity profiles for laminar
mixing of a high-velocity stream with a region of fluid at rest
has been made assuming that the prandtl number is unity . a method
which involves only quadratures is presented for calculating the
velocity profile in the mixing layer for an arbitrary value of the
free-stream mach number .
detailed velocity profiles have been calculated for free-stream mach
numbers of 0, 1, 2, 3, and 5 . for each mach number, velocity profiles
are presented for both a linear and a 0.76-power variation of viscosity
with absolute temperature . the calculations for a linear variation are

much simpler than those for a 0.76-power variation . it is shown that

by selecting the constant of proportionality in the linear approximation

such that it gives the correct value for the viscosity in the

high-temperature part of the mixing layer, the resulting velocity profiles

are in excellent agreement with those calculated by a 0.76-power

variation .

.I 475

.T

the velocity distribution in the laminar boundary layer

between parallel streams .

.A

lock,r.c.

.B

q. j. mech. app. math. 4, 1951, 42.

.W

the velocity distribution in the laminar boundary layer

between parallel streams .

a method is given for obtaining the solution of the laminar boundary

layer equations for the steady flow of a stream of viscous

incompressible fluid over a parallel stream of different density and

viscosity . an approximate solution is also obtained by means of the

momentum equation . it is shown that the solutions depend only on the

ratio of the velocities of the two streams and on the product of the

corresponding density and viscosity ratios . numerical results are

given, in the case where the lower fluid is at rest, for four values of

and also when for one non-zero value of the velocity ratio .

.I 476

.T

the blasius equation with three-point boundary conditions .

.A

napolitano, i. g.

.B

quart. appl. math. v. 16, no. 4, pp 397-408, 1958 .

.W

the blasius equation with three-point boundary conditions .

the blasius equation subject to three-point boundary conditions,

describing the interaction between two parallel streams, is solved by way

of a series in terms of ascending powers of the ratio equals /u1 -dash

u2//u1, where the u1's are the outer streams' velocities .

the first three terms of the series are analytically expressed in terms

of the repeated integrals of the complementary error function /im erfc /

and of the repeated integrals of the square of the successive integrals

of the complementary error function /jmin erfc n/ . these functions

often appear in problems leading to extended heat-conduction type of

equations . a recurrence formula for jmin erfc n is established and

formulae relating the functions in erfc /-dashn/ and jmjn erfc to

available tabulated values of the functions in erfc /n/ are derived .

the first three approximations to the blasius function and to its first

two derivatives are also presented in tabulated form with four

significant figures . test on the convergence of the series has been made by

comparison with some exact solutions obtained by high speed computing

machine . the comparison, extended to the physically essential

quantities, shows that ..

second and first derivatives .

yield extremely accurate results . the errors in the first two

derivatives of the blasius functions are always contained within less than one

per cent .

.I 477

.T

laminar boundary layers at the interface of co-current

parallel streams .

.A

potter,o.e.

.B

q. j. mech. app. math. 10, 1957, 302.

.W

laminar boundary layers at the interface of co-current

parallel streams .

the approximate solution of keulegan(1) for the steady flow of a

stream of viscous incompressible fluid over another at rest is extended

to the case where both fluids are moving co-current but at different

velocities . this solution utilizes a sextic polynomial for the

velocity distribution in the boundary layers . the solutions depend

only on the ratio of the velocities of the two streams and on the

product of the corresponding viscosity and density ratios . numerical

results are given for seven values of at one value of . lock(2) has

published an exact solution with a numerical result for and the sextic

polynomial solution is evaluated f40umerical result for and the sextic

indicates that in general the sextic polynomial is more accurate than

the quartic polynomial but that the advantage is not great .

.I 478

.T

tabulation of the blasius function with blowing and

suction .

.A

emmons,h.w. and leigh,d.c.

.B

arc cp.1913, 1953.

.W

tabulation of the blasius function with blowing and

suction .

authors tabulate solutions of f''' + ff'' = 0 for the velocity

distribution in a boundary layer . for each solution f'(0) = 0,

the third boundary condition is the specification

of f(0) . f(n) and its first three derivatives are tabulated to 5d

in gaps of 0.1 in n for f(0) = -1.23849, -1.2(0.05) 0.5 (0.1) 1.5,

introduction gives method of solution and

physical meaning of boundary conditions, etc . lock's (amr

cussed .

.I 479

.T

on an equation occurring in falkner and skan's approximate

treatment of the equation of the boundary layer .

.A

hartree,d.r.

.B

proc. cam. phil. s. 33, 1937, 223.

.W

on an equation occurring in falkner and skan's approximate

treatment of the equation of the boundary layer .

the differential analyser

has been used to evaluate solutions of the equation y''' = -yy'' +

with boundary conditions y = y' = 0 at x = 0, as

which occurs in falkner and skan's approximate treatment of the

laminar boundary layer (see abstract 1081 (1932)) . a numerical

iterative method has been used to improve the accuracy of the solutions,

and the results show that the accuracy of the machine solutions is about

insufficient to specify a unique solution for negative values of,.

a discussion of this situation is given, and it is shown that for the

application to be made of the solution the appropriate condition is that

from below, and as rapidly as possible, as . the condition that

from below can be satisfied only for values of greater than a limiting

value whose value is approximately -0.199, and which is related to the

point at which the laminar boundary layer breaks away from the

boundary .

.I 480

.T

adiabatic wall temperature due to mass transfer cooling with a

combustible gas .

.A

d. b. spalding

.B

imperial college, london, england

.W

adiabatic wall temperature due to mass transfer cooling with a combustible gas .

a recent technical note by sutton (1), with the above title, discusses

the influence of the burning of a transpiration coolant on the quantity

of coolant necessary to maintain a given wall temperature . the present

note discusses the same problem in a way which has been found useful

in calculating the burning rates of solid and liquid fuels (2) .

consider the transpiration cooling of a porous surface in a gas stream .

then a simple modification of the general mass .

.I 481

.T

mass transfer cooling of a laminary boundary layer

by injection of a light weight foreign gas .

.A

eckert,e., schneider,p., hayday,a. and larson,r.

.B

jet prop. 1958, 34.

.W

mass transfer cooling of a laminary boundary layer

by injection of a light weight foreign gas .

analytical predictions are given for the development of

the velocity, temperature and concentration fields in a

laminar air boundary layer on a flat plate in high-speed

dissipative flow, the plate being considered porous and

cooled by injection of hydrogen from its surface . the

admixture of hydrogen, having a low density and high

thermal capacity relative to air, is shown to greatly

diminish the skin friction and to markedly relieve the adverse thermal effects of intense aerodynamic heating under

conditions of hypersonic flow .

.I 482

.T

a re-examination of the use of the simple concepts

for prediction the shape and location of detached shock

waves .

.A

love,e.s.

.B

naca tn.4170, 1957.

.W

a re-examination of the use of the simple concepts

for prediction the shape and location of detached shock

waves .

a reexamination has been made of the use of simple concepts for predicting the shape and location of detached shock waves . the results show that simple concepts and modifications of existing methods can yield good predictions for many nose shapes and for a wide range of mach numbers .

.I 483

.T

stagnation point shock detachment distance for flow

around spheres and cylinder .

.A

ambrosio,a. and wortman,a.

.B

ars j. 32, 1962.

.W

stagnation point shock detachment distance for flow

around spheres and cylinder .

development of an analytical relation between shock detachment

distance and free-stream mach numbers . results are presented

graphically for shock detachment distance of cylinders and spheres

in air .

.I 484

.T

the influence of two-dimensional stream shear for airfoil

maximum lift .

.A

vidal,r.j.

.B

j. ae. scs. 29, 1962, 889.

.W

the influence of two-dimensional stream shear for airfoil

maximum lift .

the effects of stream velocity gradients on airfoil maximum

lift are defined with experimental data obtained in a simulated

two-dimensional slipstream . the experimental results show

that when positioned near the slipstream plane of symmetry, the

airfoil maximum lift varies markedly with location in the

slipstream . in moving the airfoil from above to below the

slipstream plane of symmetry through a total distance corresponding

to the airfoil thickness, force data and boundary-layer observations show that boundary-layer separation is delayed to higher

angles of attack, and the airfoil maximum lift is doubled .

it is concluded that the destalling effect observed in the

non-uniform slipstream is not associated with slipstream boundary

interference, but stems from the influence of the large local

slipstream shear on airfoil characteristics . the effects of uniform

and nonuniform shear on airfoil lift and pressure distribution are

discussed, within the framework of existing first-order,

small-shear theory, to show that these effects of shear tend to promote

stall . a pohlhausen calculation of the laminar boundary layer

in a stream with shear is used to identify and to assess the effects

of stream shear on boundary-layer separation criteria . it is

demonstrated that these effects are negligibly small, and that the

uniform-flow criterion applies . it is concluded on the basis of

the experimental data that the observed destalling phenomenon

stems from a shear effect of higher order than those treated in the

inviscid theories . it is hypothesized that it is a second-order

effect, fixed by the product of the stream shear and the derivative

of the shear, which was large in the present experiments .

.I 485

.T

linear heat flow in a composite slab .

.A

reid,w.p.

.B

j.ae.scs. 29, 1962.

.W

linear heat flow in a composite slab .

the temperature is determined as a function of position and

time in the case of linear heat conduction in a composite slab of

ture throughout, and the two external surface temperatures are

considered to be prescribed functions .

.I 486

.T

similarity laws for aerothermoelastic testing .

.A

dugundji,j.

.B

j.ae.scs. 29, 1962, 935.

.W

similarity laws for aerothermoelastic testing .

the similarity laws for aerothermoelastic testing are presented

in the range . these are obtained by

making nondimensional the appropriate governing equations of

the individual external aerodynamic flow, heat conduction to

the interior, and stress-deflection problems which make up the

combined aerothermoelastic problem .

for the general aerothermoelastic model, where the model is

placed in a high-stagnation-temperature wind tunnel, similitude

is shown to be very difficult to achieve for a scale ratio other

than unity . the primary conflict occurs between the

free-stream mach number reynolds number aeroelastic

parameter heat conduction parameter and

thermal expansion parameter .

means of dealing with this basic conflict are presented . these include (1) looking at more specialized situations, such as the behavior of wing structures and of thin solid plate lifting surfaces, and panel flutter, where the aerothermoelastic similarity parameters assume less restrictive forms, (2) the use of /incomplete aerothermoelastic/ testing in which the pressure and/or heating rates are estimated in advance and applied artificially to the model, and (3) the use of /restricted purpose/ models investigating separately one or another facet of the complete aerothermoelastic problem .

some numerical examples of modeling for the general aerothermoelastic case as well as for the specialized situations mentioned in (1) above are given .

finally, extension of the aerothermoelastic similarity laws to higher speeds and temperatures is discussed .

.I 487

.T

theory for supersonic two-dimensional, laminar, base-type flows using the crocco-lees mixing concepts .

.A

rom,j.

.B

j.ae.scs. 29, 1962, 963.

.W

theory for supersonic two-dimensional, laminar, base-type flows using the crocco-lees mixing concepts .

a separated flow field, in which the incoming boundary layer is undisturbed up to the separation point, is defined as a /

base-type/ flow . examples are the flows over a blunt base and over

a backward-facing step . the crocco-lees theory is applied to

the supersonic, two-dimensional, laminar, base-type flows

defined above . the separated flow is divided into a mixing region

and a recompression (or reattachment) region . calculations of

base pressure show its dependence on the mach number and on

two reynolds-number-dependent variables, and .

it is shown that existing base-pressure data can be explained

by these results .

.I 488

.T

a reaction-rate parameter for gasdynamics of a chemically

reacting gas mixture .

.A

leonard,m.

.B

j.ae.scs. 29, 1962, 995.

.W

a reaction-rate parameter for gasdynamics of a chemically

reacting gas mixture .

presented note proposes a linearized reaction rate parameter

which is applicable to any reacting gas mixture provided all the

pertinent reactions and their rate constants are known at the

thermodynamic conditions under consideration . linearizing is

achieved by expanding equation of rate of chemical reaction in a

taylor series and neglecting higher-order terms . author announces that tables of linearized reaction rate parameters for

dissociated and slightly ionized air are now in preparation at the

space sciences laboratory, general electric co., msvo .

comparison of preliminary results with exact calculations

published by hall, i. g., et.al., /inviscid hypersonic air-flows with

coupled non-equilibrium processes/ (ias paper 62-67, 30th

annual meeting, new york, jan. 1962) indicates good agreement .

.I 489

.T

on calculation of the laminar separation point and

results of certain flows .

.A

morduchow,m.

.B

j.ae.scs. 29, 1962, 996.

.W

on calculation of the laminar separation point and

results of certain flows .

paper studies compressible laminar boundary layer in adverse

pressure gradient . after mentioning mathematical instabilities in

howarth's and like solutions, authors quote equation from one of

the references, based on the assumptions that zero

heat transfer and y = 1.4 . thence authors compute nondimensional

distances to separation, comparing with solutions by other

workers .

results are interesting, though reviewer feels rather unhappy

about approximations leading to eq. (4),. more detailed justifications should have been given . thus we have the statement

ber, as ./ surely a fuller discussion of effects

of letting is warranted .

typography in eqs. (2) and (3) is rather confusing and there is

a typographical error in heading to table 2 .

.I 490

.T

normal-shock relations in magnetohydrodynamics .

.A

gundersen,r.m.

.B

j.ae.scs.29, 1962, 997.

.W

normal-shock relations in magnetohydrodynamics .

the magnetic-field vector is perpendicular to the flow direction,.

thus for normal shocks there is no change of flow direction through

the shock front . this class of shocks is included in

investigations by several authors (five are referred to here), but the

presentation here is thought to be especially convenient . all

downstream quantities are given in terms of upstream flow conditions,

including the upstream ratio of alfven speed to sound speed, and

the shock strength (density ratio) .

.I 491

.T

on the close relationship between turbulent plane-couette

and pressure flows .

.A

burton,r.a.

.B

j. ae. scs. 29, 1962, 1004.

.W

on the close relationship between turbulent plane-couette

and pressure flows .

author studies the velocity profiles measured by others in plane

and turbulent couette flow, such as is induced in parallel

channels of which one of the walls moves in its own plane . he finds

these profiles to be satisfactorily describable in terms of the

seventh-power law, which was originally set up for plane and

turbulent pressure flow in channels where both walls are stationary .

further, he finds the shear law for pressure flow,

to be applicable also to the couette flow, in a similar range of

reynolds number, r . no attempt is made in this concise

contribution to put these findings on a firmer basis through a theoretical

explanation .

.I 492

.T

prediction of ogive-forebody pressures at angles of attack .

.A

earl r. keener

.B

aerodynamicist, nasa flight research center, edwards, calif.

.W

prediction of ogive-forebody pressures at angles of attack .

various approximations are being suggested for obtaining surface pressures on arbitrary bodies at angle of attack . this not presents a method for obtaining an approximate pressure distribution over the lower surface of an ogive forebody at angle of attack by utilizing the calculated pressures for zero angle of attack .

.I 493

.T

real-gas laminar boundary layer skin friction and heat transfer .

.A

wilson,r.e.

.B

j. ae. scs. 29, 1962, 640.

.W

real-gas laminar boundary layer skin friction and heat transfer .

the laminar-boundary-layer equations have been integrated for the case of a flat plate over a wide range of free-stream enthalpies and velocities and over a wide range of enthalpies of the gas at the wall . the range of free-stream velocities extended up to 25,000 ft sec at low free-stream enthalpies, corresponding to local conditions on a slender body traveling at high speeds . at low free-stream velocities, the range of free-stream enthalpies extended up to 400,000 btu slug, corresponding to the local conditions on a blunt body traveling at speeds up to 25,000 ft sec . the gas was assumed to be in thermodynamic equilibrium at each point in the boundary layer and diffusion effects were neglected . the solutions to the boundary-layer equations were carried out on a high-speed digital computing machine, both skin-friction and

heat-transfer coefficients being obtained from the computations .

before presenting the results, the t' method of rubesin and

johnson for computing skin-friction coefficients for the

perfect-gas case is reviewed . for the real-gas case, the average

temperature, t', is replaced by the average enthalpy, h', and the h'

method is then used to compute skin-friction coefficients . these

values are in excellent agreement with the computing-machine

results . it was found that the recovery factor for the real-gas

case can be approximated by, the best results for the

cases considered being obtained if a value of pr corresponding to

the enthalpy, h', is used . using this recovery factor and

reynolds analogy, heat-transfer rates can be computed which,

with a few exceptions, are within 5 percent of values obtained

from computing-machine results .

.I 494

.T

axisymmetric viscous flow plast very slender bodies

of revolution .

.A

yashura,m.

.B

j. ae. scs. 29, 1962, 667.

.W

axisymmetric viscous flow plast very slender bodies

of revolution .

axisymmetric viscous flow past unyawed very slender bodies of revolution is treated within the category of the perfect gas . attention is paid especially to the effect of transverse curvature of the body . from the transformed equations, the similarity conditions are deduced, and the parameter characterizing the effect of transverse curvature is obtained . several numerical solutions of similarity equations for hypersonic flows are presented, and upon the basis of these results, the effect of the transverse-curvature parameter is discussed . a method of applying the local-similarity approximation to obtain the approximate solution for nonsimilar cases is described, as are practical applications to incompressible flow past a long cylinder and to hypersonic flow past a very slender cone . comparison with experimental results shows fair agreement with calculations using the local-similarity approximation in the present range of experimental flow conditions .

.I 495

.T

on similar solutions for strong blast waves and their application to steady hypersonic flow .

.A

borcher,e.f.

.B

j. ae. scs. 29, 1962, 694.

.W

on similar solutions for strong blast waves and their application to steady hypersonic flow .

the general solution of the strong blast wave is found in the newtonian approximation--i.e., neglecting terms of order

the expressions obtained for the pressure, temperature, density,

and velocity profiles are simple . the results are applied to

power-law bodies in hypersonic flow using the equivalence

principle .

  higher-order approximations for strong blast waves are

investigated for the cases in which the shock layer is thin . a

simple pressure formula is found, which constitutes an

improvement upon the newton-busemann formula, and some of its

applications are shown .

.I 496

.T

a theory of transonic aileron buzz, neglecting viscous

effects .

.A

eckhaus,w.

.B

j. ae. scs. 29, 1962, 712.

.W

a theory of transonic aileron buzz, neglecting viscous

effects .

  usaf-sponsored analysis of the unsteady perturbations of

two-dimensional transonic flow around an airfoil, where local supersonic

regions terminated by shock waves are present in the vicinity of the

airfoil . viscous effects are neglected, and a linearized theory of

the perturbations due to harmonic oscillations of an aileron is

developed . a series solution for the pressure distribution is obtained, and numerical results for the nonsteady hinge moment, from the first approximation to the solution, are presented . as a result of flutter analysis a stability boundary for transonic aileron buzz is obtained . comparison of the theoretical results with experimental observations shows satisfactory agreement .

.I 497

.T

theoretical and experimental investigation of thermal stresses in hypersonic aircraft wing structures .

.A

tramposch,h.

.B

j. ae. scs. 29, 1962.

.W

theoretical and experimental investigation of thermal stresses in hypersonic aircraft wing structures .

a simple and relatively accurate analytic approximation is developed to determine the temperature and thermal-stress distribution in aircraft wing structures . theoretical investigations show that the results of the existing thermal-stress theories which neglect the temperature gradient through the skin thickness may exceed, in the range of higher biot numbers, the true values by more than 30 percent .

refined photothermoelastic experiments verify these results and add another significant conclusion . they indicate that thermal stresses in wing structures generated by a variable heat-transfer coefficient coincide with the theoretical predictions

which are based on a constant heat-transfer coefficient, as long as the latter represents the arithmetic average over the heating cycle and the variation is in the order of 10 percent .
however, even much greater variations in the order of 100 percent produce only relatively small differences .

.I 498

.T

calculation of potential flow about bodies of revolution having axes perpendicular to the free-stream direction .

.A

hess,j.l.

.B

j. ae. scs. 29, 1962.

.W

calculation of potential flow about bodies of revolution having axes perpendicular to the free-stream direction .

a general method is described for calculating, with the aid of an electronic computer, the potential flow about arbitrary bodies of revolution whose axes are perpendicular to the free-stream direction . when combined with the solution for the axisymmetric flow about these bodies, this method makes it possible to calculate the pressure distribution on any body of revolution at angle of attack forward of any separated region of the flow, and also to calculate the flow at points off the body surface . after the basic equations of the method have been derived, its accuracy is exhibited by comparison with analytic solutions for ellipsoids of revolution . calculated pressure distributions are then compared with experimental data for a variety of bodies . the

agreement is quite satisfactory in all cases . the calculated

velocities for other selected bodies are presented to exhibit

certain properties of this type of flow .

.I 499

.T

a closed-form solution for the oscillations of a vehicle

entering a planetary atmosphere .

.A

greensite,a.l.

.B

j. ae. scs. 29, 1962, 745.

.W

a closed-form solution for the oscillations of a vehicle

entering a planetary atmosphere .

author considers the equation of the yawing motion of a missile,

derived with a series of customary assumptions and with the

distance traveled as the independent variable . his assumptions

include the linearity of the aerodynamic forces, the constancy of the

aerodynamic coefficients with respect to mach number, the

absence of spin, and the absence of gravity . if to these assumptions

one could add the common ballistic assumption of a constant air

density, the coefficients of this equation would have been

con-damped sinusoids . in ballistics any slow variation of these

coefstant, and the solution would have been simply the

exponentially-ficients is usually treated by adding an approximate correction

term to the damping rate (which is spoken of as the wkb perturbation) . however, with a body entering the planetary atmosphere the variation of the air density is apparently of greater essence (this is a point not stated explicitly in this brief communication), and the equation is of the type .

the author shows that with a series of further transformations the equation can be reduced to the form

the solutions of which are confluent hypergeometric functions . these functions are defined as series involving gamma functions, and with a series of further assumptions can be reduced to laguerre polynomials and bessel functions .

it is certainly nice to have an exact solution to a problem which has heretofore been extensively treated by approximations and by the numerical approach . this reviewer is puzzled, however, as to the practical significance of the proposed approach . an idealization is of value in that it facilitates our understanding,. and the numerical approach, in that it allows refinements of the problem, freeing us from the necessity of idealizing . but the proposed solution is certainly more difficult to refine than the original problem,. and it is certainly not simple (the solution of the original equation is not the value of z, but the various /reverse/ transformations of z) . an evaluation of a series in practice must compete with the numerical approach,. and the equation suggested is of the zero) . viewing the problem /afresh/ (in the light of the / computer revolution/ and without the constraints imposed by the prior art), it seems at least equally easy to /standardize/ the solutions of the original equation .

.I 500

.T

joule heating in magnetohydrodynamic free-convection

flows .

.A

cramer,k.r.

.B

j.ae.scs., 29, 1962, 746.

.W

joule heating in magnetohydrodynamic free-convection

flows .

the steady, fully developed, laminar, free-convection flow of an

electrically conducting fluid between two fully submerged

open-ended, constant-temperature vertical plates located in a constant,

uniformly distributed, transverse magnetic field has been analyzed

with the joule heating term retained in the energy equation .

analytic results are obtained . such analytic results are useful in

estimating the actual magnitude of the influence of joule heating as

well as a qualitative description of the manner in which it alters

the temperature and flow fields . the present result confirms the

usual practice that the influence of joule heating is negligibly

small .

.I 501

.T

stagnation-point shock detachment of blunt bodies in

supersonic flow .

.A

ridyard,h.w.

.B

j. ae. scs. 29, 1962, 751.

.W

stagnation-point shock detachment of blunt bodies in

supersonic flow .

presentation of stagnation-point shock-detachment distances

determined by the exact numerical method of gravalos, edelfelt,

and emmons . the results are compared with those from the

previously published methods of van dyke and gordon, li and geiger,

and serbin, and with experimental data .

.I 502

.T

on squire's test of the compressibility transformation .

.A

mager,a.

.B

j. ae. scs. 29, 1962,752.

.W

on squire's test of the compressibility transformation .

discussion of a previous application, by squire, of the author's

compressibility transformation to the correlation of high-speed

boundary-layer data for air and helium . squire's suggestion that

the compressibility transformation is invalid is shown to be

incorrect .

.I 503

.T

theoretical prediction of the transonic characteristics of airfoils .

.A

sinnott,c.s.

.B

j. ae. scs. 29, 1962, 275.

.W

theoretical prediction of the transonic characteristics

of airfoils .

it is shown that the author's transonic-flow airfoil theory

can be used to estimate transonic drag-rise and onset-of-

separation-effects mach numbers without reference to experimental

results . a simple comparative method is applied to a series of

airfoils, and the results are analyzed to determine some of the

design features of importance in transonic flow . an

improvement to this scheme is shown to give results in good agreement

with experiment for both the first appearance of shock waves

and the onset of separation effects . application to finite swept

wings is briefly considered and illustrated .

.I 504

.T

stability of compressible boundary layers induced by

a moving wave .

.A

ostrach,s.

.B

j. ae. scs. 29, 1962, 289.

.W

stability of compressible boundary layers induced by
a moving wave .

the problem of determining the stability of compressible viscous flows with nonzero surface velocities is formulated and is shown to be identical to that for conventional boundary layers, with only a redefinition of the mach and reynolds numbers required . specific consideration is given to the wall boundary layer behind a moving shock wave, and the minimum critical reynolds numbers are obtained for various shock velocities . the entire stability map is determined for the limiting case of a weak wave, which is analogous to the rayleigh problem .

the minimum critical reynolds number is found to increase monotonically with shock velocity--i.e., with increasing surface cooling and stream mach number combined . for the ratio of wall to stream velocity of 2.92 with (shock mach number of 2.18) the flow is found to be infinitely stable to two-dimensional disturbances .

experimental transition data do not follow the trends predicted by the theory . in fact, the transition reynolds numbers are orders of magnitude below the computed minimum critical reynolds numbers . the lack of correlation between theory and experiment is attributed to disturbances which are external to the boundary layer .

.I 505

.T

transition measurements on cones in free flight ballistics
range tests .

.A

lyons,w.c.

.B

j. ae. scs. 29, 1962, 352.

.W

transition measurements on cones in free flight ballistics

range tests .

navy-sponsored experimental investigation of the location of

boundary-layer transition on sharp-nosed cones having 10 total

angles . the ambient temperature in a portion of the aeroballistics

range is varied so as to obtain different adiabatic recovery

temperatures at a constant nominal mach number of 3.1 . the location

of transition is expressed as a transition reynolds number, and

results are presented graphically as a function of the ratio between

the wall temperature and the adiabatic recovery temperature .

.I 506

.T

a note on havelock's shallow-water wave-resistance

curves .

.A

brandmaier,h.e.

.B

j. ae. scs. 29, 1962, 257.

.W

a note on havelock's shallow-water wave-resistance

curves .

in the continuous quest for improved means of transportation, attention is currently focused on the ground-effect

machine . as there is no physical contact between the vehicle

and the terrain over which it operates, its performance should

be similar over land and water . however, over water there is

an additional resistance to motion due to the gravity-wave

system generated by the supporting or /cushion/ pressure acting

on the water surface . estimates of this component can be made

using the analysis of t. h. havelock . it is the purpose of this

note to present an ibm 650 digital-computer solution of his

equations . as shown below, these results differ from havelock's

original results .

.I 507

.T

energy equation approximations in fluid mechanics .

.A

goldstein,a.w.

.B

j. ae. scs. 29, 1962,358

.W

energy equation approximations in fluid mechanics .

discussion of several forms of the energy equation and of their

use for the study of the flow of nearly incompressible fluids .

.I 508

.T

a correlation of nose-bluntness induced pressures on

cylindrical and conical after-bodies at hypersonic

speeds .

.A

greenberg,r.a.

.B

j. ae. scs. 29, 1962, 359.

.W

a correlation of nose-bluntness induced pressures on

cylindrical and conical after-bodies at hypersonic

speeds .

van hise, in his detailed study of the nose-bluntness-induced

pressures on cylindrical afterbodies, shows that, starting

a few nose diameters aft of the nose-afterbody junction, these

pressures are correlated with the parameter

as predicted by the blast-wave analogy . chernyi developed a

modified form of the blast-wave analogy which takes into account

the addition of energy to the flow by a thin afterbody . he

showed that for thin afterbodies and hypersonic speeds, the

pressure distribution, plotted as should correlate with

the parameter . the purpose of this note

is to show that the above correlation techniques may be

combined into a form such that pressures on cylindrical and conical

afterbodies are correlated by one parameter .

.I 509

.T

a graphical approximation for temperatures and sublimation rates at

surfaces subjected to small net and large gross heat transfer rates .

.A

adams, e. w.

.B

j. aero. sc. v. 29, march 1962, pp. 360-1 .

.W

a graphical approximation for temperatures and sublimation rates at

surfaces subjected to small net and large gross heat transfer rates .

considers a material, acted upon by heat of conduction, which changes

its state by sublimation at the heated surface . the derived method is

most suitable under conditions of severe heating such as space vehicle

re-entry .

.I 510

.T

manoeuvring technique for changing the plane of circular

orbits with minimum fuel expenditure .

.A

weiss,d.c.

.B

j. ae. scs. 29, 1962, 368.

.W

manoeuvring technique for changing the plane of circular

orbits with minimum fuel expenditure .

usaf-supported discussion of the use of an intermediate

elliptic orbit for changing the plane of a circular orbit . values of the

perigee and apogee velocities are calculated for the following cases ..

the braking impulse supplied by grazing of the atmosphere,. and (3)

re-orbit with 90 of the braking impulse supplied in this manner .

.I 511

.T

tunnel tests on a double cascade to determine the interaction

between the rotor and the nozzles of a supersonic turbine .

.A

stratford,b.s.

.B

ngte m359.

.W

tunnel tests on a double cascade to determine the interaction

between the rotor and the nozzles of a supersonic turbine .

experimental confirmation has been required that in a supersonic turbine the leading edges of the rotor governs the rotor incidence and, hence, the gas exit angle from the nozzles . evidence has also been required that, once the rotor incidence has been allowed for, there is no adverse effect of the rotors on the nozzle flow, even when the rotors have a large turning angle .

the present test cascade represented the stationary configuration of a turbine of 2.5 nozzle mach number and 74 swirl angle, the rotors being designed to operate at 1.9 relative mach number and to provide a turning angle of 140 . in the tests, fully supersonic flow could be established through the system, but the losses were fairly high and an increase in loss of about 25 per cent would have caused choking in the rotor .

.I 512

.T

quasi-cylindrical surfaces with prescribed thickness

distributions .

.A

moore,k.c.

.B

rae tn.aero.2815, 1962.

.W

quasi-cylindrical surfaces with prescribed thickness

distributions .

a formula for the supersonic velocity field in terms of a given

surface distribution of sources is applied to points lying in the

surface . an equation giving the camber shape of a quasi

circular-cylindrical surface in terms of a prescribed thickness distribution

is derived and the half ring wing with prescribed thickness distribution

is discussed as an example .

.I 513

.T

pressure measurements at supersonic speeds on three

uncambered conical wings of unit aspect ratio .

.A

britton,j.w.

.B

rae tn.aero.2821, 1962.

.W

pressure measurements at supersonic speeds on three

uncambered conical wings of unit aspect ratio .

pressure measurements were made at mach numbers between 1.3 and 2.8

over a range of incidences on three simple models representing thick

conical uncambered wings with sharp leading edges . these tests form

part of an investigation into the effects of thickness and camber on slender wings .

the aspect ratio of the models was unity in each case, and the

spanwise cross sections were bounded

by .. the measured pressure distributions are presented, along with overall

lift and drag (excluding skin friction and base drag) obtained by

integration .

.I 514

.T

pressure distributions and flow patterns on some conical

shapes with sharp edges and symmetrical cross-sections

at m=4 .0.

.A

squire,l.c.

.B

rae tn.aero.2823, 1962.

.W

pressure distributions and flow patterns on some conical

shapes with sharp edges and symmetrical cross-sections

at m=4 .0.

results are given of a wind tunnel programme made to study the

pressure distributions and flow patterns over a series of simple,

conical shapes at a mach number of 4.0 . the results have been compared

with various approximate theories and the limitations of these theories

are discussed .

it is found that at this mach number leading edge separations still

have an influence on the suction surface pressure, and that this surface

still makes a significant contribution to the overall forces .

.I 515

.T

self sustained oscillations of a system with non-linear

damping of a particular type .

.A

neumark,s.

.B

rae tn.aero.2839, 1962.

.W

self sustained oscillations of a system with non-linear

damping of a particular type .

the paper deals with self-sustained

oscillations of a dynamic system

of single degree of freedom, with linear

restoring force and non-linear

damping force . the latter is supposed

to be a function of velocity

representable by a simple /polygonal/

graph, such that the damping is

negative at small velocities but becomes

positive at velocities above a

certain value . on these assumptions,

a rigorous solution is presented,

including the equations of motion,

amplitude, maximum velocity and period .

a very simple solution is obtained

for the limiting case of vanishingly

small damping . an approximate solution

by series in powers of damping

ratio is worked out which

gives a satisfactory accuracy for

quite large values of .

.I 516

.T

free-flight measurements of the dynamic longitudinal

stability characteristics of a wind tunnel interference

model (m=0 .92 to 1. 35) .

.A

greenwood,g.h.

.B

rae tn.aero.2798, 1961.

.W

free-flight measurements of the dynamic longitudinal

stability characteristics of a wind tunnel interference

model (m=0 .92 to 1. 35) .

the dynamic longitudinal-stability characteristics of a standard

wind tunnel interference model have been investigated in free flight

over a mach number range of 0.92 to 1.35 .

measurements of lift-curve slope and manoeuvre margin were

obtained, and are compared with results from transonic-tunnel tests

under low blockage conditions .

the analysis was extended to obtain damping derivatives to allow

comparison to be made with possible future dynamic tests in wind tunnels

on the standard shape .

.I 517

.T

reaction-resisted shock fronts .

.A

clarke,j.f.

.B

coa r.150, 1961.

.W

reaction-resisted shock fronts .

it is shown that shock waves whose

structure is determined solely by the effects

of chemical reactions (reaction-resisted

shock fronts) are possible and completely

analogous to relaxation - resisted waves .

a single dissociation reaction is considered

and numerical results indicate that such

waves could be observed experimentally .

bulk viscosities equivalent to reaction

effects are possibly 10 or more times shear

viscosity values . (examples are based on

lighthill's ideal dissociating gas) .

.I 518

.T

heat conduction through a polyatomic gas .

.A

clarke,j.f.

.B

coa r.149, 1961.

.W

heat conduction through a polyatomic gas .

a heat conduction problem is set up which, in essence, simulates the conditions arising when a plane shock wave reflects from a co-planar solid boundary . the gas is assumed to be polyatomic, with one

the quantity of primary interest is the temperature of the solid at the interface, since this can be observed experimentally without much difficulty . solutions are obtained for this quantity which cover a range of practically plausible relaxation times and 'wall effect' parameters . it is essential to include proper temperature jump boundary conditions for both active and relaxing (or inert) energy modes . thus it is necessary to know accommodation coefficients for these modes of energy storage .

the temperature jump effects are found to dominate the (interface) solid's temperature time history, with relaxation effects playing a very secondary role .

the theoretical results are compared with some experimental observations and encouraging agreement is found . as a result of this agreement it proves possible to

estimate the accommodation coefficient

for the active modes (in this case for the

combination platinum air), the pressure

being about 15 atmospheres . the pressure

sensitivity of accommodation effects is commented on .

.I 519

.T

base pressure at supersonic speeds in the presence

of a supersonic jet .

.A

craven,a.h.

.B

coa r.144, 1960.

.W

base pressure at supersonic speeds in the presence

of a supersonic jet .

the effects on base pressure of jet mach number, free stream

reynolds number and jet to base diameter ratio have been investigated

experimentally .

it was found that, for jet stagnation pressures greater than that

required for the nozzle to reach its design mach number, an increase

of jet mach number reduced the base pressure . similarly the base

pressure increased with increase of the ratio of jet diameter to base

diameter and, at high jet stagnation pressures, base pressures higher

than free stream static pressure were found . the base pressure was

independent of free stream reynolds numbers greater than 2 x 10 per foot but increased with reduction of reynolds number below 2 x 10 per

foot .

unsteady wave patterns were found when the jet mach number did

not differ markedly from the free stream mach number and the jet had

just reached its design conditions .

.I 520

.T

wing-tail interference as a cause of 'magnus' effects

on a finned missile .

.A

benton,e.r.

.B

j. ae. scs. 29, 1962, 1358.

.W

wing-tail interference as a cause of 'magnus' effects

on a finned missile .

wing-tail interference is shown to cause large /magnus/ effects

on a finned missile whose wings are deflected into an aileron

setting . a simple experimental method with water as the

working medium is used to obtain low-speed magnus data on a rolling

missile . the missile is a slender cruciform configuration with

all-movable wings and fixed tail fins . magnus data are

presented for angles of attack up to 15 and for the one (high) roll

rate which accompanies a 30 aileron deflection angle of the

wings . tests conducted at zero roll rate but with the wing

deflection maintained, revealed large forces in the magnus direction,

thereby providing the basis for understanding magnus effects due to wing-tail interference .

a semiempirical theory is proposed to explain the experimental

data . a simplified model of the wake behind the wings is

introduced to predict tail-interference factors . good agreement with

the data is obtained .

this magnus effect is opposite in direction to the classical

magnus lift on a spinning cylinder ,. it is much larger than either

that effect or the one on a missile with only one set of fins .

wing-tail interference is the predominant source of the effect ,. roll rate

only modifies the basic interference mechanism .

.I 521

.T

a note on application of transonic linearization to

an airfoil with a round leading edge .

.A

hosokawa,i.

.B

j. ae. scs. 29, 1962, 1395.

.W

a note on application of transonic linearization to

an airfoil with a round leading edge .

the profile of a symmetric airfoil of unit length with a round

leading edge can be expressed, in general, as

where p(x) has a finite slope at x = 0 . it is well known that

the conventional sub- and supersonic linear theories of

compressible flow break down in the neighborhood of such a round

leading edge due to the failure of the small-disturbance assumption . the linearized transonic flow theory has the same

short-coming, but if the determination of the sonic point on the airfoil

plays an important role in any more advanced theory--e.g.,

spreiter's local-linearization method or hosokawa's method of

refinement--this theoretical barrier will become more serious

because the sonic point is usually located in a flow region near

the leading edge that may be greatly affected by the roundness .

.I 522

.T

laminar, transitional and turbulent heat transfer to

a cone-cylinder-flare body at mach 8. 0.

.A

zakkay,v. and callahan,c.j.

.B

j. ae. scs. 29, 1962, 1403.

.W

laminar, transitional and turbulent heat transfer to

a cone-cylinder-flare body at mach 8. 0.

an experimental investigation of the laminar, transitional, and

turbulent heat transfer rates over a conical cylindrical flared

body is presented . regions of favorable, zero, and adverse

pressure gradient on the body are investigated . the experimental

results are compared with the theories available in the literature .

the model chosen for this investigation is a cone-cylinder-flare

configuration consisting of a 20 semivertex conical nose portion

smoothly blended by a shoulder radius into a long cylindrical body

and terminated by a smooth large radius flare .

the model was tested at a free stream mach number of 8, over a range of reynolds number from 0.3 x 10 to 1.6 x 10 per inch based on free stream conditions . various stagnation-to-wall temperature ratios were obtained by cooling the model prior to the test with liquid nitrogen . the stagnation-to-wall temperature ratios were 10 and 3.3 .

the theoretical predictions gave good results for the heat transfer rates in the laminar region, and fair prediction in the transitional and turbulent regimes extending over the shoulder and forward portion of the cylindrical body . over the aft portion of the cylinder and over the flare the predictions are only qualitatively correct, and underestimate the heating rate by a factor as high as 3 . conversely, the /flat plate reference enthalpy/ over the aft portion of the body, but to increasingly overestimate the heating rates over the forward portion of the cylinder .

a modified equation for the heat transfer coefficient in the transitional and fully turbulent region based on the f.p.r.e. method is then presented . this method gives good agreement with the experimental results presented over the entire range of transitional and turbulent flow .

from the results the following is concluded .. cooling the wall delayed transition . by expanding the flow rapidly between the cone and the cylinder, the transition reynolds number is reached very rapidly . by making a smooth transition between the cylinder and the flare, no separation occurred at the cylindrical flare junction . the transitional and turbulent heat transfer in the presence of an adverse pressure gradient may be predicted with sufficient accuracy by the f.p.r.e. method .

.I 523

.T

approximate determination of position of the sonic

line for a blunt body in hypersonic flow .

.A

rahman,m.a.

.B

j. ae. scs. 29, 1962.

.W

approximate determination of position of the sonic

line for a blunt body in hypersonic flow .

the detached shock in front of a blunt body in hypersonic

flow tends to acquire the shape of the frontal curvature of the

body . thus the curvature of the shock can be assumed to be the

same as that of the body, at least up to the sonic point (point a,

fig. 1) . if the equation of curvature of the body is known, the

equation of curvature of the shock is also known . in this paper,

with this assumption, a method is described to determine the

approximate position of the sonic line (ao'b, fig. 1) . the

shock-detachment distance is assumed known .

the method is, of course, general . this can be applied to

any detached shock provided its equation of curvature is known

corresponding to that of the body . for simplicity the detached

shock is assumed to be circular in this paper and the procedure is

outlined below with the assumption that the sonic line ao'b is

parabolic .

.I 524

.T

stagnation point heat transfer in partially ionized

air .

.A

rozycki,r.c. and fenster,s.j.

.B

j. ae. scs. 29, 1962.

.W

stagnation point heat transfer in partially ionized

air .

comparison of heat-transfer rates, obtained by using transport

properties recently reported by peng and pindroh, with rates based

on hansen's thermodynamic and transport properties . it is shown

that the heat-transfer rates based on the peng and pindroh data

are 20 to 30 lower for the velocity range of 25,000 to 40,000

ft sec .

.I 525

.T

on hypersonic viscous flow over an insulated flat plate

with surface mass transfer .

.A

tien,c.l.

.B

j. ae. scs. 29, 1962, 1024.

.W

on hypersonic viscous flow over an insulated flat plate with surface mass transfer .

hypersonic viscous flow over an insulated flat plate with

surface mass transfer is studied . the tangent-wedge approximation

is used in the inviscid-flow region, and the integral method is

applied to the treatment of the laminar boundary layer . the law

of surface mass transfer for the present analysis is derived . a

continuous transition of the pressure variation is achieved from

the strong to the weak pressure-interaction region . first-order

formulas for the induced surface pressure and the skin-friction

coefficient are obtained for both the strong and weak

pressure-interaction regions . results are compared with those calculated

from other analyses .

.I 526

.T

leading edge attachment in transonic flow with laminar

or turbulent boundary layers .

.A

mabey,d.g.

.B

j. ae. scs. 29, 1962.

.W

leading edge attachment in transonic flow with laminar

or turbulent boundary layers .

the transonic flow round a two-dimensional airfoil at

incidence is often determined by the type of flow in the leading-edge

region . if the flow separates at the leading edge at low speeds it

is liable to attach as the speed increases, often quite suddenly .

a review of this change with laminar or with turbulent boundary

layers re-emphasizes the importance of fixing transition when

making model tests at transonic speeds in order to obtain flows

closest to full-scale conditions .

it is shown that similar airfoils with attached leading-edge

flow show transonic similarity upstream of the terminal shock .

.I 527

.T

note on the three-point boundary layer problem for

the blasius equations .

.A

martin,e.d.

.B

j. ae. scs. 29, 1962.

.W

note on the three-point boundary layer problem for

the blasius equations .

in a recent paper a method was presented for obtaining

higher accuracy in the numerical solution of the blasius

equation with three-point boundary conditions . the

well-known blasius equation was previously developed in an

investigation of the steady two-dimensional incompressible

boundary-layer flow over a flat plate, but it has been extensively used in

investigating other fluid flow problems . the three-point

boundary-value problem is encountered in the theory of laminar

mixing and in approximate analyses of separated and wake

flows as noted in ref. 1 .

.I 528

.T

first-order slip effects on the laminar boundary layer over a slender

body of revolution with zero pressure gradient .

.A

jay m. solomon

.B

research aerospace engineer, u. s. naval ordnance laboratory,

white oak, silver spring, md.

.W

first-order slip effects on the laminar boundary layer over a slender

body of revolution with zero pressure gradient .

in reference 1, the analysis given by probstein and elliott for the

zero-pressure-gradient, constant-wall-temperature, compressible,

laminar boundary layer with transverse curvature was extended to

first-order slip flow . this extension was based on a double asymptotic

expansion in a transverse-curvature parameter and a slip parameter .

the expansion in ref. 1, however, was carried out with the parameter

held constant . for and a constant wall temperature, is constant and

e varies with x due to the dependence of the local body radius on x .

thus, for arbitrary body shapes, e will not be constant . in the

present note, the analysis of ref. 1 is re-examined taking into account

the variation of e .

.I 529

.T

some effects of injection of foreign gases in a decelerating

laminar boundary layer in supersonic flow .

.A

gouse,s.w., brown,g.a. and kaye,j.

.B

j. ae. scs. 29, 1962, 1250.

.W

some effects of injection of foreign gases in a decelerating

laminar boundary layer in supersonic flow .

the purpose of this research program was to investigate the

effects of a diffusion field on a laminar boundary layer in a

supersonic flow . specifically, helium, nitrogen, and argon were

uniformly injected into the laminar boundary layer of a

high-speed flow in a tube with the objective of determining the effects

of such injection on the pressure, temperature, and recovery

factor distribution along and downstream of the injection region .

a continuously operating axially-symmetric wind tunnel has

been designed, constructed, and operated . this tunnel consists

of an air supply system, a flowmeter, an upstream stagnation

tank, a supersonic nozzle (throat diameter 0.262 and exit

diameter 1.400), a test section of variable length (zero to 81 diameters,

test section diameter of 1.400), a downstream stagnation tank,

an exhaust system, a foreign gas supply system, and all necessary

instrumentation . the overall performance of this apparatus in

terms of the design specifications was excellent .

the tunnel was instrumented with 109 thermocouples . all

temperatures except ambient temperatures were automatically

measured and recorded by means of a self-balancing recording

potentiometer . there was 29 pressure taps distributed along the tunnel, 23 along the test section itself . pressures were measured

by means of an interconnected micromanometer and a vacuum

referenced manometer system with overlapping ranges .

for all of the results reported herein, the overall test section

was 41 diameters in length,. composed of a porous test section

approximately 7.2 diameters in length (leading edge

approximately 1.8 diameters from the nozzle exit plane) and four nylon

test sections of 8 diameters each .

.I 530

.T

an aerodynamic analysis for flutter in oseen-type viscous

flow .

.A

chu,wen-hwa.

.B

j. ae. scs. 29, 1962, 781.

.W

an aerodynamic analysis for flutter in oseen-type viscous

flow .

oseen's equations for unsteady flow are employed to obtain

a linearized solution based on a discontinuous-wake model .

the analysis is employed to estimate the viscous correction to

unsteady lift and moment at large reynolds number . if the

asymptotic solution is not too slowly convergent, the correction

is of the order of the ratio of the logarithm of reynolds number

to the reynolds number . the theory is preliminary in nature

as it is limited by the accuracy of oseen's equations and is

restricted to small angle of attack . however, it also shows that
the generalized trailing-edge condition for potential flow is
reasonable and might predict the essential correction in a real
fluid .

.I 531

.T

the flow about a moving body in the upper ionosphere .

.A

bird,g.a.

.B

j. ae. scs. 29, 1962.

.W

the flow about a moving body in the upper ionosphere .

a particle approach is used to study the flow pattern around a
body moving in the upper layers of the ionosphere . the effects
of distant encounters between charged particles (dynamic
friction) and of the earth's magnetic field are taken into account .
it is shown that, when the magnetic lines of force are parallel
to the direction of motion of the body, there may be a marked
concentration of charged particles in the vicinity of the body
and a considerable fraction of the reflected or deflected charged
particles may reimpinge on the body surface . a numerical
example is given for the size and shape of the charged-
particle-density contours in the flow field surrounding a circular disc,
and these are compared with the corresponding neutral-particle
contours .

.I 532

.T

pitch-yaw stability of a missile oscillating in roll

via the second method of lyapunov .

.A

parks,p.c.

.B

j. ae. scs. 29, 1962, 874.

.W

pitch-yaw stability of a missile oscillating in roll

via the second method of lyapunov .

the stability theory of a. m. lyapunov, a popular topic in

the u.s.s.r., is receiving increasing attention elsewhere .

this note describes lyapunov's /second method/ very briefly

and applies it to an aeronautical stability problem .

.I 533

.T

stagnation-point shock-detachment distance for flow

around spheres and cylinders in air .

.A

ambrosio,a. and wortman,a.

.B

j. ae. scs. 29, 1962.

.W

stagnation-point shock-detachment distance for flow

around spheres and cylinders in air .

author discusses the problem of deflection of a cantilevered

bar, initially in the shape of a circular arc, subjected to an arbitrarily inclined end load .

.I 534

.T

consideration of energy separation for laminar slip

flow in a circular tube .

.A

inman,r.m.

.B

j. ae. scs. 29, 1962, 1014.

.W

consideration of energy separation for laminar slip

flow in a circular tube .

the energy separation for laminar low-density-nonunity prandtl

number flow in circular cross-section tubes is the topic of this

note . a conclusion is reached as to the effect of prandtl number

on the velocity profiles for these flows . however, in order to

reach valid quantitative conclusions the reviewer feels that more

detailed analysis is in order, and that the analysis as presented

here is of qualitative value only .

.I 535

.T

shroud design for simulating hypersonic flow over the nose of a

hemisphere .

.A

roger dunlap

.B

associate research engineer, dept, of aeronautical and astronautical engineering, ann arbor, mich.

.W

shroud design for simulating hypersonic flow over the nose of a

hemisphere .

following is an analytical method for designing a shroud which will generate the hypersonic pressure distribution on a hemisphere . the method was found to be successful throughout the region of subsonic flow . this shroud was designed as part of a low-turbulence wind tunnel used for investigating the effects of cooling on boundary-layer transition on a hemisphere .

the design of the shroud contour was carried out in two steps . first, an approximate solution for the incompressible, irrotational flow field was found in the region, and, second, the resulting contour was corrected for compressibility near the sonic region, assuming one-dimensional flow .

.I 536

.T

transition in the viscous wakes of blunt bodies at

hypersonic speeds .

.A

hidalgo,h., taylor,r.l. and keck,j.c.

.B

j. ae. scs. 29, 1962.

.W

transition in the viscous wakes of blunt bodies at

hypersonic speeds .

transition from laminar to turbulent flow in the hypersonic wakes of spheres was detected in laboratory measurements of the radiation from the flow field . a hypervelocity gun facility was used to fire models, 0.22-in. in diameter, into a range at velocities from 10,000 to 17,000 ft sec . experiments were performed by changing .. (a) the material of the projectile ,. (b) the ambient gas in the range ,. and (c) the pressure in the range . three optical techniques were used to observe the wake radiation .. which show a turbulent viscous wake as the pressure in the range is decreased from one atmosphere to about 20 cm hg . which show the luminous flow field at pressures between 30 and ence of short luminous streaks, which disappear suddenly as the pressure is decreased below 3 cm hg for air, and below 0.8 cm hg for argon .

both air and argon, which show the main features of the flow field . above the transition pressure, the intensity of radiation from the wake is always associated with fluctuations that appear to be the same phenomenon as the drum-camera streaks .

the appearance of the streaks in the drum camera and photo-multiplier data is interpreted as transition from laminar to turbulent flow in the viscous wake, because experimental evidence shows that their appearance is not controlled by chemical, radiative, or ablative processes, but depends on aerodynamic effects . this conclusion is supported by other experiments based on optical and schlieren techniques . the transition in the wake at positions very close to the body is given by a local reynolds number of 10 for air, and 3 x 10 for argon . the results indicate a possible local-mach-number effect .

.I 537

.T

stagnation point viscous hypersonic flow .

.A

stoddard,f.j.

.B

j. ae. scs. 29, 1962, 1138.

.W

stagnation point viscous hypersonic flow .

several methods have been developed for computing the

hypersonic, low reynolds number flow in the stagnation

region of a blunt body . in general, these methods involve

complicated numerical solutions . simultaneous iterations on several

parameters are usually required in view of the boundary-value

nature of the problem .

the purpose of this note is to present an approximate

closed-form solution to axisymmetric stagnation point hypersonic flow

in the viscous layer regime .

.I 538

.T

the conpressibility transformation and the turbulent

boundary layer equations .

.A

burgraf,o.r.

.B

j. ae. scs. 19, 1962.

.W

the conpressibility transformation and the turbulent

boundary layer equations .

the compressibility transformation first introduced by

dorod-nitzyn has been applied in this paper to the equations of the

turbulent boundary layer on a flat plate, considering heat transfer

and arbitrary prandtl numbers . assuming the shear

distribution to be invariant under the transformation, the stream

function and the momentum equation take the proper form for

incompressible flow, allowing the use of incompressible velocity

profiles in the transformed coordinates . application of crocco's

method to the transformed energy equation permits integration

of the energy equation resulting in a formulism remarkably

similar to that proposed by eckert . finally, the reference

condition was chosen to correspond to the edge of the sublayer from

considerations of the assumptions made regarding the

shear-stress distribution . with this choice, the reference enthalpy is in

good agreement with eckert's formula over the ordinary range of

test conditions . in view of these results, the analysis may be

considered to provide a theoretical basis for the

reference-enthalpy method .

.I 539

.T

local heat transfer to a yawed, infite, circular cylinder

in laminar compressible flow .

.A

weiss,d.

.B

j. ae. scs. 29, 1962.

.W

local heat transfer to a yawed, infite, circular cylinder

in laminar compressible flow .

this note presents a simplification of a method for

calculating the ratio of local to stagnation-line heat-transfer

coefficients on a yawed, infinite, circular cylinder in laminar

compressible flow . a brief description of the method of ref. 1 is

presented, followed by a discussion of the assumptions and

mathematical procedure which lead to a considerable simplification .

.I 540

.T

use of local similarity concepts in hypersonic viscous

interaction problems .

.A

forbes dewey,c.

.B

a.i.a.a. j. 1963, 20.

.W

use of local similarity concepts in hypersonic viscous

interaction problems .

the problem of predicting the characteristics

of a hypersonic laminar boundary layer that

interacts with the external flow field is approached

using the tangent wedge formulation for

the inviscid flow field and the method of similar

solutions for the viscous flow . it is shown that the concept of local similarity which allows

the pressure gradient parameter to vary in

the streamwise direction leads to an explicit

relation between the viscous and inviscid flows

for all values of the hypersonic interaction

parameter . the conditions of /strong/ and

limits of the general relations . the present theory

is compared with three independent experimental

investigations . in each case, the

agreement is found to be excellent over the range of

investigated . it is shown, using asymptotic

solutions to the exact boundary layer equations,

that the present theory is applicable to a wide

variety of viscous interaction problems .

a large number of solutions to the laminar

boundary layer similarity equations for a perfect

gas with cross flow and surface mass transfer are

given . these numerical results, when combined

with the solutions of previous authors, are

sufficient to describe the range of conditions with high precision .

.I 541

.T

similitude of hypersonic flows over slender bodies

in non-equilibrium dissociated gases .

.A

inger,g.r.

.B

a.i.a.a. j. 1963, 46.

.W

similitude of hypersonic flows over slender bodies in non-equilibrium dissociated gases .

this paper is concerned with the similitude laws governing inviscid, nonequilibrium gas flows around blunt or sharp-nosed slender bodies at zero angle of attack, based on the hypersonic small disturbance flow theory . some related features of the interaction between the effects of nose bluntness and nonequilibrium dissociation and vibration and the influence of a dissociated freestream are also discussed . the hypersonic equivalence principle and the related similitude for affinely related bodies are set forth for nonequilibrium flows in either diatomic gases or a gas mixture such as air . for a family of diatomic gases, as opposed to a given gas such as air, a generalized ambient gas state scaling condition is obtained, whereby the ambient density and temperature need not be simulated . a detailed discussion is given of blunted cylinders and slabs or sharp-nosed cones and wedges, including example nonequilibrium flow field correlations of numerical solutions available in the literature . low density nonequilibrium flows with a negligible shock layer atom recombination rate are also

examined ,. as expected, a less restrictive small

disturbance similitude law is obtained in this

case .

.I 542

.T

biot's variational principle in heat conduction .

.A

lardner,t.j.

.B

a.i.a.a. j. 1963, 196.

.W

biot's variational principle in heat conduction .

biot's variational principle is applied to a number of different

one-dimensional heat conduction problems . these problems

show the applicability of the variational principle to problems

involving prescribed heat flux boundary conditions and to those

with temperature-dependent material properties .

a method is introduced for including boundary conditions

when these are expressed as prescribed heat fluxes . the idea

behind this is overall energy balance within the body, which is a

constraint condition to be satisfied by the time histories of the

generalized coordinates .

the variational principle is then applied to the well-known

problem of constant surface heat flux in order to present the

technique and provide a basis for the remaining sections . the

equivalence of the result obtained in applying the variational principle for a prescribed surface temperature history to that obtained for a prescribed heat flux is also pointed out . radiation cooling due to fourth power radiation from semi-infinite solids and finite slabs together with radiation according to newton's law of cooling is then treated . finally, the introduction of temperature-dependent material properties is discussed and the determination of the temperature distribution in a semi-infinite solid with variable properties is investigated .

.I 543

.T

the stacking of compressor stage characteristics to give an overall compressor performance map .

.A

doyle,m.d.c.

.B

aero. aquart. 13, 1962.

.W

the stacking of compressor stage characteristics to give an overall compressor performance map .

a method of calculation is developed to compute the overall performance of a multi-stage axial compressor, from a knowledge of the individual stage characteristics, by a /stacking/ technique . compressor models are designed and their overall performance calculated . these results are compared to show, qualitatively, the effect of alterations in design and stage performance on overall performance and to find how compressors should be designed for optimum performance .

.I 544

.T

a theoretical and experimental study of oscillating wedge shaped

aerofoils in hypersonic flow .

.A

r. a. east

.B

.W

a theoretical and experimental study of oscillating wedge shaped

aerofoils in hypersonic flow .

aerodynamic stiffness and damping derivatives have been measured

in a /hypersonic gun/ wind tunnel for sharp and blunt-nosed two

dimensional single wedge shapes oscillating in the pitching mode in

hypersonic flow . the results, which have been compared with

theoretical prediction, modified to account for leading edge bluntness,

show that this may increase the damping by up to 50 percent for certain

axis positions . details of the experimental technique designed to

measure the derivatives in the short running times available are

described .

.I 545

.T

calculation of sideslip derivatives and pressure distribution

in asymmetric flight conditions on a slender wing-fin

configuration .

.A

sells,c.c.l.

.B

rae tn.aero.2805, 1962.

.W

calculation of sideslip derivatives and pressure distribution

in asymmetric flight conditions on a slender wing-fin

configuration .

the flow around slender wing-fin

configurations having curved leading

edges, whose shape is defined by polynomials,

is considered . a general

expression for the pressure distribution

on such a configuration in

asymmetric flow is derived and the

derivatives due to the particular case

of sideslipping motion are also given .

no numerical results are given for

wing-fin load distribution, but the

sideslip derivatives have been

evaluated in a number of cases for

gothic and ogee wings .

.I 546

.T

measurements of aerodynamic heating on a 15 cone of

graded wall thickness at a mach number of 6. 8.

.A

woodley,j.g.

.B

rae tn.aero.2847, 1962.

.W

measurements of aerodynamic heating on a 15 cone of

graded wall thickness at a mach number of 6. 8.

this note describes transient wall

temperature measurements made on a

in an airstream of mach number 6.8 .

the skin of the model was

sufficiently thin to allow it to reach zero

heat transfer conditions within a

running time of one minute .

in order to reduce effects

of longitudinal heat conduction during a

run the electroformed-nickel skin

of the model was made with graded

thickness, and as a result fairly

uniform temperature distributions along

the surface were obtained at all

times in both the laminar and turbulent

regions .

values of heat transfer, calculated from the wall temperature-time

histories using the thin-wall

temperature are compared to theoretical

estimates using the intermediate

enthalpy method 10, 11 .

.I 547

.T

boundary layer characteristics of caret wings .

.A

catherall,d.

.B

rae tn.aero.2835, 1962.

.W

boundary layer characteristics of caret wings .

the theory of laminar boundary layers along flat surfaces has been used in conjunction with eckert's approximations to the displacement thickness, skin friction and temperature profiles on the undersurface of a caret wing configuration . to a first approximation it has been assumed that parallel flow exits behind the shock outside the boundary layer, and the displacement of the shock by the boundary layer near the leading edge is neglected .

conduction of heat within the body and along the surface is neglected but radiation is included, so that are found . examples are given for various altitudes and configurations and the effect of the skin friction on the lift drag ratio calculated, assuming the undersurfaces to be plane .

.I 548

.T

the contraction of satellite orbits under the influence

of air drag . pt .iv with scale height dependent on

altitude .

.A

king-hele,d.g., and cook,g.e.

.B

rae tn.space 18, 1962.

.W

the contraction of satellite orbits under the influence

of air drag . pt .iv with scale height dependent on

altitude .

the effect of air drag on satellite

orbits of small eccentricity e

was studied in part i (tech. note

gw 533), on the assumption that

atmospheric density varies exponentially

with distance r from the earth's

centre, so that the 'density scale height'

h, defined as, is

constant . in practice h varies with height

in an approximately linear

manner, and in the present note the theory

is developed for an atmosphere

in which h varies linearly with r . equations

are derived which show how

perigee distance and orbital period vary

with eccentricity, and how
eccentricity varies with time . expressions
are also obtained for the
life-time and air density at perigee in terms
of the rate of change of orbital
period . the results are also presented
graphically .

the results are formulated in two
ways . the first is to specify the
extra terms to be added to the constant-h
equations of part i . the second
the best constant value of h for
use with the equations of part i . for
example, it is found that the
constant-h equations connecting perigee
distance (or orbital period) and
eccentricity can be used unchanged without
loss in accuracy, if h is taken
as the value of the variable h at a height
above the mean perigee height
during the time interval being considered,
where, and
decreases from to 0 as e decreases
from 0.02 to 0 . similarly the
constant-h equations for air density at
perigee can still be used if h is
evaluated at a height above perigee,

where, and

decreases to zero as e decreases from

constant-h equations can still be used

if h is evaluated at the scale height

below the initial height . variation of

h with altitude has a small effect

on the lifetime - about 3 - and on the

e-versus-time curve .

.I 549

.T

experimental study of the velocity and temperature

distribution in a high-velocity vortex-type flow .

.A

hartnett,j.p. and eckert,e.r.g.

.B

asme trans. 70, 1957.

.W

experimental study of the velocity and temperature

distribution in a high-velocity vortex-type flow .

  the vortex tube represents a simple device in which a

particular type of vortex motion may be studied in the

laboratory in an attempt to obtain a better understanding

of such flows . such an investigation has been pursued in

the heat transfer laboratory of the university of

minnesota . the present paper summarizes the major results of

this vortex-tube investigation .

.I 550

.T

laminar heat transfer in tubes under slip-flow conditions .

.A

sparrow,e.m. and lin,s.h.

.B

asme paper 61-wa-165.

.W

laminar heat transfer in tubes under slip-flow conditions .

the effects of low-density phenomena on the

fully developed heat-transfer characteristics

for laminar flow in tubes has been studied

analytically . consideration is given to the

slip-flow regime wherein the major rarefaction

effects are manifested as velocity and

temperature jumps at the tube wall . the

analysis is carried out for both uniform wall

temperature and uniform wall heat flux .

in both cases, the slip-flow nusselt numbers

are lower than those for continuum flow

and decrease with increasing mean free path .

extension of the results is made to include

the effects of shear work at the wall,

temperature jump modifications for a moving fluid,

and thermal creep .

.I 551

.T

analysis of a loaded cantilever plate by finite difference methods .

.A

livesley,r.k. and birchall,p.c.

.B

rae tn. ms26, 1956.

.W

analysis of a loaded cantilever plate by finite difference

methods .

the various difference patterns necessary for finite difference

solution of rectangular plate problems, with various boundary conditions

and under various transverse loads, are developed . the solution of

one particular problem on deuce is also described .

.I 552

.T

chemical kinetics of high temperature air .

.A

wray,k.l.

.B

hypersonic flow research, p 181, academic press, new york, 1962.

.W

chemical kinetics of high temperature air .

when a hypersonic object enters earth's atmosphere, a shock

wave is formed in front of it, and the air passing through this

shock wave is heated to high temperatures . the shock heated

molecules equilibrate their translational and rotational

degrees of freedom within a distance of a few mean free paths .

to achieve equilibrium, it is necessary to excite vibration, dissociate molecules, produce new molecules and produce ions and electrons . the problem is complex, since all these phenomena occur simultaneously and because the reaction rates depend on the temperature, density and composition which are changing during the relaxation toward equilibrium .

the experimental techniques used to investigate these reactions are briefly discussed along with the resulting rate expressions obtained by the various investigators . a compilation of the rate expressions for these reactions representing the author's evaluation of all the available data is presented . several pertinent problems which are not yet completely understood and which still require theoretical and experimental investigation are outlined . computed concentration, temperature and density time histories are shown for three different shock speeds in air . the time rate of change of concentration for each chemical reaction is also shown and regimes of importance for the various processes are discussed .

.I 553

.T

ablation of glassy materials around blunt bodies of revolution .

.A

hidalgo,h.

.B

ars j. 30, 1960.

.W

ablation of glassy materials around blunt bodies of revolution .

the steady-state equations of motion for a thin layer of an incompressible glassy material on the surface of an ablating and radiating blunt body are reduced to a first-order ordinary differential equation which is integrated numerically . this solution is coupled with the solution of the air boundary layer for both laminar and turbulent heat transfer with or without mass vaporization of the ablating material . the distribution of the effective energy of ablation around the body is thus obtained for a cone cylinder with a hemispherical cap that re-enters the atmosphere at hypersonic flight speeds, and has quartz as the ablating material . it is found that the ablation process from turbulent heating is more efficient than from the laminar case because of increased vaporization . this solution of the equations of motion at the stagnation point has been verified by are wind tunnel experiments . the present state of development of the are wind tunnel does not permit its use for experimental investigations of ablation around blunt bodies under turbulent heating .

.I 554

.T

generalized heat transfer formulas and graphs .

.A

detra,r.w. and kidalgo,h.

.B

ars j. 1961.

.W

generalized heat transfer formulas and graphs .

utilizing the research results of previously

reported investigations of the laminar, turbulent and

radiative heat transfer in dissociated air, some

generalized formulas for calculating heat transfer

are given . graphs for determining the laminar

heat transfer, momentum thickness reynolds

number, and turbulent heat transfer distributions

around an axisymmetric body are also given .

these heat transfer correlations are valid for velocities

between 6000 and 26,000 fps and for altitudes

up to 250,000 ft . this range of velocities and

altitudes covers the important re-entry regime of

practical re-entry trajectories having interest today .

in the last section of this report these

generalized results are specialized for icbm nose cone re-entry

applications . these formulas and graphs

may be found useful for making rapid engineering

estimates and preliminary design evaluations

of the heating problems associated with re-entry

into earth's atmosphere .

.I 555

.T

closing reply to comments on generalized heat transfer

formulas and graphs for nose cone re-entry into the

atmosphere .

.A

hidalgo,h.

.B

ars j. 1962.

.W

closing reply to comments on generalized heat transfer

formulas and graphs for nose cone re-entry into the

atmosphere .

in a recent paper (1), detra and hidalgo have shown

that, when the boundary layer is turbulent, the heat

flux per unit area at the sonic point of a nose cone may exceed

the corresponding laminar heat flux per unit area at the

stagnation point . the ratio of turbulent sonic-point to

laminar stagnation-point heat flux per unit area has been

estimated (2) to vary from about 1.0 to 10 for a hemispherical

nose as the reynolds number (based on nose diameter)

increases from 10 to 10 . since for an axisymmetric body

the surface area in the vicinity of the sonic point greatly

exceeds the area in the vicinity of the stagnation point, the

ratio of turbulent to laminar heat fluxes to the entire body

will be much greater than the above quoted ratios of heat

fluxes per unit area .

.I 556

.T

numerical comparison between exact and approximate theories of hypersonic inviscid flow past slender blunt nosed bodies .

.A

feldman,s.

.B

ars j. 30, 1960.

.W

numerical comparison between exact and approximate theories of hypersonic inviscid flow past slender blunt nosed bodies .

this paper presents numerical results of exact calculations of the inviscid equilibrium flow about a long hemisphere-cylinder in motion at hypersonic velocity . a comparison is made with blast wave as well as free layer theories of hypersonic flow . as a result of the comparison, it is concluded that the second-order blast wave theory can be used for the purpose of finding the shock shape and the body pressure distribution . however, this procedure is definitely empirical and cannot be justified on rational or theoretical grounds . we show that the presently calculated radial distribution of energy is radically different than that given by blast wave theory . if body shapes other than those considered here are of interest, the only reliable approach at the present time is to carry out

numerical calculations . it was found

that for certain flight velocities the pressure on the body

does not decay to free stream pressure

monotonically but overexpands .

.I 557

.T

a numerical comparison between exact and approximate

theories of hypersonic inviscid flow past slender blunt

nosed bodies .

.A

cheng,k.h. and chang,a.l.

.B

ars j. 31, 1961.

.W

a numerical comparison between exact and approximate

theories of hypersonic inviscid flow past slender blunt

nosed bodies .

this note refers to paper of same title by feldman in ars j. 30,

validity of blast wave theory cannot be justified on rational or

theoretical grounds because of different values of energy in cross

flow field as calculated by this theory and by method of

characteristics . present note questions this conclusion, shows reasonably

good agreement when energy is calculated for points where shock

location, streamline pattern, and velocity, temperature, and

pressure profiles are adequately defined, and still better agreement

when energy is calculated from flow quantities provided by-

characteristics method . results are checked using data from independent source . conclusion is reached that blast wave theory is

still valid .

.I 558

.T

experimental measurements of turbulent transition motion,

statistics and gross radial growth behind hypervelocity object.

.A

slattery,r.e. and clay,w.g.

.B

.W

experimental measurements of turbulent transition motion,

statistics and gross radial growth behind hypervelocity object.

the laminar-turbulent transition behind 0.500-in.-diameter

spheres at 8500 ft sec and behind

measured as a function of pressure . schlieren

motion-picture techniques were used to analyze the

turbulent motion and the results are described .

autocorrelation functions of the density fluctuations

of the turbulence have been measured . from

these values has been calculated and the results

are given for several positions in the turbulent

trail at 30 mm hg downstream air pressure . in addition

the authors' previous measurements of the

gross radial growth of the turbulent wake have been

extended to pressures of 10 mm hg for the case

of 0.500-in.-diameter spheres and to the trail behind

.I 559

.T

heat transfer at the forward stagnation point of blunt

bodies .

.A

reshotko,e. and cohen,c.b.

.B

naca tn.3513, 1955.

.W

heat transfer at the forward stagnation point of blunt

bodies .

relations are presented for the calculation of heat transfer at the forward stagnation point of both two-dimensional and axially symmetric blunt bodies . the relations for the heat transfer, which were obtained from exact solutions to the equations of the laminar boundary layer, are presented in terms of the local velocity gradient at the stagnation point . these exact solutions include effects of variation of fluid properties, prandtl number, and transpiration cooling . examples illustrating the calculation procedure are also included .

.I 560

.T

a theoretical study of the effect of upstream transpiration-cooling on the heat transfer and skin friction characteristics of a compressible laminar boundary layer .

.A

rubesin,m.w. and inouye,m.

.B

naca tn.3969, 1957.

.W

a theoretical study of the effect of upstream

transpiration-cooling on the heat transfer and skin friction characteristics

of a compressible laminar boundary layer .

an analysis is presented which predicts

the skin-friction and

heat-transfer characteristics of a compressible,

laminar boundary layer on a

solid flat plate preceded by a porous

section that is transpiration cooled .

the analysis is restricted to a prandtl

number of unity and linear

variation of viscosity with temperature .

the local skin friction has been

found to have a low value in the

region of transpiration cooling and then

to increase until it approaches

the value for a completely nonporous surface

asymptotically . the initial

increase in local skin friction is rapid

as half of the ultimate increase

occurs in a distance beyond the porous

region that is about 20 percent of

the length of the porous region for all rates

of injection . when the

total coolant flow rate is kept constant

and the porous length is varied,

it is found that the average skin friction

on a partially porous plate is

slightly lower than that on a fully porous plate .

the local heat transfer behaves in

a manner similar to that of the

local skin friction . it is found, in

an example, that the temperature

at the end of a partially porous plate

could be maintained at about the

same temperature as a fully porous plate

by doubling the total rate of

coolant flow .

.I 561

.T

a geometric problem related to the optimum distribution

of lift on a planar wing in supersonic flow .

.A

graham,e.w.

.B

j. ae. scs. 1958, 771.

.W

a geometric problem related to the optimum distribution

of lift on a planar wing in supersonic flow .

the problem studied may be regarded as a problem of

geometry . its simplest form (loosely stated) is then as follows .. a

mountain rises up from the x-y plane . determine the exact

shape of the mountain knowing only the cross-sectional area of every possible cut which can be made through the mountain with a vertical plane . in a more complicated version of the problem, the given information might be restricted to the cross-sectional area of every cut which can be made by a vertical plane inclined less than 45 to the y-axis .

this latter case has direct applications to certain minimum drag problems in supersonic flow . the shape of the mountain corresponds to the (unknown) shape of the optimum lift distribution on a planar wing . the cross-sectional area of a cut is the integrated value of the lift along a straight line crossing the wing plan form . for a restricted range of line inclinations, these optimum integrated lift values can sometimes be determined directly . here it is assumed that they are given .

the problem in its simplest form was originally solved by radon, who found solutions for a large class of such problems . the derivation presented here may perhaps be more readily understood .

.I 562

.T

concerning the effect of compressibility on laminar

boundary layers and their separation .

.A

howarth,l.

.B

proc. roy. soc. a, 194, 1948.

.W

concerning the effect of compressibility on laminar boundary layers and their separation .

the theory of compressible flow in a laminar boundary layer has been developed for the case when the viscosity is assumed to be proportional to the absolute temperature and the prandtl number is unity . (these assumptions may be compared with the empirical relations suggested by cope .)
it is shown that a transformation of the ordinate normal to the layer can lead to a simplified form of equation of motion very similar to the ordinary incompressible equation but modified by a multiplicative factor g in the pressure term . this factor is greater than unity at the boundary and tends to one at the outside of the layer .

several particular solutions are considered including accelerated flow with a linearly increasing velocity and retarded flow along a flat plate with a linearly decreasing velocity .

the general implications of the theory are discussed and qualitative conclusions are drawn when the mainstream velocity starts from a stagnation point, rises to a maximum and subsequently falls . it is concluded that for such a velocity distribution increasing compressibility will reduce the skin friction, increase the boundary layer thickness and cause

earlier separation as compared with the incompressible

flow with the same mainstream velocity

distribution and the kinematic viscosity corresponding

to conditions at the stagnation point .

.I 563

.T

the law of the wake in the turbulent boundary layer .

.A

coles,d.

.B

j. fluid mech. 1, 1956,191.

.W

the law of the wake in the turbulent boundary layer .

after an extensive survey of mean-velocity profile measurements

in various two-dimensional incompressible turbulent

boundary-layer flows, it is proposed to represent the profile by a linear

combination of two universal functions . one is the well-known

law of the wall . the other, called the law of the wake, is

characterized by the profile at a point of separation or reattachment .

these functions are considered to be established empirically, by

a study of the mean-velocity profile, without reference to any

hypothetical mechanism of turbulence . using the resulting

complete analytic representation for the mean-velocity field,

the shearing-stress field for several flows is computed from the

boundary-layer equations and compared with experimental data .

the development of a turbulent boundary layer is ultimately interpreted in terms of an equivalent wake profile, which supposedly represents the large-eddy structure and is a consequence of the constraint provided by inertia . this equivalent wake profile is modified by the presence of a wall, at which a further constraint is provided by viscosity . the wall constraint, although it penetrates the entire boundary layer, is manifested chiefly in the sublayer flow and in the logarithmic profile near the wall .

finally, it is suggested that yawed or three-dimensional flows may be usefully represented by the same two universal functions, considered as vector rather than scalar quantities . if the wall component is defined to be in the direction of the surface shearing stress, then the wake component, at least in the few cases studied, is found to be very nearly parallel to the gradient of the pressure .

.I 564

.T

local heat transfer and recovery temperature on a yawed cylinder at a mach number of 4. 15 and high reynolds numbers .

.A

beckwith,i.e. and gallagher,j.j.

.B

nasa memo 2-27-59l, 1959.

.W

local heat transfer and recovery temperature on a yawed cylinder at a mach number of 4. 15 and high reynolds numbers .

local heat transfer, equilibrium temperatures, and wall static pressures have been measured on a circular cylinder at yaw angles of 0, 10, 20, 40, and 60 . the reynolds number range of the tests was from 1x10 to 4x10 based on cylinder diameter .

increasing the yaw angle from 0 to 40 increased the stagnation-line heat-transfer coefficients by 100 to 180 percent . a further increase in yaw angle to heat-transfer coefficients .

at zero yaw angle the boundary layer over the entire front half of the cylinder was laminar but at yaw angles of 40 and 60 it was evidently completely turbulent, including the stagnation line, as determined by comparison of local heat-transfer coefficients with theoretical predictions . the level of heating rates and the nature of the chordwise distribution of heat transfer indicated that a flow mechanism different from the conventional transitional boundary layer may have existed at the intermediate yaw angles of 10 and 20 . at all yaw angles the peak heat-transfer coefficient occurred at the stagnation line and the chordwise distribution of heat-transfer coefficient decreased monotonically from this peak .

the average heat-transfer coefficients over the front half of the cylinder are in agreement with previous data for a comparable reynolds number range .

the theoretical heat-transfer distributions for
both laminar and turbulent boundary layers are
calculated directly from simple quadrature formulas
derived in the present report .

.I 565

.T

similar solutions for the compressible boundary layer
on a yawed cylinder with transpiration cooling .

.A

beckwith,i.e.

.B

naca tn.4345, 1958.

.W

similar solutions for the compressible boundary layer
on a yawed cylinder with transpiration cooling .
heat-transfer and skin-friction
parameters obtained from exact
numerical solutions to the laminar
compressible-boundary-layer equations
for the infinite cylinder in yaw are
presented . the chordwise flow in
the transformed plane is of the
falkner-skan type . solutions are given
for chordwise stagnation flow with
both a porous and a nonporous wall .
the effect of a linear
viscosity-temperature relation is compared with

the effect of the sutherland viscosity-temperature relation at the stagnation line of the cylinder for a prandtl number of 0.7 . the effects of pressure gradient, mach number, yaw angle, and wall temperature are investigated for a linear viscosity-temperature relation and a prandtl number of 1.0 with a nonporous wall .

the results indicate that compressibility effects become important at large mach numbers and yaw angles, with larger percentage effects on the skin friction than on the heat transfer . the use of the two different viscosity relations gives about the same results except when large changes in temperature occur across the boundary layer, as for a highly cooled wall . the present solutions predict that a larger amount of coolant would be required at a given large mach number and yaw angle than would be predicted from solutions of the corresponding incompressible-boundary-layer equations .

.I 566

.T

investigation of local heat transfer and pressure drag

characteristics of a yawed circular cylinder at supersonic

speeds .

.A

goodwin,g., creager,m.o. and winkler,e.l.

.B

naca rm.a55h31, 1956.

.W

investigation of local heat transfer and pressure drag

characteristics of a yawed circular cylinder at supersonic

speeds .

local heat-transfer coefficients,

temperature recovery factors, and

pressure distributions were measured

on a circular cylinder at a nominal

mach number of 3.9 over a range of

free-stream reynolds numbers from

from 0 to 44 .

it was found that yawing the

cylinder reduced the local heat-transfer

coefficients, the average heat-transfer

coefficients, and the pressure

drag coefficients over the front side

of the cylinder . for example, at

is reduced by 34 percent and the

pressure drag by 60 percent . the

amount of reduction may be predicted by

a theory presented herein . local

temperature recovery factors were also

reduced by yaw, but the amount of

reduction is small compared to the

reduction in heat-transfer coefficients .

a comparison of these data with

other data obtained under widely

different conditions of body and stream

temperature, mach number, and

reynolds number indicates that these

factors have little effect upon the

dropoff of heat transfer due to yaw .

.I 567

.T

aerodynamic characteristics of a circular cylinder

at mach number of 6. 86 and angles of attack up to

90 .

.A

penland,j.a.

.B

naca tn.3861, 1957.

.W

aerodynamic characteristics of a circular cylinder

at mach number of 6. 86 and angles of attack up to

90 .

pressure-distribution and force

tests of a circular cylinder have

been made in the langley 11-inch

hypersonic tunnel at a mach number of

based on diameter, and angles of

attack up to 90 . the results are

compared with the hypersonic

approximation of grimminger, williams, and

young and with a simple modification

of the newtonian flow theory . the

comparison of experimental results

shows that either theory gives adequate

general aerodynamic

characteristics but that the modified newtonian

theory gives a more accurate

prediction of the pressure distribution .

the calculated crossflow drag

coefficients plotted as a function

of crossflow mach number were found

to be in reasonable agreement with

similar results obtained from other

investigations at lower supersonic

mach numbers . comparison of the

results of this investigation with

data obtained at a lower mach number

indicates that the drag coefficient

of a cylinder normal to the flow is

relatively constant for mach numbers

above about 4 .

.I 568

.T

shock wave effects on the laminar skin friction of

an insulated flat plate at hypersonic speeds .

.A

li,t.y. and nagamatsu,h.t.

.B

j. ae. scs. 20, 1953, 345.

.W

shock wave effects on the laminar skin friction of

an insulated flat plate at hypersonic speeds .

an approximate theory on the phenomena of interaction

between the shock wave and the laminar boundary layer on an

insulated flat plate at hypersonic speeds has been formulated .

results on the rate of growth of the boundary-layer thickness and

the rate of decay of the shock-wave strength have been found

that hold for . a new set of formulas

for the average skin-friction coefficient, over an insulated

flat plate at hypersonic speeds has been obtained . calculations

on the basis of the new formulas yield the data shown in figs.

steady decrease in as increases, the present results indicate

that may increase with at hypersonic mach

numbers .

.I 569

.T

an experimental investigation of leading edge shock wave boundary layer interaction at mach 5.8 .

.A

kendall, j. m.

.B

j. aero. sc. v. 24, pp 47-56, 1957 .

.W

an experimental investigation of leading edge shock wave boundary layer

interaction at mach 5.8 .

the boundary layer on a slender body tends to be very thick at

hypersonic speeds . it interacts with the external flow by producing larger

flow deflections near the leading edge than those due to the body alone

flow around the boundary layer gives rise to an induced pressure with a

negative gradient which thins the boundary layer and increases the skin

friction with respect to the zero pressure gradient value .

experiments on a flat plate with a sharp leading edge have been

performed in the galcit 5-dash by 5-dash in. mach 5.8 hypersonic wind

tunnel . the induced pressure was measured by means of orifices in the

plate surface . profiles of mach number, velocity, mass flow, pressure,

and momentum deficiency were calculated from impact pressure surveys

normal to the plate surface made at various distances from the leading

edge .

the results are as follows . /1/ the induced pressures are 25 per cent

higher than the weak interaction theory . /2/ the boundary layer and the

external flow are distinctly separate for as low as 6,000 . /3/ the

shock wave location is in good agreement with that predicted by the

friedrichs theory for a body shape equivalent to the observed

boundary-layer displacement thickness . /4/ expansion waves reflected from the shock are weak . /5/ the average skin-friction coefficient tends toward and nearly matches the zero pressure gradient value downstream, but increases to approximately twice that value as the leading edge is approached .

.I 570

.T

on the boundary layer equations in hypersonic flow

and their approximate solutions .

.A

lees,l.

.B

j. ae. scs. 20, 1953, 143.

.W

on the boundary layer equations in hypersonic flow

and their approximate solutions .

analytical solutions of the prandtl boundary-layer equations are obtained for the problem of the /strong/ interaction between the leading-edge shock and the viscous layer over a flat plate at hypersonic velocities . as the mach number increases and the interaction region spreads downstream over the plate, the local skin-friction coefficient increases rapidly over its conventional value . the local heat-transfer coefficient at first remains practically unaffected but then also begins to increase with mach number .

.I 571

.T

heat transfer to flat plate in high temperature rarefied

ultra-high mach number flow .

.A

nagamatsu,h.t., weil,h.a. and sheet,r.e.

.B

ars j. 32, 1962, 533.

.W

heat transfer to flat plate in high temperature rarefied

ultra-high mach number flow .

an investigation was conducted in a hypersonic

shock tunnel to determine the local heat transfer

rates for a sharp leading edge flat plate . the free

stream mach number range was 7.95 to 25.1 with

stagnation temperatures of approximately 2550

and 6500 r . for these temperature and mach

number conditions, the strong interaction parameter,

varied from 2.35 to 826 . the

corresponding knudsen numbers, based on the

ratio of the free stream mean free path and the

leading edge thickness, varied from 0.38 to 85.5 .

for free stream mach numbers greater than 10,

knudsen numbers of approximately unity, and

perfect gas conditions, the calculated heat transfer

coefficients were found to vary as as predicted

by the noninsulated flat plate theory of li

and nagamatsu . for the case of,

the leading edge slip phenomenon

drastically reduced the local heat transfer coefficients

as compared to the theoretical values predicted with no slip at the surface . for the extreme case of and,

the measured local

heat transfer rate was an order of magnitude

less than the analytical value . both the knudsen

number and the free stream mach number are

important physical parameters that determine the

extent of the slip-flow region .

.I 572

.T

boundary layer displacement and leading edge bluntness effects in high

temperature hypersonic flow .

.A

cheng, h.k., hall,j.g., golian, t.c. and hertzberg, a.

.B

j. aero. sc. v. 28, pp 353-381, 410. 1961 .

.W

boundary layer displacement and leading edge bluntness effects in high

temperature hypersonic flow .

two important features of hypersonic flow over slender or thin bodies

are the displacement effect of the boundary layer and the large

down-stream influence of leading-edge bluntness . the present paper

contributes new theoretical and experimental results on this problem .

the interaction of the two effects is treated theoretically by extending

the basic shock-layer concept . in the outer inviscid flow, a model

consisting of a detached shock layer and an entropy layer is introduced

to account for bluntness . in the boundary layer, the approximate

solution is found to be governed by a local flat-plate similarity . under the assumption of a strong bow shock and a specific heat ratio close to unity, a theory is developed for an arbitrary thin body . for flat-plate afterbodies, the theory yields a solution agreeing with blast-wave theory at one limit and strong-interaction theory at the other . within the framework of the present theory, the problems involving angle of attack are also analyzed . complementary to the above study, a hypersonic similitude involving strong shocks, but not requiring close to one a natural comparison with experimental data correlated on the basis of this similitude .

flat-plate experiments in air, conducted in the c.a.l. 11 x 15-dashin. hypersonic shock tunnel under cold-wall conditions, included measurement of surface heat-transfer distributions and schlieren studies for zero and nonzero angle of attack . steady laminar heat-transfer rates were measured by means of thin-film resistance thermometers at air test-flow mach numbers around 12, free-stream reynolds numbers from 1.4 x 10 to 1. for most of the experiments, airflow stagnation temperatures ranged from ratios of about 0.15 . the range of test conditions at this stagnation temperature encompassed the limiting cases of dominant bluntness and dominant viscous-interaction effects . heat-transfer distributions were also measured on a sharp plate for air stagnation temperatures ranging from 2,000degreek up to 4,000degreek .

the experimental data are quite well correlated in terms of the foregoing theoretical similitude variables characterizing combined effects of boundary-layer displacement and bluntness . the correlations obtained suggest that for the present experimental conditions, at least, the hypersonic viscous similitude is valid even with leading-edge bluntness in the paper, is generally fair .

.I 573

.T

viscous hypersonic similitude .

.A

hayer,w.d. and probstein,r.f.

.B

j. ae. scs. 26, 1959, 815.

.W

viscous hypersonic similitude .

an extension of classical hypersonic similitude is developed

which takes into account the interaction effect of the

displacement thickness of the boundary layer . a basic result of this

viscous similitude is that the total drag including frictional drag

obeys the classical similarity law for the pressure drag .

additional similarity conditions governing viscous effects must be

imposed in this similitude .

underlying the similitude is a new hypersonic boundary-layer

independence principle . according to this principle, the

principal part of a hypersonic boundary layer with given pressure

and wall temperature distributions and free-stream total

enthalpy is independent of the (high) external mach number

distribution outside the boundary layer .

various features of viscous hypersonic similitudes are

discussed . it is found, for example, that it applies to three-

dimensional boundary-layer interaction effects on flat bodies, provided

the concepts of strip theory may be applied, and provided the

aspect ratio is an invariant .

.I 574

.T

inviscid flow with nonequilibrium molecular dissociation for pressure

distributions encountered in hypersonic flight .

.A

bloom, m.g. and steiger, m.h.

.B

j. aero. sc. v. 27, pp 821-835, 1960 .

.W

inviscid flow with nonequilibrium molecular dissociation for pressure

distributions encountered in hypersonic flight .

one-dimensional inviscid nonequilibrium flows of a two-component model

gas are studied for prescribed pressure variations and an average

reaction rate based on recent data for oxygen recombination . these flows

are interpreted in relation to the flow along streamlines around blunt

hypersonic bodies . assuming equilibrium conditions in the subsonic

region, it is estimated that the flow in the initial supersonic

expansion region, which is approximately of prandtl-meyer character, will be

chemically frozen with respect to the molecular dissociation of the

primary components under the hypersonic, high-altitude flight conditions

considered . the flight conditions consist of flight velocities between

furthermore, on bodies of small surface inclination beyond the nose, the

flow will continue to be effectively frozen for at least 20 ft

down-stream of the nose . these conclusions may lead to the simplification of

procedures for theoretical calculation and testing .

the problem of distinguishing a dimensionless length-reaction rate

parameter, which characterizes the extent of departures from equilibrium or from frozen behavior in the flow fields of interest here, is discussed

.I 575

.T

atomic recombination in a hypersonic wind tunnel nozzle .

.A

bray, k.n.c.

.B

j. fluid mech. v.6 part 1, 1-32. july, 1959 .

.W

atomic recombination in a hypersonic wind tunnel nozzle .

the flow of an ideal dissociating gas through a nearly conical nozzle is considered . the equations of one-dimensional motion are solved numerically assuming a simple rate equation together with a number of different values for the rate constant . these calculations suggest that deviations from chemical equilibrium will occur in the nozzle if the rate constant lies within a very wide range of values, and that, once such a deviation has begun, the gas will very rapidly 'freeze' . the dissociation fraction will then remain almost constant if the flow is expanded further, or even if it passes through a constant area section . an approximate method of solution, making use of this property of sudden 'freezing' of the flow, has been developed and applied to the problem of estimating the deviations from equilibrium under a wide range of conditions . if all the assumptions made in this paper are accepted, then lack of chemical equilibrium may be expected in the working sections of hypersonic wind tunnels and shock tubes . the shape of an optimum nozzle is derived in order to minimize this departure from equilibrium .

it is shown that, while the test section conditions are greatly affected

by 'freezing', the flow behind a normal shock wave is only changed

slightly . the heat transfer rate and drag of a blunt body are estimated

to be reduced by only about 25 per cent even if complete freezing

occurs . however, the shock wave shape is shown to be rather more

sensitive to departures from equilibrium .

.I 576

.T

viscous and inviscid stagnation flow in a dissociated hypervelocity free

stream .

.A

inger, g.r.

.B

proc. 1962. heat transfer and fluid mech. inst .

.W

viscous and inviscid stagnation flow in a dissociated hypervelocity free

stream .

high reynolds number hypersonic stagnation flow over a blunt-nosed body

in a nonequilibrium dissociated free stream is analyzed and compared to

a similar flow in an initially undissociated ambient gas . free stream

dissociation effects on various equilibrium stagnation flow properties

in air are presented as a function of the ambient atom mass fraction and

dissociation energy for velocities ranging from 15,000 to 25,000 fps .

significant changes in the bow shock geometry, stagnation gas state, and

boundary layer behavior are found when the free stream dissociation

involves more than 10(of the total energy . it is observed that for

large amounts of both atomic oxygen and nitrogen ahead of the body, the

equilibrium shock layer properties converge toward those pertaining to chemically and vibrationally-frozen flow across the bow shock .
moreover, under certain conditions, the ionization level can be increased by an order of magnitude and the usual reduction in frozen boundary layer heat transfer due to a highly-cooled noncatalytic surface can increase from stall of adjacent stages .
the effects of compromises of stage matching to favor part-speed operation were also considered . this phase of the study indicated that such compromises would severely reduce the complete-compressor-stall margin . furthermore, the low-speed stage stall problem is transferred from the inlet stages to the middle stages, which are more susceptible to abrupt-stall characteristics .
the analysis indicates that inlet stages having continuous performance characteristics at their stall points are desirable with respect to part-speed compressor performance . these characteristics must, however, be obtained when the stages are operating in the flow environment of the multistage compressor . alleviation of part-speed operational problems may also be obtained by improvement in either stage flow range or stage loading margin .
the results of this analysis are only qualitative . the trends obtained, however, are in agreement with those obtained from experimental studies of high-pressure-ratio multistage axial-flow compressors, and the results are valuable in developing an understanding of the off-design problem . in addition to these stage-matching studies, a general discussion of variable-geometry features such as air bleed and adjustable gas model . numerical solutions of non-equilibrium airflows with fully coupled chemistry provide a preliminary verification of such scaling for

benser, w.a.

.W

limit characteristics . the analysis indicated that all these problems

could be attributed to discontinuities in the performance

characteristics of the front stages . such discontinuities can be due to the type

of stage stall or to a deterioration of stage performance resulting

blades is included .

.I 577

.T

on hypersonic similitude .

.A

hayes,w.d.

.B

q. app. math. 5, 1947, 105.

.W

on hypersonic similitude .

tsien in a recent paper (j. math. phys. mass. inst. tech.

sonic flows around slender bodies and has pointed out that

the product of mach number and fineness ratio is a basic

similarity parameter . the author enlarges on this notion,

indicating that the problem of hypersonic flow about a

slender body in three dimensions is the same as that of a

certain two-dimensional nonsteady flow (with time replacing

the lengthwise spatial coordinate) characterized by

essentially the same similarity parameter .

.I 578

.T

dissociation scaling for nonequilibrium blunt nose flows .

.A

gibson, w.e.

.B

ars journal. vol. 32pp. 285-287. 1962 .

.W

dissociation scaling for nonequilibrium blunt nose flows .

stage-stacking study . the principal problems considered were poor

low-speed efficiency, multiple-valued performance characteristics at

intermediate speeds, and poor intermediate-speed compressor surge or

stall-naca rm e56b03b, 1956 . chapter xiii

.W

compressor operation with one or more blade rows stalled .

an analysis of the part-speed operating problems of high-pressure-ratio

air .

.I 579

.T

further developments of new methods in heat flow analysis .

.A

biot,m.a.

.B

j.ae.scs.26, 1959.

.W

further developments of new methods in heat flow analysis .

lagrangian methods in heat-flow problems and transport

phenomena were introduced by the writer in some previous work .

the present paper develops further one particular aspect of the method,--i.e., the elimination of /ignorable coordinates ./ this is accomplished by a special choice of generalized coordinates, each of which is constituted by an arbitrary temperature distribution and an /associated flow field ./ the latter is a vector field which is derived from the corresponding scalar field by a variational method . the procedure is valid for a certain class of nonlinear problems, provided we replace the temperature by the heat content as the unknown . it is shown that for normal coordinates derivation of the associated flow field is immediate . the use of normal coordinates and their associated flow fields is illustrated by an example . introduction of dirac functions and associated flow fields yields a procedure which constitutes a generalization of the classical formulation by green's functions and integral equations . this is illustrated by application to one-dimensional problems of heating of a homogeneous or composite slab and directly verified by classical methods in the appendix .

.I 580

.T

new thermo-mechanical reciprocity relations with application

to thermal stress analysis .

.A

biot,m.a.

.B

j. ae. scs. 26, 1959.

.W

new thermo-mechanical reciprocity relations with application

to thermal stress analysis .

based on the variational formulation of linear thermodynamics as developed previously by the writer, thermomechanical reciprocity relations are discussed which lead to new methods of analysis of thermal stresses . these reciprocity relations are quite different from the usual ones derived from the analogy of thermal loading with a combination of surface and body-force distribution . the results are applicable to stationary and transient temperatures in elastic and viscoelastic structures . the methods are entirely variational and do not require the evaluation of the temperature field . the stresses at one point are expressed directly in terms of any arbitrary distribution temperatures applied externally, including the effect of surface heat-transfer layer . the concepts and procedures are illustrated on a simple example . the relation is pointed out between the reciprocity property and the generalization of castigliano's principle to thermomechanics .

.I 581

.T

approximate formulas for thermal-stress analysis .

.A

d. j. johns

.B

lecturer, the college of aeronautics, cranfield, england

.W

approximate formulas for thermal-stress analysis .

the basis of any thermal-stress analysis is the determination of the temperature distributions in the structure . for arbitrary flight histories, the determination of such distributions is rather tedious

and not completely general . this latter fact handicaps optimization

studies in the project design stage when it is desirable to be able to

express the thermal-stress distributions in a general manner .

in this note, general expressions are derived for the thermal-stress

distributions in a typical i-section using similar assumptions to

those of biot .

.I 582

.T

the melting of finite slabs .

.A

goodman,t.r. and shea,j.j.

.B

j. app. mech. 27, 1960, 16.

.W

the melting of finite slabs .

an approximate method, known as the

heat-balance integral, is used to determine the

melting rate of a finite slab which is initially

at a uniform temperature below the melting

point . the slab is acted upon by a

constant heat input at one face and has its other

face either insulated or kept at its initial

temperature . the first three terms of series

solutions in an intrinsically small parameter

are obtained for the time histories of

melting and the temperature distribution in the slab .

.I 583

.T

influence coefficients for real gases .

.A

mario william cardullo

.B

u.s. naval air rocket test station, lake denmark, dover, n.j.

.W

influence coefficients for real gases .

in the analysis of one-dimensional fluid-flow problems, it is often

assumed that the behavior of the medium is that of a perfect gas .

this assumption is justified, provided the pressure and temperature

range of interest is small and near atmospheric . at higher pressures

and temperatures various deviations are introduced thereby causing

deviations from the results obtained by using the ideal fluid-flow

equations .

in this note, influence coefficients, similar to those developed by

shapiro, are presented for the case of real gases . this analysis is

based upon the use of various functions of the compressibility factor

emmons . some of the assumptions made were as follows.. (1) the flow

is one-dimensional and steady, (2) changes in the stream properties

are continuous, and (3) the flow is comprised of imperfect gases

.I 584

.T

conduction of heat in a solid with a power law of heat

transfer at its surface .

.A

jaeger,j.c.

.B

proc. cam.phil. s. 46, 1950, 634.

.W

conduction of heat in a solid with a power law of heat

transfer at its surface .

the nonlinear boundary value problem, where

and m are constants, is solved formally by first

introducing power series in t for the unknown temperature

and flux at the surface and then determining the coefficients

in those series . in this manner the temperature function is

determined as a series of repeated integrals of error

functions . the convergence is rapid only for small values of t .

the special cases and generalizations of

the condition at the surface for which the same method

applies, are noted . surface temperatures are also found by

methods of difference equations, where t is not limited to

small values . graphs of these temperatures corresponding

to various laws of heat transfer at the surface are shown .

.I 585

.T

nonlinear heat transfer problem .

.A

chambre,p.l.

.B

j. app. phys. 30, 1959, 1683.

.W

nonlinear heat transfer problem .

a study has been made of the time-dependent heat conduction in a semi-infinite medium subject to a boundary condition which can involve the temperature in a nonlinear manner . a formulation for the determination of the surface temperature, which is often of greatest physical interest, leads to a nonlinear volterra integral equation . a simple iterative solution method, with an accuracy suitable for many practical purposes is presented . as an example, the problem of the time-dependent surface temperature of a body receiving heat according to the stefan-boltzmann law is treated . the analysis is also applicable to physical adsorption or chemisorption processes which occur at the boundary .

.I 586

.T

an approximate treatment of unsteady heat conduction in semi-infinite solids with variable thermal properties .

.A

yang,k.t. and szewczyk,a.

.B

j. heat trans. asme trans. 81, 1959, 251.

.W

an approximate treatment of unsteady heat conduction in semi-infinite solids with variable thermal properties .

this very short paper presents an approximate procedure for the calculation of unsteady heat conduction in semi-infinite solids

with variable thermal properties . it is claimed to be an

improvement over previous efforts in this area since it yields physically

sensible results for cases where thermal properties have a large

dependence on temperature . instead of using polynomials to

represent an unsteady temperature profile an exponential form is used .

good agreement is shown for several cases where the method of

the paper is compared with exact solutions .

.I 587

.T

variational analysis of ablation .

.A

m. a. biot and h. daughaday

.B

consultant and principal research engineer, respectively, cornell

aeronautical laboratory, inc., buffalo 21, n.y.

.W

variational analysis of ablation .

the variational and lagrangian thermodynamics developed in earlier

publications are directly applicable to problems of heat conduction

with melting boundaries . these techniques are used here in treating

the problem of a half-space subjected to a constant rate of heat input

at the melting surface (fig. 1) .

the applicability of the lagrangian equations to this case follows

from the fact that the basic variational principle is valid whether the

boundaries are fixed or move as arbitrary functions of time . this

can be seen if we remember that the equations govern only the instantaneous configuration of the flow rates for a given geometry and temperature field .

.I 588

.T

compressor operation with one or more blade rows stalled .

.A

.B

naca rm e56b03b, 1956 . chapter xiii

.W

compressor operation with one or more blade rows stalled .

an analysis of the part-speed operating problems of high-pressure-ratio ratio multistage axial-flow compressors was made by means of a simplified stage-stacking study . the principal problems considered were poor low-speed efficiency, multiple-valued performance characteristics at intermediate speeds, and poor intermediate-speed compressor surge or stall-limit characteristics . the analysis indicated that all these problems could be attributed to discontinuities in the performance characteristics of the front stages . such discontinuities can be due to the type of stage stall or to a deterioration of stage performance resulting from stall of adjacent stages .

the effects of compromises of stage matching to favor part-speed operation were also considered . this phase of the study indicated that such compromises would severly reduce the complete-compressor-stall margin . furthermore, the low-speed stage stall problem is transferred from the inlet stages to the middle stages, which are more susceptible to abrupt-stall characteristics .

the analysis indicates that inlet stages having continuous performance

characteristics at their stall points are desirable with respect to
part-speed compressor performance . these characteristics must, however,
be obtained when the stages are operating in the flow environment of the
multistage compressor . alleviation of part-speed operation problems may
also be obtained by improvement in either stage flow range or stage-
loading margin .

the results of this analysis are only qualitative . the trends
obtained, however, are in agreement with those obtained from experimental
studies of high-pressure-ratio multistage axial-flow compressors, and
the results are valuable in developing an understanding of the
off-design problem . in addition to these stage-matching studies, a general
discussion of variable-geometry features such as air bleed and
adjustable blades is included .

.I 589

.T

some stall and surge phenomena in axial flow compressors .

.A

.B

.W

some stall and surge phenomena in axial flow compressors .
observations of rotating stall have shown that a wide variety
of stall patterns is possible .

hot-wire anemometer data on a multistage compressor have
shown a progressive-type stall at low speeds . the amplitude
of the flow fluctuations increases in magnitude through the first
few stages and then diminishes rapidly to a small value in the
latter stages . a stage-stacking analysis has shown that rotating

stall will exist over a large portion of the compressor map at low speeds but will be instigated almost simultaneously with compressor surge at high speeds .

blades failures attributable to resonant vibrations excited by rotating stall have been experienced in single and multistage compressors .

in the stage-stacking analysis no deterioration of stage performance due to unsteady flow resulting from stall of adjacent stages was considered . in general, the pressure drop at the stall point is believed to be much larger than indicated by an analytical formulation of compressor performance . compressor surge is attributed to a limit cycle operation about the compressor stall point, and, as indicated in a few compressor tests and in jet-engine tests, a small compressor discharge receiver volume may result simply in stall of the compressor without the cyclic characteristics of compressor surge . in this event, engine operation will be limited because of the large drop in performance which accompanies compressor stall .

.I 590

.T

effects of stage characteristics and matching on axial flow compressor performance .

.A

stone, a.

.B

trans. asme, v. 80, p. 1273, 1958 .

.W

effects of stage characteristics and matching on axial flow compressor

performance .

the use of stage characteristics obtained from test data in the

performance analysis and development of an axial-flow compressor is described

relative stage matching as shown by an idealized example and also by

test experience . factors governing major performance parameters are

discussed and certain development problems and possible solutions are

reviewed .

.I 591

.T

an approximate equation for the /choke line/ of a compressor .

.A

csanady, g.t.

.B

j. aero. sc. p. 637, august 1960 .

.W

an approximate equation for the /choke line/ of a compressor .

discussion of a similarity between the pressure-ratio versus

inlet-mass-flow-coefficient characteristic of a stream or gas turbine and the

analogous characteristic of an expansion /laval/ nozzle . this idea is

extended to a compressor and a compression nozzle, and an approximate

expression for the /choke line/ of the compressor is developed .

.I 592

.T

design of axial compressors .

.A

howell, a.r.

.B

proc. i mech. e. v. 153, 1945 .

.W

design of axial compressors .

the main types of axial compressors are described, and the use of

generalized design curves to make performance estimates is advocated . the

different variables are weight, power, pressure ratio, temperature rise,

mass flow, rotational speed, stage efficiency, blade bending stresses

due to aerodynamic loading, and methods and materials of construction .

air outlets, flow coefficients and different blade forms are also

considered .

.I 593

.T

theoretical considerations of flutter at high mach

number .

.A

morgan,h.g., runyam,h.l. and huckell,v.

.B

j. ae. scs. 25, 1958.

.W

theoretical considerations of flutter at high mach

number .

some of the theories for two-dimensional oscillatory air forces

which may be applied in flutter calculations at high mach

numbers are discussed . these include linear theory, van dyke's

second-order theory, piston theory, landahl's method,

tangent-wedge and tangent-cone approximations, newtonian theory, and a new nonlinear-pressure method . a comparison of the theories

is made by showing the results of flutter calculations for mach

numbers up to 10, and the possibility of flutter at these higher

mach numbers is pointed out .

results of flutter calculations are shown to illustrate the

various effects arising from a nonlinear thickness theory . the

possibility of large flutter speed thickness effects which depend on

frequency ratio is shown . the influence of airfoil shape is discussed

and flutter speed trends with center of gravity and elastic axis

locations are presented . some possible refinements of piston

theory are discussed for use at very high mach numbers . these

include the use of local flow conditions and the use of newtonian

theory over the leading edge of a blunt-nosed airfoil .

.I 594

.T

wind tunnel techniques for the measurements of oscillatory

derivatives .

.A

bratt, j.b.

.B

arc 22, 146, 1960

.W

wind tunnel techniques for the measurements of oscillatory

derivatives .

this paper discusses the basic principles employed in techniques for the

measurement of oscillatory derivatives in wind tunnels, and gives some

account of the associated instrumentation . the suitability of the various techniques for different test conditions is also discussed and brief reference is made to wind tunnel effects on the measurements .

.I 595

.T

the equilibrium piston technique for gun tunnel operation .

.A

east,r.a. and pennelegion,l.

.B

arc 22, 852, 1961.

.W

the equilibrium piston technique for gun tunnel operation .

a modified technique for the operation of a gun tunnel is suggested based on experimental results . if the piston mass and the initial barrel pressure are chosen correctly, then the peak pressures associated with the gun tunnel may be eliminated . under these conditions the piston is brought to rest with no overswing . some measurements of the piston motion using a microwave technique are reported which confirm this idea .

the wave diagram associated with this mode of operation is shown, and some calculations of the stagnation pressure are given which show that during the suggested running time, the stagnation pressure may be considerably greater than the driving pressure if the driving chamber cross-sectional area is large compared with that of the driven section . for a uniform shock tube the stagnation pressure will always be less than the driving pressure . the use of air, helium and hydrogen as driving gases has been considered .

experiments in a gun tunnel are reported which show that the equilibrium piston technique enables steady stagnation pressures to be

achieved over a time of approximately 15 ms using air as the driving

gas . the expansion caused by the piston acceleration is shown to

interact with the stationary piston, but this is found to produce only a

small drop in stagnation pressure .

.I 596

.T

the properties of crossed flexure pivots, and the influence of the

point at which the strips cross .

.A

wittrick, w.h.

.B

aero. quart. v. 2, 1950-1 .

.W

the properties of crossed flexure pivots, and the influence of the

point at which the strips cross .

it is shown that the rotational stiffness of a crossed flexure pivot

varies considerably when subjected to an applied force . the type of

variation can be radically changed simply by moving the point at which

the strips cross . the relation between torque and rotation for a given

applied force is not exactly linear and the extent of the non-linearity

is determined by taking into account the small movements of the centre

of rotation of the pivot . finally, for design purposes, an analysis of

the maximum stresses in the strips is given .

.I 597

.T

measurements of pitching moment derivatives for blunt-nose aerofoils oscillating in two-dimensional supersonic flow .

.A

pugh, p.g. and woodgate, l.

.B

arc. 23, 012, 1961 .

.W

measurements of pitching moment derivatives for blunt-nose aerofoils

oscillating in two-dimensional supersonic flow .

direct pitching moment derivatives have been measured using the method

of scruton, woodgate et al for two single wedge blunt-nosed aerofoils .

these measurements were made at mach numbers of 1.75 and 2.47 and

frequency parameters less than 0.02 . in general, nose blunting was found

to have little effect on the derivatives although changes were observed

for the thinner wedge at a mach number of 1.75 .

.I 598

.T

new test techniques for a hypervelocity wind tunnel .

.A

stalmach, c.j. and cooksey, j.m.

.B

aerospace engineering. vol. 21. march 1962 .

.W

new test techniques for a hypervelocity wind tunnel .

the measurement of rocket exhaust effects on vehicle stability and the

measurement of aerodynamic damping were made in an arc-discharge type of

hypervelocity wind tunnel . sample data are given to indicate the

quality of data obtainable in this tunnel, and samples of self-luminous and shadowgraph photographs are also presented .

.I 599

.T

aerodynamic forces, moments and stability derivatives

for slender bodies of general cross section .

.A

sacks,a.h.

.B

naca tn.3283, 1954.

.W

aerodynamic forces, moments and stability derivatives

for slender bodies of general cross section .

the problem of determining the total

forces, moments, and stability

derivatives for a slender body performing

slow maneuvers in a compressible

fluid is treated within the assumptions

of slender-body theory . general

expressions for the total forces (except

drag) and moments are developed

in terms of the geometry and motions of

the airplane, and formulas for

the stability derivatives are derived in

terms of the mapping functions

of the cross sections .

all components of the motion are

treated simultaneously and second

derivatives as well as first are obtained,
with respect to both the
motion components and their time rates of
change . coupling of the
longitudinal and lateral motions is thus
automatically included . a number of
general relationships among the various
stability derivatives are found
which are independent of the configuration,
so that, at most, only 35
of a total of 325 first and second
derivatives need be calculated
directly . calculations of stability
derivatives are carried out for two
triangular wings with camber and thickness,
one with a blunt trailing
edge, and for two wing-body combinations, one having a plane wing and
vertical fin .

the influence on the stability
derivatives of the squared terms in
the pressure relation is demonstrated,
and the apparent mass concept as
applied to slender-body theory is discussed
at some length in the light
of the present analysis . it is shown that
the stability derivatives can
be calculated by apparent mass although the

general expressions for the

total forces and moments involve additional terms .

.I 600

.T

the calculation of lateral stability derivatives of

slender wings at incidence including fin effectiveness,

and correlation with experiment .

.A

ross,a.j.

.B

rae r.aero.2647, 1961.

.W

the calculation of lateral stability derivatives of

slender wings at incidence including fin effectiveness,

and correlation with experiment .

comparisons are made between

low-speed experimental results and

estimates based on attached-flow

theory for the lateral stability

derivatives of slender wings at

incidence, and it is found that the flow

separation has little effect on

the sideslip derivatives . the reduction

in due to part-span anhedral

is evaluated, and a semi-empirical formula

is derived to account for important

second-order terms . for the rotary

derivatives, an attempt is made
to estimate the effect of the leading edge
vortices, but no satisfactory
conclusions have been reached .
the fin contributions to
the derivatives are evaluated on the basis
of treating the wing surface as
a total reflection plate . good agreement
with experiment is reached for
the sideslip derivatives, and for the
damping-in-yaw at moderate incidences .
sidewash is found to have a large
effect on the rolling derivatives,
and further information on the strength
and position of the leading edge
vortices in non-symmetric flow is required
before a complete calculation of
the sidewash can be given .

.I 601

.T

calculation of the flow past slender delta wings with
leading edge separation .

.A

mangler,k.w. and smith,j.

.B

rae r. aero.2593, 1957.

.W

calculation of the flow past slender delta wings with

leading edge separation .

the flow past a slender delta wing with a sharp leading edge, at incidence, usually separates along this edge, i.e. a vortex layer extends from the edge and rolls up to form a /core/ (a region of high vorticity) . a potential flow model of this is constructed in which the layer is replaced by a vortex sheet which is rolled up into a spiral in the region of the /core/ . this problem is reduced to a two-dimensional one by assuming a conical field and using slender wing theory . the shape and strength of the sheet are determined by the two conditions that it is a stream surface and sustains no pressure difference . use is made of results previously obtained for the core region and the remaining finite part of the sheet is dealt with by choosing certain functions for its shape and strength . the parameters in these functions are found by satisfying the two conditions stated above at isolated points . results are obtained for

the pressure distribution, chord

loading and norman force coefficient as

functions of the ratio of the

incidence to the apex angle . the lift for a

given incidence is about 15

below that found by brown and michael . flow

patterns are indicated in two

typical cases . the effect of separation on

the drag due to lift of a wing

with small thickness is discussed .

.I 602

.T

the 7 x 7 in . hypersonic wind tunnel at rae farnborough,

part 1, design, instrumentation and flow visualization

techniques .

.A

crabtree,l.f. and crane,j.

.B

rae tn.aero.2716.

.W

the 7 x 7 in . hypersonic wind tunnel at rae farnborough,

part 1, design, instrumentation and flow visualization

techniques .

this is the first of three parts of

the calibration report on the r.a.e.

some details of the design and lay-out

of the plant are given, together

with the calculated performance figures,

and the major components of the

facility are briefly described .

the instrumentation provided for

the wind-tunnel is described in some

detail, including the optical and other

methods of flow visualization used

in the tunnel .

later parts will describe the

calibration of the flow in the

working-section, including temperature measurements .

a discussion of the heater

performance will also be included as

well as the results of tests to determine

starting and running pressure ratios,

blockage effects, model starting loads,

and humidity of the air flow .

.I 603

.T

the 7 in. x 7 in. hypersonic wind tunnel at r.a.e. farnborough

part ii. heater performance .

.A

j. f. w. crane

.B

.W

the 7 in. x 7 in. hypersonic wind tunnel at r.a.e. farnborough

part ii. heater performance .

tests on the storage heater, which is cylindrical in form and mounted horizontally, show that its performance is adequate for operation at m=6.8 and probably adequate for flows at m=8.2 with the existing nozzles . in its present state, the maximum design temperature of 680 degrees centigrade for operation at m=9 cannot be realised in the tunnel because of heat loss to the outlet attachments of the heater and quick-acting valve which form, in effect, a large heat sink . because of this heat loss there is rather poor response of stagnation temperature in the working section at the start of a run . it is hoped to cure this by preheating the heater outlet cone and the quick-acting valve .

at pressures greater than about 100 p.s.i.g. free convection through the fibrous thermal insulation surrounding the heated core causes the top of the heater shell to become somewhat hotter than the bottom, which results in /hogging/ distortion of the shell . this free convection cools the heater core and a vertical temperature gradient is set up across it after only a few minutes at high pressure .

modifications to be incorporated in the heater to improve its performance are described .

.I 604

.T

the 7 in. x 7 in. hypersonic wind tunnel at r.a.e., farnborough

part iii - calibration of the flow in the working section .

.A

j. f. w. crane

.B

.W

the 7 in. x 7 in. hypersonic wind tunnel at r.a.e., farnborough

part iii - calibration of the flow in the working section .

the fused silica nozzle to give m=7 in the 7 in. x 7 in. hypersonic

wind tunnel produces a flow field with an average mach number of 6.85

along the centreline of the working section . the mach number gradually

decreases towards the boundary layer, and over a core of approximately

mach number .

the nozzle heats up during a run but this has little effect on the

mach number distribution . at one station the mach number was one-third

per cent greater for a run of 1 minute than for a run of 10 seconds .

the temperature field in the inviscid flow has an average variation of

in temperature with time throughout a run .

.I 605

.T

pressure measurements on a cone-cylinder-flare configuration at small

incidences for m 6.8 .

.A

woodley, j.g.

.B

r.a.e. tn. aero. 2739 .

.W

pressure measurements on a cone-cylinder-flare configuration at small

incidences for m 6.8 .

pressure measurements were made on a slender cone-cylinder-flare

configuration, slightly blunted at the nose, for 0, 3 and 6 degrees

incidence at a free-stream mach number of 6.8 .

it was found that the surface pressures obtained on the cone agreed with extrapolations to m equals 6.8 of theoretical values given in m.i.t.

tables /kopal/for yawed cones, and that impact theory gave a good

indication of the pressure level to be expected on all parts of the body

where surface incidence was sufficiently large to merit its use .

the semi-angles of the conical and flared parts of the model were both

the pressure level on the flare rose in all cases to approximately that

developed upstream on the cone surface .

no evidence of a marked over-expansion to pressures below the

free-stream value was noticed at the junction between cone and cylinder .

.I 606

.T

formulae and approximations for aerodynamic heating rates in high speed

flight .

.A

monaghan, r.j.

.B

rae. tn. aero. 2407 .

.W

formulae and approximations for aerodynamic heating rates in high speed

flight .

this note gives formulae and approximations suitable for making

preliminary estimates of aerodynamic heating rates in high speed flight .

the formulae are based on the /intermediate enthalpy/ approximation

which has given good agreement with theoretical and experimental

evidence . in the general flight case they could be used in conjunction with

an analogue computer or a step-by-step method of integration to

predict the variations of heat flow and skin temperature with time .

in the restricted case of flight at constant altitude and mach number,

simple analytical methods and results are given which include the

effects of radiation and can be applied to /thick/ as well as /thin/ skins

where h is the aerodynamic heat transfer factor, and g, d and k are the

heat capacity, thickness and thermal conductivity of the skin . if 0.1

the skin is approximately /thin/, i.e. temperature gradients across its

thickness may be neglected .

.I 607

.T

duct flow in magnetohydrodynamics .

.A

chang,c.c. and lundgen,t.s.

.B

z.angew. math.phys., 12 100-114, 1961.

.W

duct flow in magnetohydrodynamics .

this paper is an extension of the work of hartmann (2) and shercliff

transverse magnetic fields -- the simplest class of magnetohydrodynamic

problems . we are concerned here mainly with the boundary value

problems associated with flow in ducts with conducting walls .

.I 608

.T

aerodynamic noise in supersonic wind tunnels .

.A

laufer, j.

.B

j. aero. sc. v. 28, sept. 1961 .

.W

aerodynamic noise in supersonic wind tunnels .

hot-wire measurements in the free stream of a supersonic wind tunnel

were made in the mach number range of 1.6 to 5.0 . it is shown that the

mass-flow fluctuations increase very rapidly with increasing mach

number . if the fluctuation field is assumed to consist of sound waves-dash

an assumption that is consistent with the measurements-dashthe sound

intensity is approximately proportional to m, within the range of the

experiments . furthermore, the orientation of the field is found to be

different from the mach line direction,. it corresponds to a

sound-source velocity of approximately one-half the free-stream velocity for

the higher mach numbers . it is shown that the turbulent boundary layer

along the nozzle and the tunnel walls is responsible for this sound

field .

.I 609

.T

on three dimensional bodies of delta planform which

can support plane attached shock waves .

.A

peckman,d.h.

.B

rae tn. aero.2812.

.W

on three dimensional bodies of delta planform which

can support plane attached shock waves .

this note collects together in one report available theoretical work

on bodies which can support attached plane shock waves, discusses some of the possible merits of such shapes, and includes some calculations illustrating their properties . also, some preliminary results from wind tunnel tests are given, together with details of proposed future tests .

.I 610

.T

corner interference effects .

.A

gersten,k.

.B

agard r.299, 1959.

.W

corner interference effects .

the three-dimensional incompressible flow of fluid along the corner of two semi-infinite plates intersecting at right angles, especially the interference of the boundary layers of the two plates, is discussed . mainly, the more important case of turbulent boundary layer is treated by means of experimental studies carried out at the technical university of braunschweig . some theoretical results for laminar flow are also taken into account .

in order to describe the interference effects in the boundary layer, an interference displacement thickness and an interference skin friction have been introduced . it is shown from experiments and also from theoretical considerations how these two quantities depend on reynolds number . furthermore, the influence of interference on the transition from laminar to turbulent flow is investigated . in addition, some preliminary results are given about the effect of the pressure gradient on the interference effects .

.I 611

.T

an approximate solution of the compressible laminar

boundary layer on a flat plate .

.A

monaghan,r.j.

.B

rae tn. aero.2025.

.W

an approximate solution of the compressible laminar

boundary layer on a flat plate .

following a major assumption that enthalpy and velocity are dependent

only on local conditions, an enthalpy-velocity relation

is obtained for the laminar boundary layer on a flat plate where

subscripts p refer to the plate, 1 to the free stream and e to the

equilibrium temperature condition at the plate . when compared with

general results, this relation (exact for prandtl number o = 1)

gives a close approximation to crocco's numerical results for o = 0.725

and 1.25, up to .

using the above relation in conjunction with the approximate

viscosity-temperature relation suggested by chapman and rubesin, and

with young's suggested first approximation for shearing stress it is

shown that close approximations to displacement thickness and velocity

distribution are given by and where and which serves to define c .

these have the advantage of being algebraic in form whereas previous

results have involved complex numerical integrations for individual cases .

.I 612

.T

pressure distributions and flow patterns at m=4 . on

some delta wings of inverted 'v' cross section .

.A

squire,l.c.

.B

rae tn.aero.2838, 1962.

.W

pressure distributions and flow patterns at m=4 . on

some delta wings of inverted 'v' cross section .

wind tunnel tests have been made

to measure pressure distributions

and to study flow patterns on a series

of delta wings of inverted 'v'

cross-section . each of these wings

was designed to have a plane shock

wave in the plane of the leading edges

at a chosen mach number and incidence .

it was found that for a wide

incidence range about the design point

the shock wave remained virtually

attached to the leading edges and at

each incidence the pressure was

approximately constant over the lower

surface .

.I 613

.T

the contraction of satellite orbits under the influence of air drag

part i . with spherically symmetrical atmosphere .

.A

king-hele, d.g., cook, g.e. and walker, d.m.c.

.B

r.a.e. tn. gw. 533. 1959 .

.W

the contraction of satellite orbits under the influence of air drag

part i . with spherically symmetrical atmosphere .

the effect of air drag on satellite orbits of small eccentricity e/0.2/

is studied analytically by a perturbation method, on the assumption that

the atmosphere is spherically symmetrical . equations are derived which

show/1/how orbital period and perigee distance vary with eccentricity

as the orbit contracts, and/2/how each of these quantities varies with

time . the equations of type/1/are nearly independent of the oblateness

of the atmosphere . in all the equations, terms of order e and higher

are usually neglected . the results are also presented graphically, in a

manner designed for practical use .

the theory is to be extended to an oblate atmosphere in part ii, and

will later be compared with observation .

.I 614

.T

the contraction of satellite orbits under the influence of air drag .

.A

part ii . with oblate atmosphere .

.B

king-hele, d.g., cook, g.e. and walker, d.m.c.

r.a.e. tn. gw. 565. 1960 .

.W

the contraction of satellite orbits under the influence of air drag .

part ii . with oblate atmosphere .

the effect of air drag on satellite orbits of small eccentricity e/0.2/

was studied in part i/technical note no.g.w.533/on the assumption that

the atmosphere was spherically symmetrical . here the theory is extended

to an atmosphere in which the surfaces of constant density are

spheroids of arbitrary small ellipticity . equations are derived which

show how perigee distance and orbital period vary with eccentricity, and

how eccentricity is related to time . expressions are also obtained

which give lifetime and air density at perigee in terms of the rate of

change of period . in most of the equations, terms of order e and higher

are neglected . the results take different forms according as the

eccentricity is greater or less than about 0.025, while circular orbits

are dealt with in a separate section . the results are also presented

graphically in a manner designed for practical application, and examples

of the theory in use are given .

the influence of atmospheric oblateness is difficult to summarize fairly

simultaneously assume their'worst'values, some of the

spherical-atmosphere results can be altered by up to 30(as a result of oblateness

and 5-10(would be a more representative figure .

.I 615

.T

the contraction of satellite orbits under the influence of air drag .

part iii . high eccentricity orbits . /0.2 e 1/ .

.A

king-hele, d.g.

.B

r.a.e. tn. space 1, 1962 .

.W

the contraction of satellite orbits under the influence of air drag .

part iii . high eccentricity orbits . /0.2 e 1/ .

the effect of air drag on satellite orbits of eccentricity e less than

between 0.2 and 1 is presented . equations are derived which show how

perigee distance and orbital period vary with eccentricity during the

satellite's life, and how eccentricity is related to time,.and formulae

are obtained for the lifetime and the air density at perigee, in terms

of the rate of change of period . the results are also presented

graphically and their implications and limitations are discussed .

.I 616

.T

determination of upper-atmosphere air density and scale height from

satellite observations .

.A

groves, g.v.

.B

proc. roy. soc. a. 252, 16-27, 1959 .

.W

determination of upper-atmosphere air density and scale height from

satellite observations .

a solution is obtained for the rate of change of semi-major axis and

perigee distance of a satellite orbit with time due to the resistance of
the atmosphere . the logarithm of air density is assumed to vary
quadratically with height, and the oblateness of the atmosphere is taken
into account .
the calculation of perigee air density in terms of the rate of change
of satellite period is dealt with,. and the method is applied to data
at present available on six different satellites . the variation of air
density with height is obtained as
in p-28.59/0.15/-h-200//46/5/0.028/0.013//h-200///46/
for h in the range of approximately 170 to 700 km, where p is in grams/c
m, h is in kilometres and standard deviations are given in brackets .
.I 617
.T
determination of upper-atmosphere air density profile from satellite
observations .
.A
groves, g.v.
.B
proc. roy. soc. a. 252, 28-34, 1959 .
.W
determination of upper-atmosphere air density profile from satellite
observations .
the theory previously developed for the changes in the perigee distance
and semi-major axis of a satellite orbit due to air drag is extended to
enable the air-density profile/i.e.its relative variation with height/to
be derived from the motion of the orbit's perigee . the solution is
first obtained in terms of the change in perigee distance and then in

terms of the change in the radius of the earth at the sub-perigee point

the scale height in the 180 and 220 km altitude regions .

.I 618

.T

orbit decay and prediction of the motion of artificial satellites .

.A

michielsen, h.f.

.B

advances in astronautical science, vol 4 plenum press 1959 . pp 255-310

.W

orbit decay and prediction of the motion of artificial satellites .

the rate of decay of elliptic satellite orbits, due to atmospheric drag,

is investigated through variation of parameters and through use of an

atmospheric model involving a power function between density and

altitude . this model is shown to fit actual conditions better than an

exponential function .

the effects of the equatorial belt and the rotation of the earth are

investigated . the conclusion is reached that through these anomalies

atmospheric drag substantially affects the orbit elements, especially

those defining the orbit plane .

an alternate approach of variation of parameters is presented, by which

a direct relation between period decay and instantaneous density

conditions is established . this approach, by itself specifically adequate

for prediction work, also opens an avenue for systematic and unified

evaluation of observed decay .

.I 619

.T

density of the upper atmosphere from analysis of satellite orbits ..

further results .

.A

king-hele, d.g.

.B

nature, v184, 1267-1270, 1959 . r.a.e. rep. gw. 25. appendix y .

.W

density of the upper atmosphere from analysis of satellite orbits ..

further results .

the method previously described has been refined by taking into account

atmospheric rotation . further results are given from satellites of

latitude and season and day-to-night changes are reported .

.I 620

.T

earth satellite observations and the upper atmosphere .

.A

priester, w., martin, h.a. and kramp, k.

.B

nature, 188, pp 200-204. 1960 .

.W

earth satellite observations and the upper atmosphere .

atmospheric densities have been derived from artificial satellites in

altitudes 200-700 km. and from rockets up to about 200 km. to

consolidate the two sets of data, h.k. kallmann suggested a model with a

exact form of this curve has now been derived . corrections for the

is excellent .

very close correlation between atmospheric density variations/h180 km./

and the solar 20-cm. radiation implies that the origin of the'solar effect'may lie in the absorption of solar ultra-violet radiation . the atmospheric density curve between 180 and 200 km. shows a temperature inversion in the fl-layer . it is not yet possible to decide whether solar ultra-violet radiation as well as the solar he line and solar x-ray radiation contribute to the heating of the fl-layer . diurnal and seasonal density variations at altitudes 210, 562 and 660 km. have been derived from variations in acceleration of three satellites/sputnik 3, vanguard 1 and 2/ . group averages of diurnal variations are taken from different dates within the period may 15, 1958-october 1, 1959 . physcal conditions in the upper atmosphere are briefly summarized ..the'solar effect'originates in the fl-layer as a result of heating by the solar he line at 304 a. diurnal density variation at 210 km. is only a few per cent . absorption of solar electromagnetic radiation in the f2-layer, and large heat conductivity cause intense diurnal density and temperature variations above

.I 621

.T

latitude and diurnal variations of air densities from 190 to 280 km. as derived from the orbits of discoverer satellites .

.A

groves, g.v.

.B

proc. roy. soc. a. 263. 212-216. 1961 .

.W

latitude and diurnal variations of air densities from 190 to 280 km. as derived from the orbits of discoverer satellites .

variations in air density between day and night in the region 190 to 280 km are found to be small/less than about 25(/ . the presence of a possible region of local heating at about 220 km which disappears at night . the night-time density profile conforms with a constant scale height of 35/2/km.no definite variation of air density with latitude is evident apart from a possible increase of about 60(, which is indicated by rather limited polar-region data . for other latitudes and seasons a variation of less than about 20(is indicated .

.I 622

.T

scale height in the upper atmosphere, derived from changes in satellite orbits .

.A

king-hele, d.g. and hughes, k.m.

.B

r.a.e. tn space 4, 1962 .

.W

scale height in the upper atmosphere, derived from changes in satellite orbits .

the'density scale height'h in the upper atmosphere is a measure of the rate at which air density p varies with height y, being given by h-p//dp/dy/ . the value of h, although important because/with the molecular weight of the air/it determines the air temperature, has not as yet been well determined at heights above 200 km .

this note develops methods for finding h from the decrease in a satellite's perigee height and from the decrease in the orbital period of a satellite in a small-eccentricity orbit . these methods are then applied to all the 14 satellites found suitable for the purpose . the 44 values of h obtained, for heights of 200-450 km, represent an average over day and night and probably have errors/s.d./of 5-10(. it is found that, as solar activity declined between 1957 and 1961, h decreased greatly ..e.g.at height 275 km, h decreased from 60 km in early 1958 to height becomes much less rapid above 350 km, and are consistent with the supposition that h had low values, near 35 km, at heights near 250 km, for 1959-61 . the results could be greatly extended in scope and improved in accuracy if more accurate orbits were available for short-lifetime satellites .

.I 623

.T

on the coupling between heat and mass transfer .

.A

tewfik,o.e.

.B

j. ae. scs. 28, 1962, 1009.

.W

on the coupling between heat and mass transfer .

in mixtures of two different gases or liquids, one constituent will migrate spontaneously toward the warmer parts, and the other toward the colder parts . this phenomenon, known as the soret effect, and its converse the dufour effect, were discovered as early as 1856 and 1873 respectively . the two effects can also be considered as a simultaneous transport of mass and heat, or as a coupling between heat and mass transfer . the effects of this coupling have been neglected in all investigations of heat transfer in multicomponent flow systems so far,

on the a priori assumption that they are small . in a recent

publication however, it was shown that they can be large in

laminar-boundary-layer-type flows with helium injection .

turbulent-boundary-layer measurements and an analysis conducted

at the heat transfer laboratory clearly showed significant

effects of the coupling on heat transfer and adiabatic wall

temperature . from additional measurements, the results of which

are presented below, it is possible to separate the heat flux at the

model wall into one part depending on the temperature gradient

and a second part caused by the coupling . it is shown that

the latter exceeds the former, and hence the coupling may not

be neglected a priori without careful consideration .

.I 624

.T

cruise performance of channel-flow ground effect machines .

.A

strand,t.

.B

j. ae. scs. 29, 1962, 702.

.W

cruise performance of channel-flow ground effect machines .

the performance theory for high-speed air-cushion vehicles

operating in close proximity to the ground is developed . the

analysis is restricted to cruise flight of vehicles of rectangular

planform employing an air pressure seal between the ground and

the vehicle along the two streamwise sides . the variation of the optimum rearward deflection angle of the side jet pressure seal with speed for minimum overall power expenditure and maximum range is found . it is concluded that a mixed propulsion system (jet deflection plus propeller(s)) is required . volume flow and the corresponding fan pressure rise needed are also calculated . the maximum lift drag ratio is determined .

the maximum thickness ratios of the vehicles are considered to be large compared with the ground-height vehicle-length ratio . two-dimensional airfoil theory is employed to show that close to stagnation conditions exist below the vehicles . the lower-surface lift, pitching moment, and aerodynamic-center location are determined .

the flow over the upper surface is identified with flow over mounds . upper-surface lift coefficients are determined for typical mound shapes .

it is shown that high total lift coefficients are theoretically obtainable with almost zero induced drag . the conventional induced-drag power penalty is replaced by a sealing-air power expenditure, which is shown not to be excessive .

.I 625

.T

viscous and inviscid nonequilibrium gas flows .

.A

whalen,r.j.

.B

j. ae. scs. 29, 1962, 1222.

.W

viscous and inviscid nonequilibrium gas flows .

the condition of immediate freezing of the mass fraction of dissociated species of air at the equilibrium value behind the shock envelope prevails over a major portion of the flight spectrum associated with lifting re-entry vehicles . this is observed by means of order-of-magnitude considerations within the limits of the present knowledge of chemical reaction rates for the constituents of air . accordingly, investigations of the viscous and inviscid hypersonic flow about blunt and sharp leading edge slender bodies are made . the investigations are generalized to consider an arbitrary degree of dissociation in the ambient free stream . this condition is included in order to allow comparison with the flow field about a model in the test section of a hypersonic facility with dissociated air species present in the free stream .

inviscid frozen flow investigations are made for blunt and sharp leading edge slender body power-law geometries . the results indicate that the influence of a finite leading edge, in inducing a pressure field far downstream (/blast-wave/ analogy), is considerably diminished for this model . this conclusion is verified numerically by a characteristics solution for the hypersonic flow about a /sonic-wedge/ slab .

the viscous investigations consider the boundary-layer interaction problem with a frozen degree of dissociation . in this case, as in the inviscid analysis, the governing parameter is observed to be the ratio of the

dissociation energy to the free-stream kinetic energy . the influence of this parameter on the boundary-layer interaction

mechanism for a highly cooled, noncatalytic wall is presented .

the influence of a frozen flow field on skin friction and heat

transfer is also discussed .

finally, since higher mach number gas flows may be generated

in wind tunnel nozzles where dissociation nonequilibrium effects

are present, the possibility of employing expansions with a

controlled degree of dissociation as a technique for aerodynamic

simulation is presented .

.I 626

.T

some features of supersonic and hypersonic flow about

blunted cones .

.A

traugott,s.c.

.B

j. ae. scs. 29, 1962, 389.

.W

some features of supersonic and hypersonic flow about

blunted cones .

for a family of cones of various semiapex angles blunted by

spherical caps, shock shapes and surface pressure distributions

have been obtained from both the belotserkovskii method and

experiment . these results are used to study convergence to

conical flow . conditions leading to both overexpansion and

underexpansion on the surface with respect to the asymptotic

conical pressures are described as well as conditions leading to bow shock inflection points . conditions also exist for which a second shock may occur, or for which the sonic line cannot touch the body surface . the implications of these conditions for various blunt-body methods are discussed . for cones blunted in such a manner as to keep the flow entirely supersonic, the flow field is found to exhibit certain similarities with that for genuine blunting . this is related to the fact that the surface entropy layer for blunt bodies can be most influential, in determining surface pressure, in the interior of the flow field rather than near the surface .

.I 627

.T

flutter analysis of circular panels .

.A

rattayya,j.v.

.B

j. ae. scs. 20, 1962, 534.

.W

flutter analysis of circular panels .

the flutter problem of flat circular panels with edges elastically restrained against rotation has been formulated in terms of small-deflection plate theory . the panel is subjected to uniform all-round tension or compression in its middle plane, in addition to the supersonic compressible flow passing over its upper surface with still air below . linear piston theory is employed to predict the aerodynamic load on the vibrating panel .

the problem is investigated by a rayleigh-type analysis involving chosen modes of the panel as degrees of freedom .

in order to investigate the convergence of the solution, the flutter-mode shape of the clamped-edge panel has been expressed in a series form in powers of r cos o . the results of three-, four-, and five-term approximations have displayed oscillatory behavior with apparently rapid convergence of the solution .

.I 628

.T

thermal effects on a transpiration cooled hemisphere .

.A

gollnick,a.f.

.B

j. ae. scs. 29, 1962, 583.

.W

thermal effects on a transpiration cooled hemisphere .

an approximate method is used to obtain the injection distribution which would exist on an isothermal, transpiration-cooled hemisphere in a supersonic stream . this distribution is the same for both air and helium injection, and is independent of the blowing level . a model having this distribution was tested in the naval supersonic laboratory wind tunnel at a mach number of 3.53 . it is concluded that the design technique is reasonably accurate . data taken near the nose are compared with the theories for air and helium injection . the agreement in the case of the reduction in heat-transfer coefficient is good . the values of

insulated wall temperature obtained near the nose with
helium injection are 8 percent above the local
stagnation temperature, and largely independent of injection
rate . it is believed that this phenomenon may be
attributed to the thermal diffusion of the helium within
the boundary layer . air injection causes a slight
reduction in the insulated wall temperature . it is shown
that injection of either air or helium at the hemisphere
nose considerably reduces the heat flux at the surface .
the additional reduction in heat flux resulting from
helium injection as opposed to air injection, and
predicted by existing theory, is largely absent .

.I 629

.T

second-order effects in laminar boundary layers .

.A

maslen,s.h.

.B

a.i.a.a. j. 1963, 33.

.W

second-order effects in laminar boundary layers .
  second-order boundary layer disturbances are
due to the displacement of the main flow by
the boundary layer, surface curvature, freestream
vorticity, and slip . a procedure for finding
these is given for compressible flow of a perfect gas
having a classically similar boundary layer .

solutions are given for the flat plate and circular

cylinder and for the hypersonic axisymmetric

stagnation point . for the latter flow, the dominant

effect is that of vorticity, which increases

both shear and heat flux . for the plate or cylinder,

the same conclusion tends to hold for high

speed flow . the vorticity effect is governed by the

entire outer flow--not just the wall vorticity .

.I 630

.T

stagnation region in rarefied high mach number flow .

.A

cheng,h.k. and chang,a.l.

.B

a.i.a.a. j. 1963, 231.

.W

stagnation region in rarefied high mach number flow .

paper describes results of numerical solution of the viscous

shock-layer equations for axisymmetric stagnation region, using

the viscosity-temperature law with w=0.65, pr=0.71 and

y=1.25 . purpose is to establish applicability of the simple

approximation of w=1 (obtained earlier) to air at low reynolds

numbers and low ratios of wall temperature to stagnation

temperature . using a reference temperature (closely equal to

eckert's) to interpret the linear results, excellent agreement is found,

in the limit of, over a wide range of reynolds numbers,

covering fully merged shock layers as well as boundary layers with and without vorticity interaction . agreement with recent

experiments of ferri et al is as good as to be expected from

shock-layer approximation . paper provides valuable extension of the

applicability of the reference temperature concept .

.I 631

.T

low speed wind tunnel tests on a two dimensional aerofoil

with split flap near the ground .

.A

bagley,j.a.

.B

arc cp.568, 1961.

.W

low speed wind tunnel tests on a two dimensional aerofoil

with split flap near the ground .

pressure distributions have been

measured on a 10 thick two-dimensional

aerofoil of r.a.e.101 section fitted with

split flaps deflected at 15 and 55 .

measurements were made at two distances

above a ground plate, and also without

the ground plate . the results have been

integrated to give the sectional

lift, drag and pitching-moment coefficients .

.I 632

.T

calculated lift distributions in incompressible flow on some sweptback wings .

.A

bagley,j.a. and joyce,g.m.

.B

rae tn. aero.2836, 1962.

.W

calculated lift distributions in incompressible flow

on some sweptback wings .

in the course of a larger survey of some aerodynamic characteristics

of a family of sweptback wings, the low-speed lift distributions were

calculated . the 35 planforms considered cover a range of leading-edge

sweep angles from 55 to 70, and aspect ratios from 2 to 3.9 . the

results are given here, together with a comparison with other

calculations and with experimental results on one particular wing .

.I 633

.T

an extension of the method of generalised conical flows

for lifting wings in supersonic flow .

.A

portnoy,h.

.B

rae tn. aero.2849, 1962.

.W

an extension of the method of generalised conical flows

for lifting wings in supersonic flow .

the method of generalised conical

flows has previously been developed

subject to the condition that the

upwash divided by the streamwise

co-ordinate to the power k, where

k is the order of the conical flow, must

have vanishing (k+1)th derivative

with respect to the conical co-ordinate .

in the present note this restriction is removed .

the results are also used to

discuss the effect of the application of

the leading edge attachment condition

on the wing pressure and geometry .

.I 634

.T

effects of leading edge bluntness on flutter characteristics

of some square- planform double-wedge airfoils at a

mach number of 15 .4.

.A

goetz,r.c.

.B

nasa tn.d1487, 1962.

.W

effects of leading edge bluntness on flutter characteristics

of some square- planform double-wedge airfoils at a

mach number of 15 .4.

results are presented from a wind-tunnel

investigation in helium flow at a

mach number of 15.4 . the models were

square-planform, double-wedge, shaft-mounted

airfoils with leading- and trailing-edge

radii of 0, 1, 3, and 6 percent chord .

in general, the tests indicate that bluntness

effects on the model flutter

characteristics are stabilizing as the leading-edge

radius is increased from 0 to

destabilizing with further increase in

bluntness .

results of flutter calculations made

by using newtonian theory aerodynamics

and a combination of newtonian theory and

piston theory aerodynamics in

conjunction with an uncoupled two-mode analysis

are compared with experimental results .

the piston-theory results accurately

predicted flutter speeds for the models with

.I 635

.T

heat transfer and pressure distributions on a hemisphere-cylinder and a

bluff-afterbody model

in methane-air combustion products and in air .

.A

irving weinstein

.B

national aeronautics and space administration .

technical note d-1503

.W

heat transfer and pressure distributions on a hemisphere-cylinder and a

bluff-afterbody model

in methane-air combustion products and in air .

an experimental investigation has been made to indicate the validity of

using methane-air combustion products as the test medium for aerodynamic

heating and loading tests . tests were conducted on a

hemisphere-cylinder and on a bluff-afterbody model, both in methane-air combustion

products and in air alone, and covered a range of mach numbers from 6 to

the data showed that the nondimensional heating-rate distribution

along a hemisphere-cylinder as obtained in combustion products was in

good agreement with that obtained in air, and the results were in

reasonable agreement with theory . the stagnation-point heating rates

in air and in combustion products over the hemisphere-cylinder agreed

within 10 percent of the theoretical values . the pressure

distributions around a hemisphere-cylinder obtained from tests in combution

products were in good agreement with those obtained in air and could

be predicted by newtonian flow theory . the tests in combustion

products of a bluff-afterbody model produced nondimensional

heat-transfer coefficients which were in fair agreement with results

obtained in air .

.I 636

.T

pressure distribution induced on a flat plate at a free-stream

mach number of 1.39 by rockets exhausting upstream and downstream .

.A

abraham leiss

.B

national aeronautics and space administration

technical note d-1507

.W

pressure distribution induced on a flat plate at a free-stream

mach number of 1.39 by rockets exhausting upstream and downstream .

an experimental investigation was made of the pressures induced on

a flat plate at a free-stream mach number of 1.39 by a supersonic

rocket jet exhausting upstream and downstream . measurements of the

pressure distribution on a flat plate were made at zero angle of

attack for 11 different locations of the jet exhaust nozzle beneath

the wing . measurements were made at ratios of rocket-exit total

pressure to free-stream static pressure from 6 to 60 and at a reynolds

number per foot of approximately 10 times 10 to the power of 6 . the

rocket when exhausted upstream produced a strong shock that moved

further upstream with increasing rocket-exit total-pressure ratio .

positive incremental normal-force coefficients were obtained at all test

positions . data at 11 test positions are tabulated for rocket-on

and rocket-off pressure coefficients as well as for incremental pressure

coefficients for the 48 orifices of the flat plate for the range of

ratio of rocket-exit total pressure to free-stream static pressure

of the investigation . changing the location of the model with

respect to the plate had a negligible effect when the rocket was

varied in the chordwise direction, but the pressure coefficients

were reduced as the rocket was lowered away from the flat-plate wing .

.I 637

.T

an integral equation relating the general time-dependent lift and

downwash distributions on finite wings in subsonic flow .

.A

joseph a. drischler

.B

national aeronautics and space administration

technical note d-1521

.W

an integral equation relating the general time-dependent lift and

downwash distributions on finite wings in subsonic flow .

an integral equation for obtaining the unsteady air forces on finite

wings in subsonic compressible flow is presented . this equation

is applicable for any arbitrary time-dependent motion and can be

utilized for flexible as well as rigid wings . the approach involves

the derivation of an integral equation relating the unknown pressure

the form of the equation is such that it should lend itself readily to

modern high-speed computers for obtaining pressure distributions .

special cases of the integral equation are treated for two-dimensional

incompressible flow and are presented in an appendix .

.I 638

.T

longitudinal aerodynamic characteristics at low subsonic

speeds of a highly swept wing utilizing nose deflection

for control .

.A

spencer,b.

.B

nasa tn.d1482, 1962.

.W

longitudinal aerodynamic characteristics at low subsonic

speeds of a highly swept wing utilizing nose deflection

for control .

an investigation has been conducted

in the langley 7- by 10-foot transonic

tunnel at low subsonic speeds to determine

the longitudinal aerodynamic

characteristics associated with deflection of the

nose section of a highly swept delta

wing having an aspect ratio of 1.33 . in order

to illustrate the effectiveness of

this forward control, the longitudinal control

characteristics are also presented

for the wing with upper-and lower-surface

split flaps located at the trailing

edge .

comparison between the longitudinal

aerodynamic characteristics of the wing

utilizing the nose control and those of

the wing utilizing the upper-surface

split flap located at the trailing edge

indicated similar control effectiveness

for high control deflections (15) and

similar values of trimmed lift-drag ratio

with increasing lift coefficient . use

of the nose control, however, indicated a

lower value of trimmed angle of attack

for a given value of trimmed lift

coefficient than that realized from use of

the upper-surface split flap . further

reductions in trimmed angle of attack

for a given value of trimmed lift

coefficient may be realized from deflection

of the lower-surface split flap at the

wing trailing edge in combination with

the nose control and would be accompanied

by large reductions in lift-drag ratio .

.I 639

.T

analytical study of the tumbling motions of vehicles

entering planetary atmospheres .

.A

tobak,m.

.B

nasa tn.d1549, 1962.

.W

analytical study of the tumbling motions of vehicles

entering planetary atmospheres .

the tumbling motion of vehicles

entering planetary atmospheres is

analyzed . a differential equation

governing the tumbling motion, its

arrest, and the subsequent oscillatory

motion is obtained and identified

as the equation for the fifth painleve

transcendant . an approximate

analytical solution for the transcendant

is derived . comparisons with

results obtained from numerical

integration of the exact equations of

motion indicate that the solution for the

angle-of-attack history is

sufficiently accurate to be of practical use .

.I 640

.T

the design of structures to resist jet noise fatigue .

.A

b. l. clarkson

.B

j. royal aero. soc. 66, oct. 1962

.W

the design of structures to resist jet noise fatigue .

the design of structures to resist jet noise fatigue demands a

knowledge of a wide range of subjects from pure acoustics at one

hand to metal physics at the other . at the present time the various

aspects of the problem are not sufficiently well know

quantitatively for a purely theoretical design study to be made . never-

the-less a knowledge of the behaviour of typical forms of

construction in noise environments can be used with a limited amount of theoretical work to indicate tne most efficient types of structure . this approach to the problem is adopted in this lecture as it seems to be the most promising one available at the moment . it must be emphasized, however, that although some progress has been made in dicsovering the behaviour of a structure subjected to noise it is not possible to estimate the life of any component at the drawing board stage . some prototype strain measurements and proof testing are therefore essential if one is to prove the integrity of the design .

within the structural limits of single skin construction set in this lecture the main conclusion to be reached is that no reasonable estimate of fatigue life can yet be made in the drawing board stage of a structure . nevertheless, a study of the form of behaviour of typical structures has led to a theoretical simplification of the problem of skin vibration . from this it has been possible to suggest an optimum deisgn for a skin stiffened by stringers . a suggestion for an optimum design of skin and rib for control surfaces to minimise stresses at the rib-skin intersection is put forward but no experience can check this yet .

the most resonable basis for the future estimation of fatigue life of a component appears to be the /random/ s-n curve and consierable effort should be made to obtain the necessary test data .

the life expectation of a new design will be uncertain and some proof testing is essential if the integrity of structure in high noise levels (150 db) is to be guaranteed .

.I 641

.T

reduction of the clamped plate to two membrane problems

with an application to uniformly loaded sectors .

.A

morley,l.s.d.

.B

rae r.struct.277, 1962.

.W

reduction of the clamped plate to two membrane problems

with an application to uniformly loaded sectors .

the clamped plate problem in

the classical theory for the small

deflection bending of flat plates

is reduced to the solution by variational

methods of two successive membrane

problems . the first requires the least

square minimisation of the average

curvature of the deflected surface while

the second problem concerns the

integral of the gaussian curvature . there

is a similar reduction for extensional

problems where the boundary tractions

are specified .

the method is demonstrated by

giving three distinct solutions to the

problem of the clamped sector under

a uniformly distributed load . one

solution is of special interest because

it is derived from a single membrane

problem . numerical data are given .

.I 642

.T

the buckling strength of a uniform circular cylinder

loaded in axial compression .

.A

sobey,a.j.

.B

rae r. struct,279, 1962.

.W

the buckling strength of a uniform circular cylinder

loaded in axial compression .

the theoretical estimation of the buckling strength of a cylinder

loaded in axial compression is improved by the use of a more

representative deflected form for the buckled cylinder than has

previously been used . kempner's buckling strength for dead weight

loading is reduced by 18 . the presentation of the magnitude and

distribution of the constraint system required to maintain the mode is

novel and instructive .

.I 643

.T

an investigation of wing-aileron flutter using ground

launched rocket models .

.A

gaukroger,d.r. and curran,j.k.

.B

rae tn. struct.308, 1962.

.W

an investigation of wing-aileron flutter using ground

launched rocket models .

control surface flutter of the

wing torsion-control rotation type has

been investigated for an unswept wing

with an under-massbalanced, half span,

outboard aileron . thirteen pairs of

wings were tested, using ground launched

rocket driven vehicles, and a range of

values of aileron natural frequency

was covered . the test results showed

considerable scatter, but enabled

upper and lower limits of a flutter

boundary to be determined approximately .

it was established that aileron flutter

could be eliminated on the models

tested provided the aileron frequency

exceeded the wing torsional frequency

by 20 per cent or more . in this

condition the models were also free from

single degree of freedom flutter

.I 644

.T

a study of the cantilever square plate subjected to a uniform loading .

.A

leissa,a.w. and niedenfuhr,f.w.

.B

j. aero. sc. 29, 1962.

.W

a study of the cantilever square plate subjected to

a uniform loading .

plate problems involving free edges have been historically difficult to solve, particularly when two free edges are adjacent, resulting in a free corner . the cantilevered square plate subjected to a transverse loading is one such problem for which an exact solution has not been achieved .

in the present paper results obtained by various approximate methods are presented for this problem for the case of a uniform loading . solutions obtained by the authors using the technique of point matching and the rayleigh-ritz method are compared with previously published finite-difference and experimental results and with bernoulli-euler beam and plane-strain approaches . numerical results for deflections, slope components, bending and twisting moments, and transverse distributed shears are presented for a relatively fine gridwork of points on the plate boundary and within the interior . the antielastic curvature is exhibited by all methods except beam theory . all methods present the interesting conclusion that the free edge deflection is greater when the plate is treated as a plate rather than a beam .

.I 645

.T

thermodynamic coupling in boundary layers .

.A

baron,j.r.

.B

ars space flight rep. to the nation, 2206-61, 1961.

.W

thermodynamic coupling in boundary layers .

experimental results gathered in recent years for binary

mixture mass transfer models are shown to yield consistent

evidence of discrepancies with analytic considerations . specifically,

measured recovery temperatures are appreciably higher than those

predicted ,. while heat transfer coefficients are satisfactorily

reproduced . it is shown on the basis of both approximate and exact

solutions for plates and stagnation points that the discrepancies in

previous results are related to thermal diffusion effects, a major

influence being apparent in application of the surface boundary

condition for an adiabatic wall . as a result, some reexamination is

necessary of past criteria for mass addition effects as they pertain to

specific injected media . a prime example is the /equivalence/ of

helium and air as coolants despite the heretofore suggested preference

for low density injectants on a perfect gas basis . ref. 16 .

.I 646

.T

thermal diffusion effects on energy transfer in a turbulent

boundary layer with helium injection .

.A

tewfik,o.e., eckert,e.r.g. and shirtliffe,c.j.

.B

to be presented at the 1962 heat transfer and fluid mech.inst. seattle,

.W

thermal diffusion effects on energy transfer in a turbulent

boundary layer with helium injection .

a circular cylinder with two-inch

diameter and with a porous wall

fabricated out of woven wire material was

aligned with its axis parallel to an air

stream with approximately 100 ft sec

velocity . helium gas was injected into

the turbulent boundary layer through

the cylinder walls at a uniform rate in

the range 1.55 x 10 to 1.08 x 10

of the free stream mass velocity . the

local energy transfer along the cylinder

was measured at various values of

the wall temperature level for the

situation that the energy flows from the

cylinder to the boundary layer and

vice versa . the results showed clearly

that the wall temperature for zero

energy transfer - the adiabatic wall

temperature - was larger than the free

stream temperature by up to about 40 f,

although viscous dissipation effects

are negligible . this temperature excess

increases with increasing injection

rate and is independent of reynolds

number .

an analysis in which the laminar

sublayer is treated as couette flow

with helium injection and which includes

thermal diffusion in this layer is

formulated . the results show appreciable

thermal diffusion effects on

adiabatic wall temperature, increasing it

over its value for zero injection

by amounts of the same order of magnitude

as found by measurements . thermal

diffusion however has negligible effects

on the heat transfer coefficient .

its effects on the concentration and

temperature distribution are discussed

and are shown to produce appreciable

modifications in the latter .

.I 647

.T

bending of a uniformly loaded rectangular plate with

two adjacent edges and the others either simply supported

or free .

.A

huang,m.k. and conway,d.

.B

j.app.mech. 1952, 451.

.W

bending of a uniformly loaded rectangular plate with

two adjacent edges and the others either simply supported

or free .

the distribution of deflection and bending moment in a

uniformly loaded rectangular plate having two adjacent

edges clamped and the others either simply supported or

free, are obtained by a method of superposition .

numerical values are given for square plates and, in one case, the

results are compared with those obtained by another

method .

.I 648

.T

the approximate analysis of certain boundary value

problems .

.A

conway,h.d.

.B

j. app. mech. 1960, 275.

.W

the approximate analysis of certain boundary value

problems .

a simple method is given which is suitable

for the approximate analysis of certain

boundary-value problems, including, for

example, the small deflections of clamped

plates and the torsion of prismatic bars .

the analysis is particularly simple and lends

itself well to the use of the digital computer .

the method is applied here to four

problems, the uniformly loaded, clamped square,

and equilateral-triangle plates, and the

torsion of bars of square and hexagonal cross

section . the results agree well with

the exact solutions, where these are known .

.I 649

.T

the hovercraft - a new concept in maritime transport .

.A

crewe,p.r. and eggington,w.j.

.B

trans. roy. inst. naval arch. 1960.

.W

the hovercraft - a new concept in maritime transport .

the hovercraft is the first operational british

project in the ground-effect machine field .

although there has, for a number of years, been

a tentative searching after the principles

underlying such machines, it is only now that their

possibilities as commercial transport and

service craft are beginning to be developed .

since the hovercraft is a new vehicle, the appearance

of the saunders-roe sr-n1, a manned
experimental craft, excited considerable public attention
and there have been a number of
descriptive articles in the press . papers of a more
technical type, on ground-effect machines,
are now beginning to appear and it is to be expected
that these will rapidly increase in number,
especially since american interest in both the commercial
and defence fields is expanding fast .

the authors of the present paper have, therefore,
concentrated attention upon features
about which they had something personal to say, and
which they consider to be of particular
significance for assessing the possibility of the
hovercraft becoming important in maritime
transport . these features
are ..- the hovercraft as a fundamentally new
principle in the transport field .
the powering requirements and resistance
characteristics .
the likely operating costs of hovercraft
in comparison with other forms of maritime transport .
in addition, relatively brief descriptions of
the history and the current work being undertaken
on the ground-effect machine and of the design,
construction, and testing of the saunders-roe
sr-n1 are provided . the final section discusses

outstanding problems and some future

possibilities .

.I 650

.T

some design problems of hovercraft .

.A

stanton-jones,r.

.B

inst. aero. sc. paper 61-45, 1961.

.W

some design problems of hovercraft .

analysis of the influence various aerodynamic parameters have on

the performance of a simple peripheral jet system . power weight

ratio, lift drag ratio, and effect of jet angles and thickness are

each considered . structural requirements, optimum cushion

pressure, and dynamic stability over waves are examined and then

related to the economics of ground-effect machine operation .

.I 651

.T

heat transfer to separated and reattached subsonic

turbulen flows obtained downstream of a surface step .

.A

seban,r.a., emery,a. and levy,a.

.B

j. ae. scs. 1959, 809.

.W

heat transfer to separated and reattached subsonic turbulen flows obtained downstream of a surface step .

local heat-transfer coefficients and recovery factors are presented for separated and reattached turbulent flows as obtained by a downward step in an otherwise flat surface in a two-dimensional, subsonic, air flow . the region downstream of the step, the focus of this investigation, contained a region of separated flow with reattachment at about five step heights downstream, followed by a section of reattached flow . the salient feature of the results is the maximum in the local heat-transfer coefficient at the reattachment point, with values thereof diminishing in the separated region and also in the reattached region, where they tend toward values characteristic of turbulent boundary-layer flow . it is found that for most of the region the heat-transfer coefficient depends on the velocity to about the 0.8 power, though a decreased dependence may exist in the separated region . recovery factors have the characteristically low values associated with separated flows, and do not attain values typical of turbulent boundary-layer flows within the downstream lengths available .

.I 652

.T

pressure distribution on two dimensional wings near the ground .

.A

bagley,j.a.

.B

rae r. aero.2625, 1960.

.W

pressure distribution on two dimensional wings near

the ground .

a simple method of calculating

the pressure distribution in

incompressible flow on two-dimensional aerofoils

of arbitrary section at moderate

distances from the ground is developed .

comparisons with an /exact/ potential

flow solution, and with measurements

on a 10 thick aerofoil of rae.101

section, provide a satisfactory

verification of the adequacy of the method ,.

but it is shown that it is necessary

to take account of the boundary layer

on the aerofoil in the calculations .

.I 653

.T

transient magnetohydrodynamic duct flow .

.A

lundgen,t.s., atabeck,b.h. and chang,c.c.

.B

phys. fluids 4, 1961, 1006.

.W

transient magnetohydrodynamic duct flow .

parallel flow of an electrically conducting viscous

incompressible fluid in a rectangular duct with transverse magnetic field is considered . the walls

of the duct which are parallel and perpendicular

to the imposed magnetic field are taken to be

nonconducting and perfectly conducting, respectively .

assuming the fluid to be at rest at the initial

moment, exact solutions for the velocity and magnetic

field components are obtained in the form of

convolution integrals taking the longitudinal pressure

gradient as an arbitrary given function of time .

later, taking a step function for the pressure gradient,

these expressions are integrated . for this case,

the effect of the strength of the imposed magnetic

field on the development behavior of the flow is

studied . it is found that except for very large magnetic

fields, the flows are over damped .

.I 654

.T

on the propagation and structure of the blast wave .

part 1.

.A

sakuri,a.

.B

j. phys. soc. japan, 9, 1954.

.W

on the propagation and structure of the blast wave .

part 1.

as a continuation of part 1 (j. phys. soc. japan 8 (1953) 662), the second approximation for the propagation and structure of a blast wave is now discussed . the solution for r=1.4 is obtained by a numerical method, using the results of the first approximation obtained in part 1 . by use of this solution, u-r curves, distance-time curves and the changing feature of distributions of velocity, pressure and density behind the shock front are discussed .

further, the approximate solution of the equation is discussed by a refinement of the wkb method due to imai .

.I 655

.T

effects of boundary layer displacement and leading edge bluntness on pressure distribution, skin friction, and heat transfer of bodies at hypersonic speeds .

.A

bertram, m.h. and henderson, a.

.B

naca tn 4301, july 1958 .

.W

effects of boundary layer displacement and leading edge bluntness on pressure distribution, skin friction, and heat transfer of bodies at hypersonic speeds .

results are presented of an investigation to determine the effect of boundary-layer displacement and leading-edge bluntness on surfaces in hypersonic flow . the presence of the boundary layer and the blunt leading edge induce pressure gradients which in turn affect the skin friction and heat transfer to the surface . methods for predicting these phenomena on two-dimensional surfaces are given and a brief review of recent three-dimensional results is presented .

.I 656

.T

departure from dissociation equilibrium in a hypersonic nozzle .

.A

bray, k.n.c.

.B

a.r.c. 19, 983, march 1958 .

.W

departure from dissociation equilibrium in a hypersonic nozzle .

the equations of motion for the flow of an ideal dissociating gas

through a nearly conical nozzle have been solved numerically, assuming

a simple equation for the rate of dissociation, and a number of

different values of the rate constant . the results of these

calculations suggest that deviations from dissociation equilibrium will

occur in the nozzle if the rate constant lies within a very wide range

of values . they also suggest that once such a deviation has begun the

gas will very rapidly/freeze/, so that the dissociation fraction will

remain almost constant if the flow is expanded further, or even if it

passes through a constant area test section . an approximate method

of solution, making use of this property of sudden/freezing/of the flow,

has been developed and applied to the problem of estimating the

deviations from equilibrium under a wide range of conditions . if all

the assumptions made in this report are accepted, then lack of

dissociation equilibrium may be expected in the working sections of

hypersonic wind tunnels and hypersonic shock tubes .

it is shown, however, that the flow behind a normal shock wave in such a wind tunnel will not be greatly affected by any freezing that may take

place in the nozzle upstream of the shock wave . even so, the stand-off

distance of a shock wave in front of a blunt model may be quite

sensitive to deviations from equilibrium .

.I 657

.T

interferometric studies of supersonic flows about truncated

cones .

.A

giese,j.h. and bergdolt,v.e.

.B

j.app.phys. 24, 1953, 1389.

.W

interferometric studies of supersonic flows about truncated

cones .

fringe shifts on interferograms of flows at m=2.45

about variously truncated 15 (half-angle) cone

cylinders in free flight in a pressurized range have been

examined for similarity of the flow fields, occurrence of

scale effects, and convergence to conical flow . it was

found that flows over similar objects with equal tip

reynolds numbers were similar and that convergence

to conical flow occurred before the disturbance at the

tip had been reflected the second time along

characteristics to the body . density distributions have been

determined, and a number of comparisons have

been made with theoretical predictions .

.I 658

.T

review of panel flutter and effects of aerodynamic noise

part i.. panel flutter .

.A

l. e. goodman and j. v. rattayya

.B

university of minnesota, minneapolis, minnesota

.W

review of panel flutter and effects of aerodynamic noise

part i.. panel flutter .

with the development of high-speed aircraft and missiles, vibration

of panels has become a problem of practical significance . many

of the failures of the early german rockets after attaining supersonic

speed have been attributed to the development of such panel

oscillations . it appears this

phenomenon is not of much concern in the subsonic

speed range., however, in the supersonic speed range panels may develop

oscillations which cause instability of the structure . this effect has

been exhibited experimentally under controlled laboratory conditions

motion is limited and buckling may not be a serious design problem .

in these cases panel flutter is still of importance because of its

effect on the fatigue life and the allowable stresses for design of

the panel material .

the oscillations of panels may be due either to aerodynamic force

induced by the motion of the panel, or to aerodynamic noise, or

buffeting (irregular motion induced by turbulence in the flow) .

the interaction between aerodynamic forces and panel motions, usually

referred to as /panel flutter,/ has been investigated by several workers

in recent years . since the problem is too complex to be dealt with

in its entirety, simplifying assumptions have been made in these

investigations . the literature is marked by a certain degree of

controversy over the validity of these assumptions and the applicability of

the results obtained . a brief review of the literature with reference

to several of the approximations made and the results obtained follows .

.I 659

.T

nonuniform shear flow past cylinders .

.A

murray,j.d.

.B

q. j. mech. app. math. 10, 1957, 406.

.W

nonuniform shear flow past cylinders .

a general method is described whereby

an approximation of any desired degree

of accuracy to the stream functions for

two types of variable shear flows past

finite cylinders can be obtained . the two

shear distributions in the free stream can

be approximated to the linear shear

distribution and the shear present in an

unretarded incompressible boundary layer

respectively . in every case the stagnation streamline is displaced from the position

opposite the line of symmetry of the

cylinder, and general expressions are

obtained for this displacement . the line of

symmetry may be in the direction of

or perpendicular to the direction of flow .

the two particular examples cited are

those of a general elliptic cylinder and

cylinders of the form where and being the polar

coordinates, and 2p the maximum width

of the cylinder .

.I 660

.T

the fundamental solution for small steady three dimensional

disturbances to a two dimensional parallel shear flow .

.A

lighthill,m.j.

.B

j. fluid mech. 3, 1957, 113.

.W

the fundamental solution for small steady three dimensional

disturbances to a two dimensional parallel shear flow .

after a brief review of methods of calculating the flow

fields produced by disturbances in rotational basic flows,

the author points out a fundamental difficulty in the

treated as a perturbation of the disturbance field that

would occur if the basic flow were uniform) .. slow attenuation of the secondary-flow disturbance with distance from the obstacle . the author conjectured (same j. 1 the trouble was caused by nonuniform validity of the approximation sequence in the region far from the obstacle . the analogy with /stokes' and whitehead's paradoxes/ is mentioned, and a solution analogous to oseen's is suggested, one in which disturbances, but not the shear, are assumed to be small . in this paper, such a solution is found, and is shown to overlap with the small-shear, secondary-flow solution . the basic flow is a parallel, steady, inviscid, two-dimensional shear flow . the / fundamental solution/ due to a weak source is sought .

the method of fourier transforms is used . simple solutions are found for a uniformly sheared basic flow (where the result coincides with the secondary-flow solution) and for an exponential basic-flow profile . in the general case it is assumed that the parallel basic flow becomes uniform at, where the x-axis lies in the flow direction .
the character of the solution is determined by studying its hankel transform, especially for the class of flows where the total variation of the basic stream speed v(y) is small . an interpretation in terms of images, due to m. b. glauert, is given, and finally the relationship of the present work to theories of the displacement of the stagnation streamline (displacement effect of pitot tubes) is discussed .

.I 661

.T

summary of laminar boundary layer solutions for

wedge-type flow over convection and transpiration cooled

surfaces .

.A

livingood,j.n.b. and donoughe,p.

.B

naca tn.3588, 1955.

.W

summary of laminar boundary layer solutions for

wedge-type flow over convection and transpiration cooled

surfaces .

a summary of exact solutions of the

laminar-boundary-layer equations

for wedge-type flow, useful in estimating

heat transfer to such

arbitrarily shaped bodies as turbine blades,

is presented . the solutions are

determined for small mach numbers and

a prandtl number at the wall of 0.7 ,.

ranges of mainstream pressure gradients

and rates of coolant flow through

a porous wall are considered for the

following cases .. (1) small

temperature changes in the boundary layer

along a constant- and along a

variable-temperature wall, and (2) large temperature changes in the boundary layer

along a constant-temperature wall .

dimensionless forms of heat-transfer

and friction parameters and

boundary-layer thicknesses are tabulated .

the results indicate that

coolant emission and increased stream-to-wall

temperature ratios diminished

the friction and heat transfer for a

constant wall temperature . for a

variable wall temperature with small

temperature differences in the

boundary layer, the friction was unaffected,

but the heat transfer was greatly

increased for increased wall-temperature

gradient . heat-transfer results

in the literature reveal that transpiration

cooling is much more effective

for prandtl numbers of the order of 5.0 than for 0.7 .

.I 662

.T

theoretical and experimental investigation of

aerodynamic-heating and isothermal heat transfer parameters on

a hemisphere nose with laminar boundary layer at supersonic

mach numbers .

.A

stine,h.a. and wanlass,k.

.B

naca tn.3344, 1954.

.W

theoretical and experimental investigation of

aerodynamic-heating and isothermal heat transfer parameters on

a hemisphere nose with laminar boundary layer at supersonic

mach numbers .

the effect of a strong, negative

pressure gradient upon the local

rate of heat transfer through a laminar

boundary layer on the isothermal

surface of an electrically heated,

cylindrical body of revolution with a

hemispherical nose was determined from

wind-tunnel tests at a mach number

of 1.97 . the investigation indicated

that the local heat-transfer

parameter, based on flow conditions

just outside the boundary layer,

decreased from a value of 0.65 0.10 at

the stagnation point of the

hemisphere to a value of 0.43 0.05 at the

junction with the cylindrical

afterbody . because measurements of the

static pressure distribution over

the hemisphere indicated that the local

flow pattern tended to become

stationary as the free-stream mach number was increased to 3.8, this distribution of heat-transfer parameter is believed representative of all mach numbers greater than 1.97 and of temperatures less than that of dissociation . the local heat-transfer parameter was independent of reynolds number based on body diameter in the range from 0.6x10 to 2.3x10 .

the measured distribution of heat-transfer parameter agreed within theoretical distribution calculated with foreknowledge only of the pressure distribution about the body . this method, applicable to any body of revolution with an isothermal surface, combines the mangler transformation, stewartson transformation, and thermal solutions to the falkner-skan wedge-flow problem, and thus evaluates the heat-transfer rate in axisymmetric compressible flow in terms of the known heat-transfer rate in an approximately equivalent two-dimensional incompressible flow .

measurements of recovery-temperature

distributions at mach numbers
of 1.97 and 3.04 yielded local recovery
factors having an average value
of 0.823 0.012 on the hemisphere which
increased abruptly at the shoulder
to an average value of 0.840 0.012 on
the cylindrical afterbody . this
result suggests that the usual representation
of the laminar recovery
factor as the square root of the prandtl
number is conservative in the
presence of a strong, accelerating pressure gradient .

.I 663

.T

viscous flow along a flat plate moving at high speeds .

.A

kuo,y.h.

.B

j.ae.scs. 23, 1956, 125.

.W

viscous flow along a flat plate moving at high speeds .

by the distortion of coordinates, it is shown that, in the case
of supersonic viscous flow past a flat plate, the boundary-layer
and simple wave theories can be combined to give a complete
representation of the velocity and pressure fields . consistent
first-order solutions are considered . an expression for the
induced pressure on the plate, correct to the second order, is

obtained . at high mach numbers the important parameter

satisfies the hypersonic similarity law ,. and for arbitrary mach and

reynolds numbers and for different gases, the theoretical curve

correlates closely the experimental data . asymptotic shock

curve and skin-friction coefficient are also deduced, but the

experimental verifications are yet to be made .

.I 664

.T

the boundary layer on a flat plate in a stream with

uniform shear .

.A

murray,j.d.

.B

j. fluid mech. 11, 1961, 309.

.W

the boundary layer on a flat plate in a stream with

uniform shear .

the incompressible laminar boundary layer

on a semi-infinite flat plate is

considered, when the main stream has uniform

shear . a solution is obtained for

the first two terms of an asymptotic solution

for small viscosity . it is shown that

one of the principal effects of free-stream

vorticity is to introduce a modified

pressure field outside the boundary-layer region .

.I 665

.T

on the theory of hypersonic gas flow with a power law

shock wave .

.A

sychev,v.v.

.B

j. app. math. mech. 24, 1960, 756.

.W

on the theory of hypersonic gas flow with a power law

shock wave .

plane and axisymmetric hypersonic gas flows are considered with shock

waves of very great intensity that have a power-law form . on the basis

of an investigation of the portions of the flow with high entropy

adjoining the surface of the body (not necessarily for a shock wave of the

given form) it is shown that the use in the flow problem of the exact

solution for the corresponding unsteady self-similar gas motion requires

a supplementary refinement of the thickness of the high entropy layer .

a method is shown for introducing such a correction and constructing the

shape of the body contour, on which is to be applied the pressure

distribution obtained on the basis of the theory of small disturbances .

.I 666

.T

blunt body heat transfer at hypersonic speed and low reynolds numbers .

.A

ferri, a. zakkay, v. and ting, l.

.B

j. aero. sc. v. 28. p. 862. 1961 .

.W

blunt body heat transfer at hypersonic speed and low reynolds numbers .

an analytical method for the determination of effect of shock curvature

on heat transfer in the region of the nose has been developed . it is

shown that for practical body shape the viscous terms in the

navier-stokes equations are not important in the region of the flow far from

the wall, and the displacement thickness can be neglected . then the

flow can be approximately represented by an inviscid-flow solution

having as boundary conditions the body shape, which is not affected by

the reynolds number, and by a boundary-layer type of flow near the

wall, having appropriate boundary conditions . this approach permits us

to determine the heat transfer in the region of the nose even at very

low reynolds numbers .

experimental results are presented . the experimental results agree

with the values given by the analysis .

.I 667

.T

hypersonic shock layer theory of the stagnation region at low reynolds

number .

.A

cheng, h.k.

.B

proc. 1961, heat transfer and fluid mech. inst. stanford. univ. press.

1961 . p.161

.W

hypersonic shock layer theory of the stagnation region at low reynolds

number .

cheng, h.k.

hypersonic flow at low reynolds number is studied utilizing the shock-layer concept . the present formulation takes into account the salient features of the transport processes within the shock layer in a manner consistent with the shock-layer approximation . the rankine-hugoniot shock relations are modified to include contributions due to heat conduction and viscous effects immediately behind the shock . the specific problem of an axisymmetric stagnation region is treated . the flow regimes for this problem can be classified according to whether or not the transport effects are important immediately behind the shock . in one regime where the ordinary rankine-hugoniot relations hold across the shock, the vorticity-interaction theory based on the boundary-layer approximation is shown to be sufficient . in the other regime where the rankine-hugoniot relations have to be modified but the continuum-flow model applies, an approximate, an analytical solution is obtained . this solution reveals a substantial reduction of the temperature behind the shock and of the shock stand-off distance in the presence of strong surface cooling . the present study is intended to provide a knowledge to bridge the gap between the free-molecule flow regime and that of the boundary layer via the continuum theory . in this respect, the solution obtained appears to be satisfactory in that it yields the correct free-molecule limits for the skin friction and surface-heat transfer rate .

.I 668

.T

measurements of stagnation point heat transfer at low reynolds number .

.A

ferri,a. and zakkay,v.

.B

j. ae. scs. 28, 1962, 847.

.W

measurements of stagnation point heat transfer at low

reynolds number .

measurements of stagnation point heat transfer are presented

in the reynolds number range between the free molecular flow

and the range where modified boundary layer theory still applies .

the measurements are compared with the analytical methods

set forth by ferri, zakkay, and ting . the results show smooth

transition between the two regions and indicate that the

predicted reynolds number for which the modified boundary layer

theory can be used is in agreement with experiments . in the

lower range of reynolds number the ratio of

decreases and reaches a value of 1 at a reynolds number of 40 .

.I 669

.T

subsonic potential flow past a sphere inside a cylindrical duct .

.A

william l. haberman

.B

david taylor model basin, carderock, md.

.W

subsonic potential flow past a sphere inside a cylindrical duct .

the subsonic potential flow of a compressible fluid past a sphere in an infinite medium was first determined by rayleigh . subsequently,

caplan and tamada extended the solution to include the fourth power

of the mach number . to the author's knowledge, no solution for

subsonic flow past a sphere in a finite medium has been published . it

is the purpose of this note to present a solution for subsonic

potential flow past a sphere inside a circular cylindrical duct .

.I 670

.T

on blunt-body heat transfer at hypersonic speed and low reynolds

number .

.A

ferri, a. zakkay, v. and ting, l.

.B

j. aero. sc. v. 29. p. 882, 1962 .

.W

on blunt-body heat transfer at hypersonic speed and low reynolds

number .

a discussion of differences arising between experimental and analytical

results, in particular those due to inconsistencies introduced in the

presentation of data and the way the comparison is made .

.I 671

.T

pressure and boundary-layer measurements on a two dimensional wing at

low speed .

.A

brebner, g.g. and bagley, j.a.

.B

a.r.c. r + m 2886, july, 1962 .

.W

pressure and boundary-layer measurements on a two dimensional wing at

low speed .

results are given of pressure measurements and boundary-layer traverses

on a two-dimensional wing with 10 per cent rae 101 section at reynolds

numbers of 1.6x10 and 3.2x10 . these results which have been integrated

to give lift, drag and aerodynamic-centre characteristics, are used to

check some calculation methods for the growth of the turbulent boundary

layer and for the effect of a known boundary layer on the pressure

distribution .

it is concluded that the calculation of the boundary layer still needs a

little refinement before it is accurate enough to predict viscosity

effects on pressure distribution, lift, drag and aerodynamic center,.but

that these effects can be calculated if the actual boundary-layer

characteristics are known .

.I 672

.T

tunnel interference effects .

.A

pankhurst, r.c. and holder, d. w.

.B

wind tunnel techniques, chapter 8 pitman. 1952 .

.W

tunnel interference effects .

the problems of solid blockage, wake blockage, lift effect, and the

influence of boundary constraint at high mach number are considered in detail . corrections are given for various open and closed tunnels, rectangular, circular and octagonal, and different speeds, two and three dimensional flows, with several aerofoils and wings . other interferences include the wall boundary layer, gradient of static pressure and problems with the working fluid used .

.I 673

.T

investigation of full scale split trailing edge wing flaps with various chords and hinge locations .

.A

wallace,r.

.B

naca r.539, 1935.

.W

investigation of full scale split trailing edge wing flaps with various chords and hinge locations .

an investigation was conducted in the n. a. c. a. full-scale wind tunnel on a small parasol monoplane equipped with three different split trailing-edge wing flaps . the object of the investigation was to determine and correlate data on the characteristics of the airplane and flaps as affected by variation in flap chord, flap deflection, and flap location along the wing chord . the chords of the flaps were 10, 20, and 30 percent of the wing chord and each flap was tested at deflections from 0 to 75 when located successively at 68, 80, and 88.8 percent of the wing chord aft of the leading edge . the investigation included force tests, pressure-distribution tests, and downwash surveys . the results give the lift, the drag, and the pitching-moment characteristics of the airplane, the flap forces and moments, the pressure distribution over the flaps and wing at one section, and the downwash characteristics of the flap and wing combinations .

an increase in flap chord or distance of the flap from the leading edge of the wing increased the lift of the airplane but had an adverse effect on the wing pitching moment . the ld ratio of the airplane decreased with increase in flap deflection or flap chord . flap normal-force coefficients were primarily a function of flap deflection and were relatively independent of flap chord, hinge-axis location, and airplane attitude . the location of the flap center of pressure in percentage of flap chord aft of the hinge axis remained practically constant irrespective of airplane attitude and of flap deflection, chord, or location . flap hinge-moment coefficients varied with a power of flap chord greater than the square so that with regard to hinge moments narrow flaps were the most efficient in producing a given increase in lift .

split trailing-edge flaps materially affected the magnitude and distribution of pressures over the entire wing profile . at low angles of attack the predominant effect of the flaps was to increase positively the lower-surface pressures ,. at high angles of attack, to increase negatively the upper-surface pressures . downwash surveys

indicated that horizontal tail planes located above the wing

chord line would be more effective than those below the

chord in counteracting the increased diving moment of

the airplane with flaps deflected .

.I 674

.T

the shapes and lift-dependent drags on some sweptback

wings designed for m= 1. 2.

.A

bagley,j.a. and beasley,j.a.

.B

rae r. aero.2620, 1959.

.W

the shapes and lift-dependent drags on some sweptback

wings designed for m= 1. 2.

the camber and twist distributions

needed to produce a constant

span-wise -distribution and certain linear

chordwise load distributions have

been calculated by linearised supersonic

theory at for a set of 34

thin sweptback wings . the wing planforms

cover a range of aspect ratios

from 2.0 to 3.5 and leading-edge sweep

angles from 55 to 70 . both leading

and trailing edges are subsonic at the

design mach number, and the

slenderness parameter is between 0.19 and 0.40 .

the lift-dependent vortex and wave

drags associated with these loadings

have also been calculated, and appear not

to be excessive in almost all the

cases considered .

.I 675

.T

pressure distribution and surface flow on 5(and 9(thick wings with

curved tip and 60degree sweepback .

.A

garner, h.c. and walshe, d.e.

.B

a.r.c. 20,982, r + m 3244. may 1959 .

.W

pressure distribution and surface flow on 5(and 9(thick wings with

curved tip and 60degree sweepback .

extensive tables are given of pressure coefficients measured at reynolds

numbers from 1.3x10 to 3.9x10 on two half-models of identical planform

with 5(rae 101 and 9(rae 101 streamwise sections . the planform of

aspect ratio 3.899 has a straight trailing edge with 60degree of

sweepback, constant chord over most of the span and a parabolic outer portion

of the leading edge curving to a pointed tip . the overall wing

characteristics are obtained from integrated normal pressures and are

compared with lifting-surface theory .

the low-speed experimental pressure distributions and surface oil-flow

patterns are analysed and discussed in relation to the onset of separation and the distinct vortex flows that develop at high incidence . series of contrasting upper-surface isobars illustrate some features of the different stalling processes of the two wings . the direct influence of the main vortex on local surface pressures is assessed in general terms . a fuller appraisal of secondary surface flow is obtained from the oil patterns, observations in water and measurements of high suction near the trailing edge .
studies of the extent of leading-edge stall and location of part-span vortices, in particular two simultaneous leading-edge vortices on the thinner wing, follow from further analysis of local surface pressures . after a detailed discussion of the effect of reynolds number and the distinct types of separated flow, a few results with leading-edge roughness are considered in relation to scale effect on separation and the extensive influence of part-span roughness .

.I 676

.T

a simple method for calculating the span and chordwise loading on straight and swept wings of any aspect ratio at subsonic speeds .

.A

kuchemann, d.

.B

r + m 2935, r.a.e. rep. aero. 2476, a.r.c. 15,633. august 1962 .

.W

a simple method for calculating the span and chordwise loading on straight and swept wings of any aspect ratio at subsonic speeds .
the methods of the classical aerofoil theory are used to derive a

general theory for wings of any given planform . the load over the whole surface of a given wing can be calculated at a given subcritical mach number, and the procedure is as simple and rapid as that of the classical aerofoil theory . the calculated results are confirmed by experiments .

.I 677

.T

methods for calculating the lift distribution of wings /subsonic lifting surface theory/ .

.A

.B

r + m 2884, r.a.e. rep. aero. 2353. a.r.c. 13,439. january 1950 .

.W

methods for calculating the lift distribution of wings /subsonic lifting surface theory/ .

this report contains some fairly simple and economic methods for calculating the load distribution on wings of any plan form based on the conceptions of lifting surface theory . the computer work required is only a small fraction of that of existing methods with comparable accuracy . this is achieved by a very careful choice of the positions of pivotal points, by plotting once for all those parts of the downwash integral which occur frequently and by a consequent application of approximate integration methods similar to those devised by the author for lifting line problems .

the basis of the method is to calculate the local lift and pitching moment at a number of chordwise sections from a set of linear equations satisfying the downwash conditions at two pivotal points in each section . interpolation functions of trigonometrical form are used for spanwise integration both in setting up the downwash equations and in getting the resultant forces on the wing from the local forces . the preliminary chordwise integrations for the downwash are predigested in a series of charts/figs.1-6/,.it is these which make the method a practical computing proposition .

the theory is outlined in sections 2-5,.section 6 deals with the solution of the linear equation and section 7 with the resultant forces on the wing . some examples are worked out in section 8 to compare with other methods,. one solution is given in full detail in tables 8-30 as a guide for computers . appendices i-vi discuss more carefully some salient points of the mathematical theory, and appendix vii is intended to instruct the computer how to carry out the steps of the calculation .

.I 678

.T

the effect of end plates on swept wings .

.A

kuchemann,d. and kettle,d.j.

.B

rae r.aero.2429, 1952.

.W

the effect of end plates on swept wings .

existing methods of calculating the effect of endplates on straight wings are modified so as to apply to swept wings . the changes in overall lift and drag, and also the spanwise distribution of the additional load,

can be calculated .

the theoretical results are

compared with experimental results

obtained on swept wings, including

new measurements of lift, drag and

pitching moment, made on an untapered

the method of calculation is also

extended to cover the effect of

the tip vortex which is formed on wings

without endplates .

.I 679

.T

low speed tests on 45 sweptback wings .

.A

weber,j., brebner,g.g. and kuchemann,d.

.B

rae r. aero.2374, 1958.

.W

low speed tests on 45 sweptback wings .

this report contains the results

of pressure measurements on three

and aspect ratio 5, over an

incidence range up to 10 . chordwise

and spanwise lift distributions

are given, mostly near the centre

where, on two of the wings,

modifications had been made to the section

shape . it was found that altering

the thickness distribution in the

centre did not affect the loading but

that approximately straight isobars

could be obtained at values of

below about 0.1 . by the incorporation

of twist and camber in the central

part the distortion of the lift

distribution in the centre could be

avoided at one particular incidence,

and thus the same chordwise

distribution obtained over most of the span .

twist and camber alone do not improve

the isobar pattern and

therefore a thickness modification would be

needed to give the desired

lift distribution and isobar pattern at one

particular incidence .

the results of experimental investigations

of the boundary layer

and of the effect of aspect ratio will be given

in a later report .

.I 680

.T

generalized conical flow fields in supersonic wing theory .

.A

lomax, h. and heaslett, m.a.

.B

naca tn 2497, september 1951 .

.W

generalized conical flow fields in supersonic wing theory .

linearized, compressible-flow analysis is applied to the study of

quasi-conical supersonic wing theory . single-integral equations are

derived which relate either the loading to the shape of a lifting

surface or the thickness of a symmetrical wing to the pressure

distribution for triangular wings with subsonic leading edges . the forms of

these equations and their inversions are simplified through the

introduction of the finite part and the generalized principal part of an

integral .

applications of the theory, in the lifting case, include previously

known results . in the nonlifting case, it is shown that for a specified

pressure distribution the theory does not always predict a unique

thickness distribution . this is demonstrated for a triangular plan form

having a constant pressure gradient in the stream direction .

.I 681

.T

integrals and integral equations in linearized wing theory .

.A

lomax, h. haslet, m.a. and fuller, f.b.

.B

naca rep. 1054, 1951 .

.W

integrals and integral equations in linearized wing theory .

the formulas of subsonic and supersonic wing theory for source, doublet, and vortex distributions are reviewed, and a systematic presentation is provided which relates these distributions to the pressure and to the vertical induced velocity in the plane of the wing . it is shown that care must be used in treating the singularities involved in the analysis and that the order of integration is not always reversible . concepts suggested by the irreversibility of order of integration are shown to be useful in the inversion of singular integral equations when operational techniques are used . a number of examples are given to illustrate the methods presented, attention being directed to supersonic flight speeds .

.I 682

.T

the lift of twisted and cambered wings in supersonic flow .

.A

lance, g.n.

.B

aero. quart. v. 6/2/, may, 1955 .

.W

the lift of twisted and cambered wings in supersonic flow .

a generalised conical flow theory is used to deduce an integral equation relating the velocity potential on a delta wing/with subsonic leading edges/to the given downwash distribution over the wing . the complete solution of this integral equation is derived . this complete solution is composed of two parts, one being symmetric and the other antisymmetric with respect to the spanwise co-ordinate,. each part represents a velocity potential . for example, if y is the spanwise co-ordinate and x is measured in the free stream direction, then a

downwash of the form w-a ux/y/is symmetric and will give rise to a

symmetric potential, whereas w-a ux/y/sgn y is anti-symmetric and gives

rise to an anti-symmetric potential . the velocity potentials of such

flows are given in the form of tables for all downwashes up to and

including homogenous cubics in the spanwise and streamwise

co-ordinates . table iii gives similar formulae in the limiting case

were used over a cycle of the tumbling motion . the analytical

expression was in good agreement with numerical solutions of the complete

non-linear equations of motion .

.I 683

.T

the use of conical camber to produce flow attachment at the leading

edge of a delta wing and to minimize the lift-dependent drag at sonic

and supersonic speeds .

.A

smith,j.h.b. and mangler, k.w.

.B

r.a.e. rep. aero. 2584. arc 19,961, september 1957 .

.W

the use of conical camber to produce flow attachment at the leading

edge of a delta wing and to minimize the lift-dependent drag at sonic

and supersonic speeds .

in an attempt to avoid flow separation at the leading edge of a thin

delta wing with subsonic leading edges, an attachment line is prescribed

there . this is done by requiring the load, as predicted by attached

flow theory, to vanish along the leading edge at the design lift

coefficient . for sonic speed, a complete account of this flow is given
in terms of slender wing theory and the load distributions corresponding
to arbitrary conical camber are calculated . for supersonic speeds
load distributions arising in the slender wing theory are considered and
the corresponding conical camber distributions are found by linearized
theory . the lift-dependent drag for a given lift is then minimized with
respect to the coefficients of a linear combination of these load
distributions . it is found that the lift-dependent drag factor for these
conically cambered wings approaches the value it takes for the attached
flow/in which leading edge suction occurs/past the uncambered wing at
the same mach number, as more terms are included in the linear
combination . however, when the leading edge is almost sonic an appreciable
reduction is predicted . the corresponding load distributions and wing
shapes are calculated and drawn . the optimum shapes for a fixed number
of terms resemble flat plates drooped downwards near their edges, so
that the localised leading edge suction is replaced by a distributed
force on a forward-facing surface, producing an effect of similar
magnitude .

.I 684

.T

tables of complete elliptic integrals .

.A

heuman,c.

.B

j. maths. and phys. v. 20, 1941, pp 127-206 .

.W

tables of complete elliptic integrals .

the present paper contains a set of tables of complete elliptic

integrals computed and collected especially for applications to certain

dynamical problems .

the tabulated functions are four in number and are denoted by

f/a/, g/a/, e/a/, and/a,b/respectively . the definitions of these

functions and their connections with the functions of legendre will be

discussed in the following .

.I 685

.T

aerodynamic effects of some configuration variables

on the aeroelastic characteristics of lifting surfaces

at mach numbers from 0. 7 to 6. 86 .

.A

hanson,p.w.

.B

nasa tn.d984, 1961.

.W

aerodynamic effects of some configuration variables

on the aeroelastic characteristics of lifting surfaces

at mach numbers from 0. 7 to 6. 86 .

results of flutter tests on

some simple all-movable-control-type

models are given . one set of models,

which had a square planform with

double-wedge airfoils with four

different values of leading- and

trailing-edge radii from 0 to 6 percent chord and airfoil thicknesses of 9, 11, at mach numbers from 0.7 to 6.86 . the bending-to-torsion frequency ratio was about 0.33 . the other set of models, which had a tapered planform with single-wedge and double-wedge airfoils with thicknesses of 3, 6, 9, and 12 percent chord, was tested at mach numbers from 0.7 to 3.98 and a frequency ratio of about 0.42 .

the tests indicate that, in general, increasing thickness has a destabilizing effect at the higher mach numbers but is stabilizing at subsonic and transonic mach numbers . double-wedge airfoils are more prone to flutter than single-wedge airfoils at comparable stiffness levels . increasing airfoil bluntness has a stabilizing effect on the flutter boundary at supersonic speeds but has a negligible effect at subsonic speeds . however, increasing bluntness may also lead to divergence at supersonic speeds .

results of calculations using second-order piston-theory aerodynamics in conjunction with a coupled-mode

analysis and an uncoupled-mode analysis

are compared with the experimental

results for the sharp-edge airfoils at

supersonic speeds . the uncoupled-mode

analysis more accurately predicted

the flutter characteristics of the

tapered-planform models, whereas the

coupled-mode analysis was somewhat

better for the square-planform models .

for both the uncoupled- and coupled-mode

analyses, agreement with the

experimental results improved with

increasing mach number . in general,

both methods of analysis gave unconservative

results with respect to the

experimental flutter boundaries .

.I 686

.T

flutter tests of some simple models at a mach number

of 7. 2 in helium flow .

.A

morgan,h.g. and miller,r.w.

.B

nasa memo 4-8-59l, 1959.

.W

flutter tests of some simple models at a mach number of 7. 2 in helium flow .

results of hypersonic flutter tests on some simple models are presented . the models had rectangular plan forms of panel aspect ratio 1.0, no sweepback, and bending-to-torsion frequency ratios of about . two airfoil sections were included in the tests ,. double wedges of 5-, 10-, and 15-percent thickness and flat plates with straight, parallel sides and beveled leading and trailing edges . the models were supported by a cantilevered shaft .

the double-wedge wings were tested in helium at a mach number of 7.2 . an effect of airfoil thickness on flutter speed was found, thicker wings requiring more stiffness to avoid flutter . a few tests in air at a mach number of 6.9 showed the same thickness effect and also indicated that tests in helium would predict conservative flutter boundaries in air . the data in air and helium seemed to be correlated by piston-theory calculations . piston-theory calculations agreed well with experiment for

the thinner models but began to deviate

as the thickness parameter

approached and exceeded 1.0 .

a few tests on flat-plate models

with various elastic-axis locations

were made . piston-theory calculations

would not satisfactorily predict

the flutter of these models, probably

because of their blunt leading

edges .

.I 687

.T

oscillating airfoils at high mach number .

.A

lighthill, m.j.

.B

j. aero. sc. v. 20. june 1953. pp 402-406 .

.W

oscillating airfoils at high mach number .

a simple formula is given for the pressure distribution on an

oscillating airfoil in two-dimensional flow at high mach number . the

formula is expected to be reasonably accurate if the pressure on the

surface remains within the range 0.2 to 3.5 times the mainstream

pressure . to illustrate the application of the formula, some results

for symmetrical airfoils performing pitching oscillations are obtained

and compared with results obtained from existing theories in the case of high mach number .

.I 688

.T

tables of aerodynamic coefficients obtained from developed newtonian expressions for complete and partial conic and spheric bodies at combined angles of attack and sideslip with some comparisons with hypersonic experimental data .

.A

wells, w.r. and armstrong, w.o.

.B

nasa tr r -dash 127, 1962 .

.W

tables of aerodynamic coefficients obtained from developed newtonian expressions for complete and partial conic and spheric bodies at combined angles of attack and sideslip with some comparisons with hypersonic experimental data .

closed-form expressions and tables composed from these expressions are presented for complete and partial conic and spheric bodies at combined angles of attack and sideslip in newtonian flow . aerodynamic coefficients of these bodies are tabulated for various body segments over a range of angles of attack from 1degree to 85degree and angles of sideslip from 0degree to 15degree .

some comparisons between newtonian predictions and hypersonic experimental aerodynamic characteristics were made for conic bodies having various surface slopes, nose bluntnesses, and body cross sections to indicate the range of validity of the theory . in general, the theory is shown to agree quite well with experimental results for sharp-nose complete cones and for configurations having large blunted noses and steep surface slopes . however, agreement between theory and experiment generally is poor for the more slender, slightly blunted complete or half conic bodies and also for sharp-nose half conic bodies where real-flow phenomena such as forebody interference, viscous forces, leeward surface contributions, or leading-edge pressure reductions may have significant effect . the agreement between theory and experiment for the bodies considered can be improved by using the stagnation pressure coefficient behind a normal shock rather than 2 as the newtonian coefficient, although for the sharp-nose half conic bodies there is no theoretical justification for this modification .

.I 689

.T

investigation of the laminar aerodynamics heat transfer characteristics of a hemisphere cylinder in the langley 11-inch hypersonic tunnel at a mach number of 6. 8.

.A

crawford,d.h. and mccauley,w.d.

.B

naca r.1323, 1957.

.W

investigation of the laminar aerodynamics heat transfer characteristics of a hemisphere cylinder in the langley 11-inch hypersonic tunnel at a mach number of 6. 8.

a program to investigate the aerodynamic heat transfer of a nonisothermal hemisphere-cylinder has been conducted in the langley 11-inch hypersonic tunnel at a mach number of 6.8

and a reynolds number from approximately 0.14x10 to

experimental heat-transfer coefficients were slightly less over the

whole body than those predicted by the theory of stine and

wanlass (naca technical note 3344) for an isothermal

surface . for stations within 45 of the stagnation point the

heat-transfer coefficients could be correlated by a single relation

between local stanton number and local reynolds number .

pitot pressure profiles taken at a mach number of 6.8 on a

hemisphere-cylinder have verified that the local mach number or

velocity outside the boundary layer required in the theories may

be computed from the surface pressures by using isentropic flow

relations and conditions immediately behind a normal shock .

the experimental pressure distribution at a mach number of

velocity gradients calculated at the stagnation point by using

the modified newtonian theory vary with mach number and

are in good agreement with those obtained from measured

pressures for mach numbers from 1.2 to 6.8 .

at the stagnation point the theory of sibulkin, in which the

diameter and conditions behind the normal shock were used,

was in good agreement with the experiment when the velocity

gradient at the stagnation point appropriate to the free-stream

mach number was used .

.I 690

.T

investigaion of the flow over a spiked-nose hemisphere

cylinder at a mach number of 6. 8.

.A

crawford,d.h.

.B

nasa tn.d118, 1959.

.W

investigaion of the flow over a spiked-nose hemisphere

cylinder at a mach number of 6. 8.

the shape and nature of the

flow over a spiked-nose

hemisphere-cylinder was studied in detail

at a nominal mach number of 6.8 and in a

reynolds number range (based on

diameter and stream conditions ahead of

the model) of 0.12 x 10 to 1.5 x 10 .

schlieren photographs showed

the effect of varying the spike length

and reynolds number upon the shape

of the separated boundary and upon the

location of transition . the heat

transfer and pressure distribution over

the body were then correlated

with the location of the start of

separation, the location of

reattachment, and the location of the start of

transition .

.I 691

.T

calculation procedure for thermodynamic transport, and flow properties of the combustion products of a hydrocarbon fuel mixture burned in air

with results for ethylene-air and methane-air mixtures .

.A

.B

nasa tn d-914, 1962 .

.W

calculation procedure for thermodynamic transport, and flow properties

of the combustion products of a hydrocarbon fuel mixture burned in air

with results for ethylene-air and methane-air mixtures .

a procedure is presented whereby the composition, thermodynamic

properties, and transport properties of the dissociated combustion

products of a fuel consisting of a mixed hydrocarbon compound burned in

air may be calculated . equations and procedures for determining

supersonic nozzle ordinates and flow properties for the dissociated

combustion products are presented in an appendix . results are presented

for the respective hydrocarbon fuels, methane and ethylene, at the

equivalence ratios of 1.0, 0.9, 0.8, and 0.7 for pressures varying

between 10 and 8 x 10 atmospheres and temperatures from 200degree k to

.I 692

.T

investigation of the jet effects on a flat surface downstream of the

exit of a simulated turbojet nacelle at a free-stream mach number of

2.02 .

.A

bressette, w.e.

.B

naca rm l54e05a, 1954 .

.W

investigation of the jet effects on a flat surface downstream of the exit of a simulated turbojet nacelle at a free-stream mach number of 2.02 .

an investigation at a free-stream mach number of 2.02 was made to determine the effects of a propulsive jet on a wing surface located in the vicinity of a choked convergent nozzle . static-pressure surveys were made on a flat surface that was located in the vicinity of the propulsive jet . the nozzle was operated over a range of exit pressure ratios at different fixed vertical distances from the flat surface .

within the scope of this investigation, it was found that shock waves, formed in the external flow because of the presence of the propulsive jet, impinged on the flat surface and greatly altered the pressure distribution . an integration of this pressure distribution, with the location of the propulsive jet exit varied from 1.450 propulsive-jet exit diameters to 3.392 propulsive-jet exit diameters below the wing, resulted in an incremental lift for all jet locations that was equal to the gross thrust at an exit pressure ratio of 2.86 .

this incremental lift increased with increase in exit pressure ratio, but not so rapidly as the thrust increased, and was approximately constant at any given exit pressure ratio .

.I 693

.T

investigation of jet effects on a flat surface downstream of the exit of a simulated turbojet nacelle at a free-stream mach number of 1.39 .

.A

bressette, w.e. and leiss, a.

.B

.W

investigation of jet effects on a flat surface downstream of the exit of a simulated turbojet nacelle at a free-stream mach number of 1.39 .

an investigation at a free-stream mach number of 1.39 utilizing a blowdown-type tunnel was made to determine the effects of a propulsive jet on a zero angle-of-attack wing surface located in the vicinity of both a choked convergent nozzle and a convergent-divergent nozzle . staticpressure surveys were made on a flat surface that was located in the vicinity of the propulsive jet . the nozzles were operated over a varied range of both exit static- and total-pressure ratios at different within the scope of this investigation, it was found that shock waves, formed in the external flow because of the presence of the jet exhaust, impinged on the flat surface and greatly altered the pressure distribution . an integration of this pressure distribution for the choked convergent nozzle, with the location of the propulsive-jet exit varied from 1.747 jet-exit diameters to 4.981 jet-exit diameters below the wing surface, resulted in a positive incremental normal force on the wing at all positions .

.I 694

.T

pressure distribution induced on a flat plate by a supersonic and sonic jet exhaust at a free-stream mach number of 1.80 .

.A

leiss, a. and bressette, w.e.

.B

naca rm l56106, 1957 .

.W

pressure distribution induced on a flat plate by a supersonic and sonic

jet exhaust at a free-stream mach number of 1.80 .

as a continuation of previous research at mach numbers of 2.02 and 1.39,

an experimental investigation was made of the pressures induced on a

flat plate by a propulsive jet exhausting from sonic and supersonic

nozzles at a free-stream mach number of 1.80 . measurements of the

pressure distribution on a flat-plate wing were made at zero angle of

attack for four different locations of the jet exhaust nozzle beneath

the wing . both a choked convergent nozzle and a convergent-divergent

nozzle on the nacelle were used . the nozzles were operated at

nacelle-exit total-pressure ratios from 2 to 16 and the reynolds number per foot

was approximately 13 x 10 .

two distinct shock waves impinged on the wing surface and greatly

altered the pressure distribution at all nozzle positions . positive

incremental normal force resulted on the wing at all positions .

comparisons are presented for two free-stream mach numbers .

.I 695

.T

some experiments relating to the problem of simulation of hot jet

engines in studies of jet effects on adjacent surfaces at a free-stream

mach number of 1.80 .

.A

bressette, w.e.

.B

naca rm l56e07. 1956 .

.W

some experiments relating to the problem of simulation of hot jet engines in studies of jet effects on adjacent surfaces at a free-stream mach number of 1.80 .
an investigation at a free-stream mach number of 1.80 in a blowdown type tunnel was made to study the effect on the pressure distribution of a zero angle of attack wing surface when certain exhaust parameters of a hot turbojet engine are varied . static-pressure surveys were made on a wing surface that was located in the vicinity of a small-scale propulsive jet . this propulsive jet was operated with four types of jet exhausts . these jet exhausts were a hot jet /hydrogen burned in air/, a cold air jet, a cold helium jet, and a jet composed of a mixture of two cold gases /hydrogen and carbon dioxide/ . the hot jet, because of its high exhaust temperature /3,300degreer/ and because combustion was performed in air, was believed reasonably able to simulate the exhaust parameters of an actual afterburning turbojet engine . the cold jets used were selected in order that the effects of a variation in the exhaust parameters of jet-exit static-pressure ratio, ratio of specific heats, density, and velocity, could be obtained by comparing each cold jet with the hot jet or with another cold jet . the tests were made over a range of jet-exit staticpressure ratios from 1 to 9 with values of the ratio of specific heats of 1.27, 1.40, and 1.66 and at variations in density and velocity of the order of approximately 8 and 3 times, respectively .
within the scope of this investigation, it was found that jet-exit static-pressure ratio and the ratio of specific heats affected the pressure distribution on the wing associated with jet interference while a variation in exit velocity and density did not . the jet-exit

staticpressure ratio affected the wing pressure distribution in a major

way while the ratio of specific heats had only a minor effect . the

addition of temperature in the propulsive jet exhaust at a jet-exit

staticpressure ratio of 4 had little or no effect on the pressure

distribution associated with jet interference on the wing .

.I 696

.T

pressure loads produced on a flat-plate wing by rocket jets exhausting

in a spanwise direction below the wing and perpendicular to a

free-stream flow of mach number 2.0 .

.A

falangan, r.a. and janos, j. j.

.B

nasa tn d-893, 1961 .

.W

pressure loads produced on a flat-plate wing by rocket jets exhausting

in a spanwise direction below the wing and perpendicular to a

free-stream flow of mach number 2.0 .

an investigation at a reynolds number per foot of 14.4 x 10 was made to

determine the pressure loads produced on a flat-plate wing by rocket

jets exhausting in a spanwise direction beneath the wing and

perpendicular to a free-stream flow of mach number 2.0 . the ranges of the

variables involved were /1/ nozzle types - one sonic /jet mach number of

two-dimensional supersonic /jet mach number of 1.71/,. /2/ vertical

nozzle positions beneath the wing of 4, 8, and 12 nozzle-throat

diameters,. and /3/ ratios of rocket-chamber total pressure to

free-stream static pressure from 0 to 130 .

the incremental normal force due to jet interference on the wing varied

from one to two times the rocket thrust and generally decreased as the

pressure ratio increased . the chordwise coordinate of the

incremental-normal-force center of pressure remained upstream of the nozzle center

line for the nozzle positions and pressure ratios of the investigation .

the chordwise coordinate approached zero as the jet vertical distance

beneath the wing increased . in the spanwise direction there was little

change due to varying rocket-jet position and pressure ratio . some

boundary-layer flow separation on the wing was observed for the rocket

jets close to the wing and at the higher pressure ratios . the magnitude

of the chordwise and spanwise pressure distributions due to jet

interference was greatest for rocket jets close to the wing and decreased as

the jet was displaced farther from the wing .

the design procedure for the rockets used is given in the appendix .

.I 697

.T

effects on adjacent surfaces from the firing of rocket jets .

.A

bressette, w.e. and leiss, a.

.B

naca rm l 57d19a, 1957 .

.W

effects on adjacent surfaces from the firing of rocket jets .

this paper is a preliminary and brief account of some research currently

being conducted to determine the jet effects on adjacent surfaces from

the firing of rocket jets . measurements of jet-effect pressures on a

flat plate as well as shadowgraphs are presented that were obtained when a rocket jet at a mach number of 3 was exhausted downstream and upstream into free-stream flow at a mach number of 2 located from 2 to 4.7 rocket-jet-exit diameters from the plate . the jet effects on the flat plate with the rocket jet exhausting downstream are of the same order of magnitude as those previously obtained from sonic exits with a total pressure 10 times lower . a maximum pressure coefficient on the plate of rocket-jet-exit diameters below the plate, and an integration of the measured jet-effect pressures at this position resulted in a normal force on the plate equal to 2.3 times the thrust output of the rocket jet .

.I 698

.T

the unsteady lift of a wing of finite aspect ratio .

.A

jones, r.t.

.B

1940, naca rep. 681 .

.W

the unsteady lift of a wing of finite aspect ratio .

unsteady-lift functions for wings of finite aspect ratio have been calculated by correcting the aerodynamic inertia and the angle of attack of the infinite wing . the calculations are based on the operational method .

the starting lift of the finite wing is found to be only slightly less than that of the infinite wing,. whereas the final lift may be considerably less . the theory indicates that the initial distribution of lift is similar to the final distribution .

curves showing the variation of lift after a sudden unit change in angle of attack, during penetration of a sharpedge gust, and during a continuous oscillation are given . operational equivalents of these functions have been devised to facilitate the calculation of lift under various conditions of motion . as an application of these formulas, the vertical acceleration of a loaded wing caused by penetrating a gust has been calculated .

.I 699

.T

approximate indical lift functions for several wings of finite span in incompressible flow as obtained from oscillatory lift coefficients .

.A

drischler j.a.

.B

naca tn 3639, 1956 .

.W

approximate indical lift functions for several wings of finite span in incompressible flow as obtained from oscillatory lift coefficients .

the unsteady-lift functions for a wing undergoing a sudden change in sinking speed have been presented for delta wings having aspect ratios of 0, 2, and 4 and for rectangular and elliptical wings having aspect ratios of 0, 3, and 6 . for the elliptical and rectangular wings the spanwise lift distributions were also presented . these functions were calculated from the lift coefficients associated with a wing oscillating harmonically in pure translational motion, as obtained from several sources .

the results of these calculations indicate that the normalized unsteady-lift functions are substantially independent of the shape of the plan form for elliptical, rectangular, or moderately tapered wings,. however, for delta wings the increase of lift toward the steady-state value is much more rapid than that for the aforementioned wings of the same aspect ratio . these results also corroborate the results of other investigations in that the rate of growth of lift tends to increase with a decrease in aspect ratio . the shape of the spanwise distributions of the indicial lift seems to be, for all practical purposes, independent of time for rectangular and elliptical wings .

.I 700

.T

two and three-dimensional unsteady lift problems in high speed flight .

.A

lomax et al.

.B

naca rep. 1077, 1952 .

.W

two and three-dimensional unsteady lift problems in high speed flight . the problem of transient lift on two- and three-dimensional wings flying at high speeds is discussed as a boundary-value problem for the classical wave equation . kirchhoffs formula is applied so that the analysis is reduced, just as in the steady state, to an investigation of sources and doublets . the applications include the evaluation of indicial lift and pitchingmoment curves for two-dimensional sinking and pitching wings flying at mach numbers equal to 0, 0.8, 1.0, 1.2, and triangular wings in both forward and reversed flow are presented and compared with the two-dimensional values .

.I 701

.T

numerical determination of indical lift of a two-dimensional sinking

airfoil at subsonic mach numbers from oscillatory lift coefficients with

calculations for mach number 0.7 .

.A

mazelsky, b.

.B

naca tn 2562, 1951 .

.W

numerical determination of indical lift of a two-dimensional sinking

airfoil at subsonic mach numbers from oscillatory lift coefficients with

calculations for mach number 0.7 .

the reciprocal equations for relating the incompressible circulatory

indicial lift to the lift due to harmonic oscillations have been

modified to include the noncirculatory lift associated with

apparent-mass effects . although the apparent-mass effects are impulsive in

nature in incompressible flow, the lift due to apparent-mass effects in

compressible flow is a time-dependent function . the corresponding

reciprocal equations for the total compressible lift are given . by use

of the reciprocal equations for compressible flow, the indicial lift and

moment functions due to an airfoil's experiencing a sudden acquisition

of vertical velocity are determined numerically for mach number 0.7 .

lack of sufficient flutter coefficients prevents the calculation of

these functions at other mach numbers .

although the indicial lift and moment functions due to penetration of a

sharp-edge gust may be obtained from the oscillatory tab or aileron coefficients by a similar analysis, sufficient coefficients are not

available at the present . however, an approximate method is shown for

determining a portion of this unsteady-lift function .

when a comparison is made of the indicial lift functions at mach numbers

appears to be less rapid for the compressible case than for the

incompressible case . consequently, the calculation of the gust load factor

at high subsonic mach numbers utilizing the two-dimensional

incompressible indicial lift functions and an over-all correction for

compressibility such as the prandtl-glauert factor might be conservative .

.I 702

.T

numerical determination of indical lift and moment functions for a two

dimensional sinking and pitching airfoil at mach numbers 0.5 and 0.6 .

.A

mazelsky, b. and drischler, j.a.

.B

naca tn 2739, 1952 .

.W

numerical determination of indical lift and moment functions for a two

dimensional sinking and pitching airfoil at mach numbers 0.5 and 0.6 .

the indicial lift and moment functions are determined approximately for

sinking and pitching motion at mach numbers m of 0.5 and 0.6 . these

functions are determined from a knowledge of the existing oscillatory

coefficients at the low reduced frequencies and from approximate

expressions of these coefficients at the high reduced frequencies .

the beginning portion of the indicial lift function associated with an

airfoil penetrating a sharp-edge gust in subsonic flow is evaluated by

use of an exact method . by use of an approximate method for determining
the remaining portion, the complete indicial gust function is
determined for m 0.5, m 0.6, and m 0.7 .
all the indicial lift and moment functions are approximated by an
exponential series,. the coefficients which appear in the exponential
approximations for each indicial function are tabulated for m 0.5,
m 0.6, and m 0.7 .
.I 703
.T
general airfoil theory .
.A
kussner, h.g.
.B
naca tm 979, 1941 .
.W
general airfoil theory .
on the assumption of infinitely small disturbances the author develops
a generalized integral equation of airfoil theory which is applicable
to any motion and compressible fluid . successive specializations yield
various simpler integral equations, such as possio's, birnbaum's, and
prandtl's integral equations, as well as new ones for the wing of
infinite span with periodic downwash distribution and for the oscillating
wing with high aspect ratio . lastly, several solutions and methods for
solving these integral equations are given .
.I 704
.T
a systematic kernel function procedure for determining aerodynamic

forces on oscillating or steady finite wings at subsonic speeds .

.A

watkins, c.e., woolston, d.s. and cunningham, h.j.a.

.B

nasa tr r-48, 1959 .

.W

a systematic kernel function procedure for determining aerodynamic

forces on oscillating or steady finite wings at subsonic speeds .

a detailed description is given of a method of approximating solutions

to the integral equation that relates oscillatory or steady lift and

downwash distributions on finite wings in subsonic flow . the method of

solution is applicable to general plan forms with either curved or

straight leading and trailing edges . moreover, it is directly

applicable to control surfaces such as all-movable tails but modifications

are needed to apply it to controls in general . applications of the

method involve evaluations of numerous integrals that must be handled by

numerical procedures but systematic schemes of evaluations have been

adopted that are well suited to the routines of automatic digital

computing machines . these schemes of evaluation have been incorporated in

a program for an ibm 704 electronic data processing machine . with this

machine, a pressure distribution together with such quantities as

section or total lift and moment coefficients or generalized forces can be

determined for a given value of frequency and mach number and for

several /four or five/ modes of oscillation in about 4 minutes of

machine time . in the case of steady downwash conditions corresponding

quantities can be obtained in about 2 minutes of machine time .

in order to illustrate applications of the method, results of several calculations are presented . in these illustrations total forces and

moments are compared /1/ with results of analytic procedures for a

circular plan form with steady downwash conditions, /2/ with results of

other theories and with experiment for a rectangular plan form of aspect

ratio 1 at a uniform angle of attack, and /3/ with some experimental

results for a rectangular plan form of aspect ratio 2 undergoing

pitching and flapping oscillations . also included in the illustrations are

results of flutter calculations compared with experimental results for

an allmovable control surface of aspect ratio 3.50 and for a

cantilevered rectangular plan form of aspect ratio 5.04 .

.I 705

.T

on the kernel function of the integral equation relating the lift and

downwash distributions of oscillating finite wings in subsonic flow .

.A

watkins, c.e. and runyal, h.l. and woolston, d.s.

.B

naca rep. 1234, 1955 .

.W

on the kernel function of the integral equation relating the lift and

downwash distributions of oscillating finite wings in subsonic flow .

this report treats the kernel function of an integral equation that

relates a known or prescribed downwash distribution to an unknown lift

distribution for a harmonically oscillating finite wing in compressible

subsonic flow . the kernel function is reduced to a form that can be

accurately evaluated by separating the kernel function into two parts ..

a part in which the singularities are isolated and analytically expressed and a nonsingular part which may be tabulated . the form of the kernel function for the sonic case /mach number of 1/ is treated separately . in addition, results for the special cases of mach number of o /incompressible case/ and frequency of o /steady case/ are given . the derivation of the integral equation which involves this kernel function, originally performed elsewhere /see, for example, naca technical memorandum 979/, is reproduced as an appendix . another appendix gives the reduction of the form of the kernel function obtained herein for the three-dimensional case to a known result of possio for two-dimensional flow . a third appendix contains some remarks on the evaluation of the kernel function, and a fourth appendix presents an alternate form of expression for the kernel function .

.I 706

.T

on som reciprocal relations in the theory of nonstationary flows .

.A

garrick, i.e.

.B

naca rep. 629. 1938 .

.W

on som reciprocal relations in the theory of nonstationary flows .

in the theory of nonstationary flows about airfoils, the /indicial lift/ function k /s/ of wagner and the /alternating lift/ function c /k/ of theodorsen have fundamental significance . this paper reports on some interesting relations of the nature of fourier transforms that exist between these functions . general problems in transient flows about airfoils may be given a unified broad treatment when these functions are

employed . certain approximate results also are reported which are of
notable simplicity, and an analogy with transient electrical flows is
drawn .
.I 707
.T
thermal analysis of stagnation regions with emphasis on heat-sustaining
nose shapes at hypersonic speeds .
.A
hanawalt, a.j., blessing, a.h. and schmidt, c.m.
.B
j.aero. sc. may 1959. p. 257-263 .
.W
thermal analysis of stagnation regions with emphasis on heat-sustaining
nose shapes at hypersonic speeds .
the leading edges and noses of hypersonic vehicles are subjected to
severe aerodynamic heating and must be cooled in some manner-dash e.g.,
internal convection, transpiration, or radiation . it is this latter
mode of handling the problem that is discussed in this paper .
neglecting conduction in the leading-edge region, the maximum temperature for
long-range hypersonic gliders is of the same order as the melting point
of refractory materials, with a corresponding large temperature gradient
away from the leading edge . inclusion of conduction in the aft
direction reduces the maximum temperature and distributes the heat to a
location that will radiate it out from the surface . for either
steady-state or transient conditions, the temperature at the leading edge is
reduced by conduction, while the temperature aft of the leading-edge
shoulder is increased, thus setting up a heat transmission balance

between the convective influx of heat, the redistribution of heat by

conduction, and the radiation of heat from the surface . the feasibility of

such a mechanism can be enhanced by suitably choosing leading-edge

shapes and materials . the philosophy behind the choice of leading-edge

shapes is discussed and the effects of varying parameters, such as

shape, diameter, emissivity, conductivity, thickness, etc., are shown .

.I 708

.T

aerodynamic characteristics of two winged reentry vehicles at supersonic

and hypersonic speeds .

.A

ladson, c.l. and johnston, p.j.

.B

nasa tm x 346, 1961 .

.W

aerodynamic characteristics of two winged reentry vehicles at supersonic

and hypersonic speeds .

tests were conducted at the langley research center on two winged

lifting hypersonic reentry glider configurations . performance, stability,

and control data are presented at mach numbers of 1.62 and 2.91 for

angles of attack up to 15degree and at mach numbers of 6.8 and 9.6 for

angles of attack up to 25degree .

.I 709

.T

static longitudinal aerodynamic characteristics at transonic speeds and

angles of attack up to 99degree of a reentry glider having folding

wingtip panels .

.A

olstad, w.b.

.B

nasa tm x-610, 1961 .

.W

static longitudinal aerodynamic characteristics at transonic speeds and angles of attack up to 99degree of a reentry glider having folding wingtip panels .

data are presented which were obtained from a transonic wind-tunnel investigation of a reentry glider having folding wing-tip panels . the tests were conducted at angles of attack from -4degrees to 99degrees . the reynolds number based on the mean geometric chord of the fixed planform varied from 2.35 x 10 to 2.99 x 10 .

the maximum lift-drag ratio for the model with the folding wing-tip panels fully extended decreased from a maximum value of 7.8 at a mach number of 0.60 to about 3.4 at mach numbers from 1.03 to 1.20 . the model with the folding wing panels fully extended was stable for values of the lift coefficient from 0 up to at least 0.8 . above this lift coefficient pitch-up tendencies were observed, followed by an unstable or neutrally stable region which extended up to values of angle of attack of 50degrees or 60degrees . deflecting the folding wing panels between ducing a significant change in the trim angle of attack or in any of the force or moment coefficients in the angle-of-attack range from 49degree to 99degree .

.I 710

.T

the smallest height of roughness capable of affecting boundary-layer transition .

.A

smith, a.m.o. and clutter, d.w.

.B

j. aero. sc. april, 1959. p.229-245, 256 .

.W

the smallest height of roughness capable of affecting boundary-layer transition .

an investigation was made to determine the smallest size of isolated roughness that will affect transition in a laminar-boundary layer . critical heights for three types of roughness were found in a low-speed wind tunnel . the types were /1/ two-dimensional spanwise wires, /2/ three-dimensional discs, and /3/ a sandpaper type . in addition to type of roughness, test variables included the location of roughness, pressure distribution, degree of tunnel turbulence, and length of natural laminar flow .

the most satisfactory correlation parameter was found to be the roughness reynolds number, based on the height of roughness and flow properties at this height . the value of this critical reynolds number was found to be substantially independent of all test variables except the shape of roughness . this parameter also correlates well other published data on critical roughness in low-speed flow . the value of the roughness reynolds number necessary to move transition forward to the roughness itself was also determined for the three types of roughness and was found to be approximately constant for a given type of roughness .

an investigation of the limited amount of available data on critical roughness in supersonic flow indicates that the effects of roughness may still be correlated by the roughness reynolds number . the value of

this reynolds number depends primarily on the mach number at the top of

the roughness . when this mach number is greater than 1.0, the roughness

reynolds number based on conditions behind a shock is probably the

characteristic parameter .

.I 711

.T

an investigation at subsonic speeds of aerodynamic characteristics at

angles of attack from -dash 4degrees to 100degrees of a delta-wing

reentry configuration having folding wingtip panels .

.A

spencer, b.

.B

nasa tm x-288, 1960 .

.W

an investigation at subsonic speeds of aerodynamic characteristics at

angles of attack from -dash 4degrees to 100degrees of a delta-wing

reentry configuration having folding wingtip panels .

an investigation was made at subsonic speeds in the langley highspeed

lifting reentry configuration having folding wingtip panels . the

configuration is of the type used in a high angle-of-attack /near 90degree/

reentry to minimize aerodynamic heating . by unfolding the wingtip

panels into the airstream, a moderate angle-of-attack glide is used for

a controlled landing . the basic configuration tested utilized a

whose area was 25 percent of the total wing area . the effects of

varying the plan form and size of the wingtip panels was studied as well

as the effects of unfolding the wingtip panels in a high angle-of-attack attitude . tests were made at mach numbers of 0.40, 0.60, and

.I 712

.T

low-speed longitudinal aerodynamic characteristics associated with a

series of low-aspect ratio wings having variations in leading-edge

contour .

.A

spencer, b. and hammond, a.d.

.B

nasa tn d-1374, 1962 .

.W

low-speed longitudinal aerodynamic characteristics associated with a

series of low-aspect ratio wings having variations in leading-edge

contour .

an investigation has been conducted at various reynolds numbers and low

subsonic speeds to determine the longitudinal aerodynamic

characteristics associated with a series of low-aspect-ratio wings having

variations in leading-edge contours . the planforms included a highly swept

triangular wing, a rectangular wing, and intermediate wings including

planforms having elliptic and parabolic leading-edge contours, all

having an aspect ratio of 1.33 . the effects of changing aspect ratio for a

given leading-edge contour were investigated for two of the wings

presented,. also included are the longitudinal characteristics associated

with various fuselage sizes . an effort has been made to estimate the

lift variation with angle of attack for the wing planforms of the

present investigation .

improvements in the lifting capabilities at low subsonic speeds associated with a basic triangular planform of low aspect ratio are possible

by slight alterations in leading-edge design, which should still conform

to possible design requirements at hypersonic speeds . these changes in

planform resulted in increases in lift-curve slope, lift at high angles

of attack, and in the maximum untrimmed lift-drag ratio, provided the

fuselage was sufficiently small . the longitudinal stability

characteristics of the majority of planforms indicate more desirable stability

characteristics at high lifts than either a triangular wing or

rectangular wing of the same aspect ratio . the effects of increasing

reynolds number for each of the planforms investigated generally resulted

in slight reductions in the lift at high angles of attack . a method is

presented for estimating the subsonic-lift variation with angle of

attack for the low-aspect-ratio wings of the present investigation and

indicated good agreement with experimental data throughout the

angle-of-attack range of this investigation .

.I 713

.T

static longitudinal stability characteristics of a blunted glider

re-entry configuration having 79.5degree sweepback and 45degree dihedral at

a mach number of 6.2 and angles of attack up to 20degree .

.A

mayo, e.e.

.B

nasa tm x-222. 1959 .

.W

static longitudinal stability characteristics of a blunted glider

re-entry configuration having 79.5degree sweepback and 45degree dihedral at a mach number of 6.2 and angles of attack up to 20degree .

an experimental investigation was conducted at a mach number of 6.2 to

determine the static longitudinal stability characteristics of a model

of a blunted glider reentry configuration having 79.5degree sweepback

and 45degree dihedral . the free-stream reynolds number for the

investigation was 3.0 x 10 based on the basic model length of 7.5 inches .

tests were made through an angle-of-attack range from 0degrees to

investigation showed that incorporating 10degree nose incidence in the

basic model resulted in a lower lift-curve slope, a lower lift-drag

ratio, a higher value of trim lift coefficient, and a decrease in static

longitudinal stability . in comparison, the effect of extending the

configuration length and incorporating 10degrees and 20degrees boattail

angles resulted in smaller changes in the longitudinal stability

characteristics of the model .

.I 714

.T

blockage corrections for three-dimensional flow closed

throat wind tunnels, with considerations of the effect

of compressibility .

.A

herriot,j.g.

.B

naca r.995, 1950.

.W

blockage corrections for three-dimensional flow closed

throat wind tunnels, with considerations of the effect

of compressibility .

theoretical blockage corrections are presented for a body of revolution and for a three-dimensional unswept wing in a circular or rectangular wind tunnel . the theory takes account of the effects of the wake and of the compressibility of the fluid, and is based on the assumption that the dimensions of the model are small in comparison with those of the tunnel throat . formulas are given for correcting a number of the quantities, such as dynamic pressure and mach number, measured in wind-tunnel tests . the report presents a summary and unification of the existing literature on the subject .

.I 715

.T

motion of a ballistic missile angularly misaligned with the flight path upon entering the atmosphere and its effect upon aerodynamic heating, aerodynamic loads and miss distance .

.A

allen,h.j.

.B

naca tn.4048, 1957.

.W

motion of a ballistic missile angularly misaligned with the flight path upon entering the atmosphere and its effect upon aerodynamic heating, aerodynamic loads and miss distance .

an analysis is given of the oscillating motion of a ballistic missile which upon entering the atmosphere

is angularly misaligned with respect

to the flight path . the history of

the motion for some example missiles

is discussed from the point of view

of the effect of the motion on the

aerodynamic heating and loading .

the miss distance at the target due to

misalignment and to small accidental

trim angles is treated . the

stability problem is also discussed for

the case where the missile is

tumbling prior to atmospheric entry .

.I 716

.T

study of the oscillatory motion of manned vehicles

entering the earth's atmosphere .

.A

sommer,s.c. and tobak,m.

.B

nasa memo 3-2-59a, 1959.

.W

study of the oscillatory motion of manned vehicles

entering the earth's atmosphere .

an analysis is made of the oscillatory

motion of vehicles which

traverse arbitrarily prescribed trajectories through the atmosphere .

expressions for the oscillatory motion

are derived as continuous functions

of the properties of the trajectory .

results are applied to a study of

the oscillatory behavior of re-entry

vehicles which have decelerations that

remain within limits of human

tolerance . it is found that a deficiency of

aerodynamic damping for such

vehicles may have more serious consequences

than it does for comparable

ballistic missiles .

.I 717

.T

motions of a short 10degree blunted cone entering a martian atmosphere

at arbitrary angles of attack and arbitrary pitching rates .

.A

peterson, v.l.

.B

nasa tn-d 1326 .

.W

motions of a short 10degree blunted cone entering a martian atmosphere

at arbitrary angles of attack and arbitrary pitching rates .

the dynamic behavior of two probe vehicles entering a martian atmosphere

in a passive manner with arbitrary initial angles of attack and

pitching rates to 12degree per second has been determined . results for an

entry velocity of 21,700 feet per second and an entry angle of -40degree were obtained from machine calculated solutions of the six-degree-of-freedom rigid-body equations of motion using experimental aerodynamic characteristics for the vehicles . one of the vehicles had a flat base and was statically stable in two attitudes /nose forward and base forward/ . the other vehicle, derived from the first by adding a conical afterbody, was statically stable in only one attitude /nose forward/ .
a 10-rpm vehicle spin rate, believed ample for the purpose of distributing solar and aerodynamic heating over the vehicle surface, and model atmospheres encompassing the probable extremes for the planet were also considered .
it was found that while the motion of the flat-based vehicle could be oscillatory about either the nose-forward or base-forward stable trim attitudes when aerodynamic heating rates were high, the range of initial angles of attack resulting in base-forward orientation was reduced by more than a factor of 3. when initial pitch rates were increased from body having only nose-forward stability showed that oscillatory angles of attack at maximum heating-rate conditions probably would not exceed about 25degrees although angles of attack when heating rates were 50 percent of maximum could be as high as 40degree . values of these upper bound angles of attack were essentially independent of initial pitch rates for the range considered . furthermore, the envelope of maximum probable angles of attack was increased only slightly when the vehicle was given a 10-rpm spin rate . the relationship between maximum amplitudes of oscillation and heating rates through high heating portions of the trajectories was preserved when model atmospheres believed to encompass the extreme possibilities for mars were used in the calculations

.I 718

.T

means and examples of aeronautical research in france at onera .

.A

maurice roy

.B

the twenty-second wright brothers lecture

office national d'etudes et de recherches aeronautiques

.W

means and examples of aeronautical research in france at onera .

cosmonautics is currently very much to the forefront in the news .

it embraces and extends aeronautics, and i would like to propose

including both, at least on certain occasions, under a general

denomination of /aerocosmonautics/ .

in your country, the sciences and technology of space are subjects which

have been backed by initial advances and abundantly treated .

since france has not yet launched any artificial satellite or built

any circumlunar space vehicle, i propose to confine myself here to the

field of aeronautics, where there is still so much progress of manifest

utility to accomplish .

i shall accordingly content myself with presenting some examples of

aeronautical research and experiments undertaken in my country by

onera, a body whose mission is akin to that of the illustrious naca,

now nasa, but bearing in mind the considerable difference between the

scales of the respective resources .

.I 719

.T

tumbling bodies entering the atmosphere .

.A

remmler, k.l.

.B

ars jnl. v. 32, january 1962. pp 92-95 .

.W

tumbling bodies entering the atmosphere .

the equations of motion of a tumbling flat plate entering an exponential atmosphere were linearized and solved analytically to obtain a simple expression for the altitude at which tumbling would cease and libration would commence . the plate had only three degrees of freedom, and aerodynamic forces were derived from newtonian impact theory . in the linear analysis, mean values of the drag and pitch damping coefficients so that flutter occurs in the range of a low-speed wind tunnel .

a particular type of construction for supersonic flutter models is described in detail . methods of vibration testing, static testing, and flutter testing are discussed . particular emphasis is placed on the technique of varying flow parameters rather than model parameters to precipitate flutter . the tool for varying flow parameters is the variable mach number supersonic test section of the massachusetts institute of technology blowdown wind tunnel . the aerodynamic features of the supersonic test section are presented .

.I 720

.T

a note on the use of sandwich structures in severe acoustic environments .

.A

d. j. mead, d. c. ae.

.B

d. j. mead, d.c.ae.

.W

a note on the use of sandwich structures in severe acoustic

environments .

this paper reviews some of the experience to date of using sandwich type

structures in severe acoustic pressure environments . the methods

used for testing sandwich structures for acoustic fatigue are described

and their limitations considered . experimental and theoretical work

relating to the damping and mode-frequency relationships of certain

sandwich configurations is also reviewed .

special attention is given to the estimation of the stress in the bond

of a honeycomb sandwich panel subjected to sudden pressure

fluctuations . a /uni-modal/ theory is presented, relating the

mean-square bond-stress to the random exciting pressure and panel dynamic

characteristics . this theory indicates that tensile bond stresses

may be encountered of up to six times the local r.m.s. exciting

pressure . these must be combined with bending and shear stresses

to obtain the principal stresses which precipitate bond fatigue

failures .

finally, an outline is given of some of the lines of future research

which should lead to the achieving of the maximum possible fatigue

resistance from sandwich configurations .

.I 721

.T

near noise field of a jet engine exhaust .

.A

callaghan,e.e., howes,w.l. coles,w.d. and mull,r.h.

.B

naca r.1338.

.W

near noise field of a jet engine exhaust .

aircraft structures located in the near noise field of a jet engine are subjected to extremely high fluctuating pressures that may cause structural fatigue . studies of such structures have been limited by lack of knowledge of the loadings involved .

the acoustic near field produced by the exhaust of a stationary turbojet engine having a high pressure ratio was measured for a single operating condition without afterburning . the maximum over-all sound pressure without afterburning was found to be about 42 pounds per square foot along the jet boundary in the region immediately downstream of the jet-nozzle exit . with afterburning the maximum sound pressure was increased by 50 percent . the largest sound pressures without afterburning were obtained on a constant percentage band width basis in the frequency range from 350 to 700 cps .

additional tests were made at a few points to find the effect of jet velocity on near-field sound pressures and to determine the difference in value between sound-pressure levels at rigid surfaces and corresponding free-field values . near the jet nozzle, over-all sound pressures were found to vary as a low power (approx. unity) of the jet velocity . over-all sound-pressure levels considerably greater than the corresponding free-field levels were recorded at the surface of a rigid plate placed along the jet boundary .

the downstream locations of the maximum sound pressure at any given frequency along the jet-engine-exhaust boundary and the longitudinal turbulent-velocity maximum of the same frequency along a small cold-air jet at 1 nozzle-exit radius from the jet axis were found to be nearly the same when compared on a dimensionless basis . also, the strouhal number of the corresponding spectra maximums was found to be nearly equal at similar distances downstream .

in addition to the magnitude and frequency distribution of the acoustic pressures, it is necessary to know the cross correlation of the pressure over the surface area . cross-correlation measurements with microphones were made for a range of jet velocities at locations along the jet and at a distance from the jet . free-field correlations of the over-all sound pressure and of the sound pressure in frequency bands from 100 to 1000 cps were obtained both longitudinally and laterally . in addition, correlations were obtained with microphones mounted at the surface of a rigid plate that was large compared with the distance over which a positive correlation existed .

the region of positive correlation was generally found to increase with distance downstream of the engine to 6.5 nozzle-exit diameters, but remained nearly constant thereafter . in general, little change in the correlation curves was found as a function of jet velocity or frequency-band width . the distance from unity correlation to the first zero correlation was greater for lateral than for longitudinal correlations for the same

conditions and locations . the correlation curves obtained in free

space and on the surface of the plate were generally similar .

  the results are interpreted in terms of pressure loads on

surfaces .

.I 722

.T

random excitation of a tailplane section by jet noise .

.A

clarkson,b.l. and ford,r.d.

.B

univ. southampton r. a.a.s.u.171.

.W

random excitation of a tailplane section by jet noise .

  the response of a section of tailplane structure to both discrete

and random noise pressures has been studied in detail . initially the

specimen was mounted behind a jet engine and the induced strains were

analysed with the object of determining both the resonant frequencies

and the corresponding modes of vibration . during these tests a survey

was made of the spectrum and correlation pattern of the jet noise on

the surface of the model . secondly the specimen was mounted in front

of a loudspeaker in an acoustics laboratory and the structural

resonances were excited by means of discrete frequency sound . the mode

shapes were studied in detail with the aid of a stroboscope .

  it is concluded that the tailplane skin on this particular piece

of structure only responds to any significant degree in one structural

mode . although reasonable comparison has been obtained between the random and discrete tests, it was not possible to calculate the induced

stresses using the observed mode shapes and measured pressure

excitation .

.I 723

.T

on the fatigue failure of structures due to vibrations excited

by random pressure fields .

.A

alan powell

.B

the university, southampton

.W

on the fatigue failure of structures due to vibrations excited

by random pressure fields .

on the assumption that the forced modes of vibration of a structure,

subjected to pressure fluctuations random in time and space, can be

approximated by the composition of the motions of the uncoupled

natural modes, a general analysis is made using the ideas of vibration

theory and spectrum analysis . the power spectrum, and hence the

rms value, of any quantity depending linearly upon structural

distortions is derived and it involves a quantity (called the /joint

acceptance/) concerning the spacewise structure of the pressure

field and of the geometry of the modes of vibration . it is shown how

this result may be used (on assuming /normal/ randomness) to estimate

the fatigue life on the hypothesis of cumulative damage .

.I 724

.T

structural acoustic proof testing .

.A

schjeldrup,h.c.

.B

air eng. 1959.

.W

structural acoustic proof testing .

with the introduction of high-powered

propulsion systems, and paralleling their

continued development, an

accompanying increase in acoustical problems has arisen .

of these acoustical problems, that of

acoustical fatigue failures has become paramount

in the eyes of the structural engineer . aircraft

designed to normal strength requirements have

been known literally to fall apart under acoustical

loading . this problem has required much

endeavour to produce a solution, and considerable

structural research, based upon results of siren or

other testing, have proved inadequate . this

failure to find a satisfactory solution has resulted

in the conviction that the final proof of a design

can be found only in proof testing . proof testing,

in the acoustic fatigue sense, is the testing of a

design structure in a simulated acoustical

environment for a period of time long enough to assure

equality with design life .

.I 725

.T

the response of a typical aircraft structure to jet

noise .

.A

clarkson,b.l. and ford,r.d.

.B

j. roy. aero. soc. 1962.

.W

the response of a typical aircraft structure to jet

noise .

an analysis is made of experimentally determined mode shapes

excited on the rear structure of a modern airliner by jet noise from

a pod-mounted turbojet engine . power spectra of stresses

determined from strain-gage measurements are obtained and cross

correlated . extensive measurements were made on skin panels of the

fuselage and elevator and limited ones were made on fuselage

stringers and frames . the skin-panel results are compared with

theoretical predictions . reviewer believes that this paper is of

considerable value for those concerned with response of

aircraft-type structures to jet-induced noise .

.I 726

.T

on structural fatigue under random loading .

.A

miles,j.w.

.B

j. ae. scs. 21, 1954, 753.

.W

on structural fatigue under random loading .

experience has shown that the fluctuating loads induced by a

jet may cause fatigue failure of aircraft structural components .

in order to throw some light on this and similar problems, the

stress spectrum and the /equivalent fatigue stress/ of an elastic

structure subjected to random loading are studied . the analysis

is simplified by assuming the structure to have only a single

degree of freedom and by using the concept of cumulative damage,

the results being expressed in terms of quantities that can be

directly measured . as an example, a similarity expression for

the probable value of the equivalent fatigue stress of a panel

subjected to jet buffeting is derived .

.I 727

.T

a study of the acoustic fatigue characteristics of

some flat and curved aluminium panels exposed to random

and discrete noise .

.A

hess,r.w., herr,r.w. and mayes,w.h.

.B

nasa tn.d1.

.W

a study of the acoustic fatigue characteristics of

some flat and curved aluminium panels exposed to random

and discrete noise .

a study was made of the fatigue
life of simple 2024-t3
aluminum-alloy panels measuring 11 by 13
inches and exposed to both
discrete-frequency noise from a siren and
random noise from an air jet . noise
levels varied from approximately
panel variables included thickness,
edge conditions, curvature, and
static-pressure differential .

no significant differences were noted
in the nature of failures
experienced for the two types of loadings .
at a given root-mean-square
stress level, the failure times were
generally shorter for the random
loading than for the discrete-frequency
loading . these differences in
failure times were noted to be a function
of stress level, the larger
differences occurring at the lower stress levels .

increases in time to failure were
obtained as a result of increased
panel thickness, increased panel curvature,
and particularly for increased
static-pressure differential across curved panels .

for the discrete-type loading,

the location of weak points in these

simplified structural designs can be

satisfactorily accomplished but

quantitative predictions of fatigue

life are much more difficult .

.I 728

.T

free vibrations of continuous skin stringer panels .

.A

lin, y.k.

.B

j. appl. mech. december 1960 .

.W

free vibrations of continuous skin stringer panels .

the determination of the natural frequencies and normal modes of

vibration for continuous panels, representing more or less typical fuselage

skin-panel construction for modern airplanes, is discussed in this paper

are considered . a numerical example is presented, and analytical

results for a particular structural configuration agree favorably with

available experimental measurements .

.I 729

.T

stresses in continuous skin stiffener panels under random loading .

.A

lin, y.k.

.B

j. aero. sc. january 1962 .

.W

stresses in continuous skin stiffener panels under random loading .

theoretical aspects involved in the prediction of stress levels for

continuous skin-stiffener panels subjected to a random pressure field

are considered in the light of powell's general theory for statistical

superposition of modal response . the choice of structural model is

dictated by the prevalence of skin-stiffener construction in modern

flight vehicle design . the present study clearly demonstrates that any

truly adequate prediction of stress levels in actual aircraft structures

requires a much better representation of structural characteristics

than can be provided by single panel idealizations . in an example

considering fuselage panels exposed to jet engine noise, essential

agreement is shown with experimental data, although better correlation is

shown for rms stress than for power spectrum . it is shown that

reduction of stress level by increasing damping is effective only in the

higher frequency range .

.I 730

.T

on the bending of a clamped plate .

.A

weinstein,a., rock,d.h.

.B

q. app. math. 2, 1944, 262.

.W

on the bending of a clamped plate .

the present paper contains an application

of a recently developed variational

method to the boundary value problem of the bending of a clamped plate of arbitrary shape . it will be shown that this problem can be linked to the simpler problem of the equilibrium of a membrane by a chain of intermediate problems, which can be solved explicitly and in finite form in terms of the membrane problem . in the intermediate problems, the deflection converges uniformly in the domain of the plate of the clamped plate, and the derivatives of all orders of the deflection converge uniformly in every domain completely interior to the plate . (in the ritz method, not even the convergence of the slopes can be guaranteed .) the method yields numerical results for plates of all shapes for which the membrane problem (which we shall call the base problem) admits an explicit solution . as an example we shall consider a clamped square plate under a uniform load . this problem has been the object of numerous investigations, some of which are theoretical, while others are purely numerical, use infinite simple and double series, and operate with an infinite number of linear equations and an infinite number of unknowns . an inspection of the general

formulae derived in the present paper,

formulae which become simple in numerical

applications, would show how some of the

numerical methods might be rendered

rigorous . the convergence of higher

derivatives is of great practical interest for

the approximate computation of the stresses .

.I 731

.T

upper and lower bounds for the solution of the first

biharmonic boundary value problem .

.A

diaz,g.b., and greenberg,h.g.

.B

j. math. phys. 27, 1948, 193.

.W

upper and lower bounds for the solution of the first

biharmonic boundary value problem .

let w(x,y) be a solution of the boundary value problem

where r is a plane

domain with the boundary c . the authors obtain upper and

lower bounds for, the value of w at a point in r,

by a method which is applicable to many other problems .

if u is a function satisfying the boundary conditions

and v is a function satisfying the partial differential

equation, then the authors obtain by applying green's classical identity and schwarz's inequality a pair of inequalities

of the form where .

together with the function w the authors consider a

function the solution of the boundary value problem

on c, and in

analogy with the functions u and v associated with the

function w a pair of functions and associated with the

function . in the expression for derived from green's

classical identity appears an unknown line integral

containing the values of w and on c . but the same line

integral appears also in the expressions for

to which the above inequalities are

applicable .

in this way the authors obtain two inequalities of the form

where b and b', respectively, are approximate

values of . in order to improve these bounds

one may add to u a linear set of functions and to v a

linear set of functions and then minimize h(u-v) in order

to determine the coefficients of the best linear combinations .

if the sequences and are complete in a certain sense

defined by the authors the approximations will converge to

the value .

.I 732

.T

on the analogues relating flexure and extension of

flat plates .

.A

southwell,r.v.

.B

q. j. mech. app. math. 3, 1950, 257.

.W

on the analogues relating flexure and extension of

flat plates .

the displacement of a flat plate bent

by transverse loading, and the extensional

or in 'plane stress', are governed by equations

of identical form ,. and the boundary

conditions have identical form when

edge-displacements are specified in the flexural,

edge-tractions in the extensional problem,

so mathematically, in these circumstances,

only a single problem is presented . this,

the 'first analogue' relating flexure and

extension, is well known .

a 'second analogue', relating the flexural

problem when edge-tractions with the

extensional problem when edge-displacements

are specified, is believed to have been

first propounded in 1941 . by introducing

two quantities u and v, analogous with

the components u and v of extensional

displacement, it permits a treatment of the

flexural problem by any method--e.g.

which yields extensional solutions of this second type .

in this paper both analogues are combined

in an inclusive statement covering the

perforated (multiply connected) plates which

were discussed in 1948 . reasons are

stated for believing that 'two-diagram technique'

is preferable in problems governed

by 'mixed' boundary conditions .

.I 733

.T

the bending of a sectorial plate .

.A

carrier,g.f.

.B

j. app. mech. 11, 1944, 134.

.W

the bending of a sectorial plate .

the problem of evaluating the bending moments,

existing in a uniformly loaded clamped plate having the form

of a sector of a ring, is one which arises in connection with

the stress analysis of reinforced piston heads and in other

design problems . in this paper, expressions are derived

for the bending moments along the edges of such a plate .

similar problems, i.e., those of the clamped rectangular

plate under uniform pressure, under a central

concentrated load, and that of the simply supported sector of a

disk under uniform pressure, have been discussed by previous authors . the general approach used in the

foregoing problems is adopted in the present case ,. a

considerable reduction in the computational work is achieved,

however, by the use of an integral-equation method of

solving the boundary-condition equations . numerical

results are obtained for plates of various dimensions, and

the edge moment distributions are plotted for these cases .

curves are also plotted which indicate the relationship

existing between the maximum bending moments derived

for sectorial plates and those previously obtained for

clamped rectangular plates of similar size .

.I 734

.T

the bending of uniformly loaded clamped plate in the

form of a circular sector .

.A

hasse,h.r.

.B

q. j. mech. app. math. 3, 1950, 271.

.W

the bending of uniformly loaded clamped plate in the

form of a circular sector .

the deflexion of a uniformly loaded

plate in the form of a semicircle clamped

along its boundary is obtained by a

method due to weinstein . this problem

requires the solution of the biharmonic

equation where z is given,

subject to the conditions that w = 0 and

on the boundary, n being the

direction of the outward normal . the solution

is expressed in the form

where, writing is found

by solving (in succession) two harmonic

equations of the forms where z may

be zero, and where f and

have to satisfy certain boundary conditions .

the constants are then determined

to satisfy the boundary condition .

numerical calculations show that five or six

terms of the series give a

good approximation to the accurate value as

judged by the closeness with which

the approximate solution satisfies the boundary

condition . the

procedure to be adopted in the case of the general

circular sector and for non-uniform

loading is indicated briefly .

the connexion between the deflexion problem

and that of plane strain in which

the stress function satisfies the equation,

where and have given

values on the boundary, is discussed as a preliminary

to the further consideration

of the latter problem by a method of the same type .

.I 735

.T

the bending of uniformly loaded sectorial plates with

clamped edges .

.A

conway,h.d. and huang,m.k.

.B

j. app. mech. 19, 1962, 5.

.W

the bending of uniformly loaded sectorial plates with

clamped edges .

this paper analyzes the bending of a sectorial plate,

clamped on all edges and subjected to uniformly

distributed load, by using two different methods of superposition

on the elementary solution for a uniformly loaded circular

plate with a clamped edge .

.I 736

.T

the bending of a wedge shaped plate .

.A

woinowsky-krieger,s.

.B

j. app. mech. 20, 1953, 77.

.W

the bending of a wedge shaped plate .

a general method of solution is given in this paper for

the problem of bending of a wedge-shaped thin elastic
plate with arbitrary boundary conditions on the radial edges
in the case of a single load . the solution is carried out for
a plate with clamped edges and a single load on the bisector
radius of the plate . stress distribution along the edges
is shown and the behavior of the solution near the corner
point is discussed for several opening angles of the plate .

.I 737

.T

on the analysis of elastic plates of variable thickness .

.A

mansfield,e.h.

.B

q. j. mech. app. math. 15, 1962, 167.

.W

on the analysis of elastic plates of variable thickness .

the extensional and flexural equations
governing the elastic behaviour of a plate
of variable thickness are expressed in
terms of the laplacian operator .
temperature variations in the plane of the
plate and across the thickness of the plate
are taken into account .

general solutions are given for a rectangular
plate whose thickness varies
exponentially along the length, and for a circular,
or annular, plate whose thickness varies

as a power of the radius .

the large-deflexion equations, including

effects of initial irregularities, are also

discussed .

.I 738

.T

finding zero's of arbitrary functions .

.A

frank,w.l.

.B

j. assoc. comput. mach. 5, 1958, 154.

.W

finding zero's of arbitrary functions .

a method for finding real and complex

roots of polynomial equations, due to

d. muller, is applied to finding roots of

general equations of the form f(z) = 0,

where f(z) is analytic in the neighborhood

of the roots . the procedure does not

depend on any prior knowledge of the

location of the roots nor on any special

starting process . all that is required is

the ability to evaluate f(z) for any

desired value of z . multiple roots can also be

obtained . a general purpose program,

prepared for the univac scientific 1103

and 1103a, is described and numerical

results are presented for the following

applications .. finding eigenvalues of

differential operators ,. finding eigenvalues of

arbitrary matrices ,. finding zeros of the

generalized eigenvalue problem ,. finding

roots of a number of transcendental

equations .

.I 739

.T

the buckling of thin cylindrical shells under axial

compression .

.A

von karman,t. and tsien,h.s.

.B

j. ae. scs. 8, 1941, 303.

.W

the buckling of thin cylindrical shells under axial

compression .

in two previous papers the authors have discussed

in detail the inadequacy of the classical theory of

thin shells in explaining the buckling phenomenon of

cylindrical and spherical shells . it was shown that

not only the calculated buckling load is 3 to 5 times

higher than that found by experiments, but the

observed wave pattern of the buckled shell is also different

from that predicted . furthermore, it was pointed

out that the different explanations for this discrepancy advanced by l. h. donnell and w. flugge are untenable when certain conclusions drawn from these explanations are compared with the experimental facts . by a theoretical investigation on spherical shells the authors were led to the belief that in general the buckling phenomenon of curved shells can only be explained by means of a non-linear large deflection theory . this point of view was substantiated by model experiments on slender columns with non-linear elastic support . the non-linear characteristics of such structures cause the load necessary to keep the shell in equilibrium to drop very rapidly with increase in wave amplitude once the structure started to buckle . thus, first of all, a part of the elastic energy stored in the shell is released once the buckling has started,. this explains the observed rapidity of the buckling process . furthermore, as it was shown in one of the previous papers the buckling load itself can be materially reduced by slight imperfections in the test specimen and vibrations during the testing process .

in this paper, the same ideas are applied to the case of a thin uniform cylindrical shell under axial compression . first it is shown by an approximate calculation that again the load sustained by the shell drops with increasing deflection . then the results of this calculation are used for a more detailed discussion of the buckling process as observed in an actual testing machine .

.I 740

.T

the behaviour of a cylindrical shell under axial compression

when the buckling load has been exceeded .

.A

leggett,d.m.a. and jones,r.p.n.

.B

arc r + m.2190, 1942.

.W

the behaviour of a cylindrical shell under axial compression

when the buckling load has been exceeded .

the value of the compressive stress at which

a thin circular cylindrical shell becomes unstable has been

worked out theoretically by southwell (1914) . subsequent

experimental results, however, have indicated that this

value is appreciably too high and that the form of distortion

which occurs in practice differs from that assumed in theory .

in recent years much work has been done on this problem

in america . lundquist (1933) and donnell (1934)

have concluded that the buckling of a cylindrical shell is greatly

influenced by initial irregularities,. von karman and

tsien (1941) have indicated that a thin cylindrical shell can be

maintained in a buckled state by a compressive load

considerably smaller than that previously predicted by theory .

the present paper is an extension of the work of von karman

and tsien . it shows that the smallest load which will keep a thin cylindrical shell in a buckled condition is about one-third

of that given by southwell, a result in very fair

agreement with experiment, and that once the cylinder has buckled,

and so long as the stresses remain within the elastic

range of the material, the cylinder has only about one-quarter of its

original stiffness .

.I 741

.T

the behaviour of thin cylindrical shells after buckling

under axial compression .

.A

michielsen,h.f.

.B

j. ae. scs. 15, 1948, 738.

.W

the behaviour of thin cylindrical shells after buckling

under axial compression .

the fundamental investigations of von karman and tsien

on the buckling of cylindrical shells under axial compression

are continued . the energy expression is simplified and

minimized with respect to the axial and circumferential wave-length

parameters . solution of the equations obtained yields curves

of the reduced average stress and of the wave dimensions plotted

against the reduced average strain . they illustrate the behavior

of the cylinder during the buckling process . the minimum

buckling stress is found to be 0.195e(tr) .

.I 742

.T

post-buckling behaviour of axially compressed circular

cylinder shells .

.A

kempner,j.

.B

j. ae. scs. 21, 1954, 329.

.W

post-buckling behaviour of axially compressed circular

cylinder shells .

the postbuckling characteristics of an axially compressed

thin-walled circular cylindrical shell loaded either by dead weights

or by a rigid testing machine are determined . it is shown that

for either loading condition the minimum applied stress in the

postbuckling region is 0.182(er) and that the region of stable

equilibrium corresponding to loading by the rigid testing machine

includes and extends beyond that obtained with dead weight

loading . the work here described is a continuation of work done

earlier by von karman and tsien, by michielsen, and by leggett

and jones .

.I 743

.T

new developments in the nonlinear theories of the buckling of thin

cylindrical shells .

.A

w. f. thielemann

.B

deutsche versuchsanstalt fur luftfahrt mulheim (ruhr), germany

.W

new developments in the nonlinear theories of the buckling of thin

cylindrical shells .

in the present paper a short survey will be given first of the buckling

and postbuckling behavior of isotropic cylindrical shells subjected

to different loading conditions as obtained by the nonlinear theory

of finite deflections of shells during the last twenty years .

next a report will be given on new investigations carried out in the

structures department of the dvl concerning the elastic stability of

isotropic and orthotropic cylindrical shells loaded in axial compression

and internal pressure . these studies are based on the nonlinear theory

of finite deformations . the theoretical rsults will be compared with

new experimental results obtained with a series of axially loaded

pressurized isotropic and orthotropic cylindrical shells .

.I 744

.T

lower buckling load in the non-linear buckling theory

of thin shells .

.A

tsien,h.s.

.B

q. app. math. 5, 1947, 236.

.W

lower buckling load in the non-linear buckling theory

of thin shells .

for thin shells the relation between

the load p and the deflection beyond the

classical buckling load is very often

non-linear . for instance, when a uniform thin

circular cylinder is loaded in the axial

direction, the load p when plotted against the

end-shortening has the characteristic

shown in fig. 1 . if the strain energy s and the

total potential are calculated,

their behavior can be represented by the

curves shown in figs. 2 and 3 . it can be

demonstrated that the branches oc and ab

corresponds to stable equilibrium configurations

and the branch bc to unstable

equilibrium configurations . the point b is then

the point of transition from stable to

unstable equilibrium configurations .

.I 745

.T

an automatic method for finding the greatest or least

value function .

.A

rosenbrock,h.h.

.B

computer jnl. 1960.

.W

an automatic method for finding the greatest or least

value function .

the greatest or least value of a function of several

variables is to be found when the variables

are restricted to a given region . a method is

developed for dealing with this problem and is

compared with possible alternatives . the method

can be used on a digital computer, and is

incorporated in a program for mercury .

.I 746

.T

aeroelastic problems in connection with high speed

flight .

.A

broadbent,e.g.

.B

j. roy. aero. soc. 1956.

.W

aeroelastic problems in connection with high speed

flight .

a review is given of developments

in the field of aeroelasticity during the

past ten years . the effect of steadily increasing

mach number has been two-fold .. on

the one hand the aerodynamic derivatives have

changed, and in some cases brought new

problems, and on the other hand the design for

higher mach numbers has led to thinner

aerofoils and more slender fuselages for which the required stiffness is more difficult to

provide . both these aspects are discussed,

and various methods of attack on the

problems are considered . the relative merits

of stiffness, damping and massbalance

for the prevention of control surface flutter are

discussed . a brief mention is made of

the recent problems of damage from jet efflux

and of the possible aeroelastic effects

of kinetic heating .

.I 747

.T

bodt freedom flutter of ground launched rocket models

at supersonic and high subsonic speeds .

.A

gaukroger,d.r.

.B

rae r. struct.237, 1957.

.W

bodt freedom flutter of ground launched rocket models

at supersonic and high subsonic speeds .

a theoretical investigation of symmetric

body freedom flutter of a rocket model is described . the results

confirm that structural failures of models were caused

by this type of flutter, and an extension of the investigation

indicates the parameters that are of importance . a high

ratio of body to wing mass and a well forward position of the overall centre of gravity are conditions under which flutter

may occur . increase of body pitching radius of gyration

and tailplane volume are beneficial .

it is concluded that this type of flutter may be significant

in some aircraft designs, and that the canard has no

advantage in this respect over the conventional lay-out of wing

and tailplane .

.I 748

.T

subsonic aerodynamic flutter derivatives for wings and control surfaces,

/compressible and incompressible flow/ .

.A

minhinnick, i. t.

.B

r.a.e. rep. structs. 87. july 1950 .

.W

subsonic aerodynamic flutter derivatives for wings and control surfaces,

/compressible and incompressible flow/ .

this report gives tables of the two-dimensional subsonic flutter

derivatives,. where possible the values given are based on the published

work of various authors, but some have been specially calculated for

this report . wing derivatives are given for mach numbers 0, 0.5, 0.6

and 0.7 for the frequency parameter range 0 /0.04/ 0.2 /0.2/ 1.6 and

mach numbers 0 and 0.7 for frequency parameter 5.0 . control surface

derivatives are given for mach numbers 0 and 0.7 for control surface/

wing chord ratios 0.02 /0.02/ 0.10 /0.05/ 0.50 and frequency parameters

are also given for mach numbers 0, 0.5, 0.6 and 0.7 for frequency parameter 0 /0.04/ 0.2 /0.2/ 1.4 . control surface-tab derivatives are

given for some particular values of the variables and methods of

obtaining approximate values of these derivatives for other values of

the variables are suggested . control surface and tab derivatives are in

all cases for no aerodynamic balance .

.I 749

.T

the aerodynamic effects of aspect ratio and sweepback

on wing flutter .

.A

molyneux,w.g. and hall,h.

.B

arc r + m.3011.

.W

the aerodynamic effects of aspect ratio and sweepback

on wing flutter .

the report describes tests to obtain direct

measurements of the aerodynamic effects of aspect ratio and

sweepback on wing flutter . the tests were made on rigid

wings with root flexibilities .

it is shown that measured effects of aspect ratio and

sweepback on the flutter of these wings can be represented

quite closely in flutter calculations based on two-dimensional

flow theory by multiplying the two-dimensional

aerodynamic coefficients by appropriate factors . the effect of

sweepback is represented by multiplying all aerodynamic

coefficients by cos, where is the wing leading-edge sweepback, and the effect of aspect ratio is represented by

multiplying the aerodynamic damping coefficients by 1f(a) and,

the stiffness coefficients by 1(f(a)) where a is the

aspect ratio .

for the wings tested an average value for

f(a) is f(a) = (1 + (0.8a)) .

.I 750

.T

transonic flow in two dimensional and axially symmetrical

nozzles .

.A

hall,i.m.

.B

arc 23,347, 1961.

.W

transonic flow in two dimensional and axially symmetrical

nozzles .

by means of suitable expansions

in inverse powers of r, the

radius of curvature of the nozzle profile

at the throat measured in throat

half-heights, the velocity components

in the throat region of a

convergent-divergent nozzle can be

calculated . the first three terms of

the series solution have been obtained

both for two-dimensional and for

axially-symmetric nozzles . the

numerical accuracy of the solution is

confirmed by comparison with the

known exact solution along the branchline .

.I 751

.T

a note on the use of end plates to prevent three dimensional

flow at the ends of bluff cylinders .

.A

cowdrey,c.f.

.B

npl aero. r.1025, 1962.

.W

a note on the use of end plates to prevent three dimensional

flow at the ends of bluff cylinders .

the results are given of

some observations of the effects of

end plates on the three-dimensional

separated flow at the ends of

cylindrical models . while these are

by no means exhaustive, it is felt

that they are of sufficient interest

to merit putting on record .

.I 752

.T

slender not-so-thin wing theory .

.A

cooke,j.c.

.B

rae r.aero.2660, 1962.

.W

slender not-so-thin wing theory .

a method for making an approximate thickness correction to slender

thin-wing theory is presented . the method is tested by applying it to

cones with rhombic cross-sections and the agreement is found to be good

if the cones are not too thick . it is then suggested that the

thickness correction to slender thin-wing theory may be applied

unchanged to linear thin-wing theory . this suggestion is compared with

some experiments on delta wings and it is found that there is

considerable improvement over thin-wing theory near the centre line, but that

this improvement is not maintained as the wing tips are approached .

.I 753

.T

development of a quasi-steady approach to flutter and

correlation with kernel-function results .

.A

gravitz,s.i., laidlaw,w.r., bryce,w.w. and cooper,r.e.

.B

j.ae.scs. 29, 1962, 445.

.W

development of a quasi-steady approach to flutter and

correlation with kernel-function results .

the quasi-steady approach to flutter utilizes experimental or

theoretical steady-state aerodynamic data to arrive at increased understanding of the flutter mechanism, and also, in many cases, acceptably accurate quantitative flutter predictions . circulation lag effects are neglected, but aerodynamic damping is included in the evaluation of the air forces . situations requiring the inclusion of rate aerodynamics for accurate flutter estimation are specified .

a quasi-unsteady approach is also discussed, in which the approximate magnitude of the circulation lag function at flutter is included in simple modifications of quasi-steady parameters .

closed-form solutions are derived for the flutter characteristics of a typical section with and without rate aerodynamics . application is then made to the rational flutter analysis of three-dimensional multi-degree-of-freedom lifting surfaces .

a specific planform is evaluated in the mach-number range from zero to two . quasi-steady, quasi-unsteady, and kernel-function results are compared subsonically . quasi-steady results are utilized supersonically .

primary applications of the quasi-steady approach are in the areas of preliminary design and parameter-variation studies, modification of more sophisticated flutter theories to force compatibility with available steady-state data, and flutter evaluation of complex configurations which can be rationally analyzed by steady-state aerodynamic theories, but for which no complete unsteady aerodynamic theories are presently available .

.I 754

.T

heat transfer through laminar boundary layers on semi-infinite cylinders of arbitrary cross section .

.A

bourne,d.e. and wardle,s.

.B

j. ae. scs. 29, 1962, 460.

.W

heat transfer through laminar boundary layers on

semi-infinite cylinders of arbitrary cross section .

this paper shows how to calculate the rate of heat transfer through a laminar boundary layer on a semi-infinite cylinder of arbitrary cross section . the cylinder is placed in a stream of incompressible fluid, the flow at infinity being parallel to the generators, and is maintained at a uniform temperature . a series solution for small downstream distances and an asymptotic formula for large downstream distances are given . to cover the intermediate range an approximate pohlhausen solution is obtained,. a correction of the error involved in the pohlhausen solution is suggested which, it is believed, will lead to final errors of at most 2 percent . the calculations are applied to elliptic cylinders, and illustrate the effect on the local rate of heat transfer of varying the ratio of the major and minor axes of cross section, the length of perimeter being held fixed .

.I 755

.T

oscillatory derivative measurements on sting-mounted

wind tunnel models method of test and results for pitch

and yaw on a cambered ogee wing at mach numbers up to 2. 6.

.A

thompson,j.s.

.B

rae r.aero.2668, 1962.

.W

oscillatory derivative measurements on sting-mounted

wind tunnel models method of test and results for pitch

and yaw on a cambered ogee wing at mach numbers up

to 2. 6.

this report describes a method which has been developed for measuring oscillatory derivatives on sting-mounted models in the 8 ft by 8 ft supersonic tunnel at r.a.e. bedford . direct and cross derivatives with respect to angular displacements and velocities in pitch and yaw have been measured satisfactorily, and results are given of tests on a cambered ogee wing at six mach numbers from 0.2 to 2.6 . some tests were made on this model in the course of the preliminary development work in the 13 ft by 9 ft low speed wind tunnel, and results of these are included .

.I 756

.T

further comments on the inversion of large structural matrices .

.A

charles h. samson, jr.

.B

professor, departments of aeronautical and civil engineering,

a. and m. college of texas

.W

further comments on the inversion of large structural matrices .

in a recent note, klein referred to a paper co-authored by the writer, and to ref. 3 . regarding the subject of inversion of large-order matrices, klein stated that he would show 'that the situation is not as hopeless as the anove-mentioned authors intimate' .

the purpose of this note is not to take exception to klein/s conclusions, but rather to disagree with his implication that the authors of ref. 2 were pessimistic with respect to large-matrix inversions . two general methods of analysis were treated.. the method of consistent distortion and the method of transfer matrices . the first method leads directly to a relatively large matrix of structural coefficients of both internal forces and displacements . this matrix must be inverted to solve the problem . the second method ultimately produces a relatively small matrix requiring inversion., however, to arrive at this point one must perform a number of matrix multiplications .

.I 757

.T

an investigation of the flow about a plane half-wing

of cropped delta planform and 6(symmetrical section

at stream mach numbers between 0. 8 and 1. 41 .

.A

rogers,e.w.e., hall,i.m. and berry,c.j.

.B

arc r + m.3286, 1960.

.W

an investigation of the flow about a plane half-wing of cropped delta planform and 6(symmetrical section at stream mach numbers between 0. 8 and 1. 41 .

a study has been made of the flow development over the wing as the incidence and stream mach number vary and this is illustrated by surface pressure distributions and oil-flow patterns . the growth and movement of the two main surface shocks (the rear and forward shocks) is discussed, and conditions for flow separation through these shocks are considered . for the rear shock, which has little sweep, these conditions are similar to those for shock-induced separation on two-dimensional aerofoils . the forward shock is comparatively highly swept and separation seems to correspond to two rather different but simultaneously-attained conditions, one related to the component mach number normal to the shock front and the other to the position of the reattachment line .

the flow in the region between the leading edge and the forward shock is shown to have certain characteristics analogous to those found upstream of the shock on two-dimensional aerofoils . to the rear of the forward shock, but ahead of the rear shock, the flow at low supersonic speeds resembles in some respects that about a simple cone .

the general flow development is related in the text to the wing lift and pitching moment, and the drag .

the first two are most affected by the aft movement

of the rear shock, which also stimulates the transonic

drag rise . the lift-dependent drag is shown to be

influenced by the appearance of leading-edge separation

and possibly also by some stage in the development

of the forward shock .

the flow over the cropped-delta planform is

noteworthy for the absence of the strong outboard shock

and this is attributed partly to the cropped tip and

partly to the unswept trailing edge . a comparison is made

with results obtained during preliminary tests in

which the wing planform closely resembled that of a true

delta .

.I 758

.T

the lower bound of attainable sonic-boom over-pressure

and design methods of approaching this limit .

.A

carlson,h.w.

.B

nasa tn.d1494, 1962.

.W

the lower bound of attainable sonic-boom over-pressure

and design methods of approaching this limit .

from a study of existing sonic-boom

theory it has been possible to establish

an approximate lower bound of attainable sonic-boom overpressure, which depends

only on the airplane length, weight, and

volume and on the flight conditions .

this lower bound may be approached over

a narrow range of flight conditions

through the application of appropriate

design considerations . in general, for

intermediate values of lift coefficient

the major portion of the lift generating

surfaces must be located aft of the

maximum cross-sectional area, whereas for

higher values of lift coefficient

the maximum area must be well forward and or

the lift-producing surfaces must extend well toward the airplane nose .

.I 759

.T

stability investigation of a blunted cone and a blunted

ogive with a flared cylinder afterbody at mach numbers

from 0. 30 to 2. 85 .

.A

coltrane,l.c.

.B

nasa tn.d1506, 1962.

.W

stability investigation of a blunted cone and a blunted

ogive with a flared cylinder afterbody at mach numbers

from 0. 30 to 2. 85 .

a cone with a blunt nose tip and a blunt nose tip and a 20 flared cylinder afterbody have been tested in free flight over a mach number range from 0.30 to 2.85 and a reynolds number range from 1 x 10 to 23 x 10 . time histories, cross plots of force and moment coefficients, and plots of the longitudinal-force coefficient, rolling velocity, aerodynamic center, normal-force-curve slope, and dynamic stability are presented . with the center-of-gravity location at about models were both statically and dynamically stable throughout the mach number range . for the cone, the average aerodynamic center moved slightly forward with decreasing speeds and the normal-force-curve slope was fairly constant throughout the speed range . for the ogive, the average aerodynamic center remained practically constant and the normal-force-curve slope remained practically constant to a mach number of approximately 1.6 where a rising trend was noted . maximum drag coefficient for the cone, with reference to the base area, was approximately 0.6, and for the ogive, with reference to the area of the cylindrical portion, was

approximately 2.1 .

.I 760

.T

inelastic buckling of initially imperfect cylindrical

shells subject to axial compression .

.A

lee,l.h.n.

.B

j. ae. scs. 29, 1962, 87.

.W

inelastic buckling of initially imperfect cylindrical

shells subject to axial compression .

an analytical and experimental study is made for inelastic

instability of initially imperfect cylindrical shells subject to

axial compression . donnell's equations and the principle of

virtual work are adapted to determine the effects of initial

imperfections on the buckling modes and the critical buckling

stresses . the deformation theory and the incremental theory

of plastic stress-strain relationships are both considered . the

experimental results of ten tests on specimens made of aluminum

alloy 3003-0 are presented . comparison of experimental with

theoretical results indicates that the application of the

deformation theory provides a fairly accurate prediction of buckling

strength, but fails in this case to yield a correct description of

post-buckling behavior . on the other hand, the application of

the incremental theory, which is mathematically and physically more rigorous, leads to an overestimation of buckling strength,

even though initial imperfections are considered . this paradox

has existed for years, and remains to be resolved .

.I 761

.T

buckling of sandwich under normal pressure .

.A

yao,j.c.

.B

j. ae. scs. 29, 1962, 264.

.W

buckling of sandwich under normal pressure .

a theoretical study is made of the buckling of a sandwich sphere

comprised of a core layer of low-modulus material and two thin

facing layers of higher modulus material . the solution for the

buckling resistance of the sphere under normal external pressure

is obtained by linearized theory, and is reducible to the classical

solution for monocoque spherical shells . critical buckling

pressures are calculated for various radius-thickness ratios and sphere

materials .

.I 762

.T

allowable axial loads and bending moments for inelastic structures

under nonuniform temperature distribution .

.A

b. e. gatewood and r. w. gehring

.B

the ohio state university and north american aviation, inc.

.W

allowable axial loads and bending moments for inelastic structures

under nonuniform temperature distribution .

a strain-analysis method is derived and demonstrated for the

calculation of design allowable load-strain curves for the cross section

of a structure supporting axial loads and bending moments . the

temperature effects of thermal stresses and changed material properties

and all inelastic effects are included in the calculations so that

the final curve is a design curve for the applied stresses as calculated

by room-temperature elastic procedures . the method allows for sequence

application and removal of load and temperature, as well as cycling

of load and/or temperature . applications are shown for a rectangular

bar under temperature cycling with axial loads and/or bending moments

and for a box beam with one bending-moment temperature cycle .

interaction curves beyween axial load and bending moment with inelastic

effects included are given, the calculations being done on a digital

computer . a procedure is given for using the method to construct

design curves .

.I 763

.T

effects of internal pressure on the buckling of circular-cylindrical

shells under bending.

.A

weingarten,v.i.

.B

.W

effects of internal pressure on the buckling of circular-cylindrical

shells under bending.

the effect of internal pressure on the small-deflection buckling of thin-walled cylinders under bending is investigated by means of a modified donnell equation . the results indicate that the maximum critical stress due to bending increases with internal pressure, unlike the case of pressurized cylinders under compression . these results represent the moment at which significant deformations appear in the cylinder, rather than the maximum moment able to be carried, but may be a good approximation to the latter for metal cylinders .

.I 764

.T

breathing vibrations of a circular shell with an internal

liquid .

.A

lindholm,u.s., kana,d.d. and abramson,h.n.

.B

j. ae. scs. 29, 1962, 1052.

.W

breathing vibrations of a circular shell with an internal

liquid .

resonant breathing frequencies and mode shapes are determined experimentally for a thin-walled, circular cylindrical shell containing a nonviscous incompressible liquid . the resonant frequencies determined for the full shell are in good agreement with those predicted by reissner's shallow-shell vibration theory

with the inclusion of an apparent-mass term for the liquid . the
effect of the internal liquid on the shell mode shapes is significant
only for the partially full shell . in this case the circumferential
node lines tend to shift toward the bottom or filled portion of the
shell .

excitation of low-frequency liquid-sloshing motion by
high-frequency forced oscillation of a partially filled shell occurred in
many cases . this low-frequency liquid response is tentatively
explained as being excited by a beat frequency in the forced
oscillation . a similar type of response has been reported by
yarymovych in axially excited rigid tanks .

.I 765

.T

clamped short oval cylindrical shells under hydrostatic
pressure .

.A

vafakos,w.p.

.B

j. ae.scs. 29, 1962, 1347.

.W

clamped short oval cylindrical shells under hydrostatic
pressure .

the principle of the minimum of the total potential is
employed to obtain stresses and displacements for clamped, short,
oval cylindrical shells under hydrostatic pressure . classical
shell theory, in which buckling effects are not considered, was
used . a fourier series is assumed for the deflections in the

closed circumferential direction so that the partial differential

equations of equilibrium are replaced by a set of ordinary

differential equations . the energy solution is compared with a

simplified approximation which can be considered an equivalent circular

cylinder solution . graphs of the significant stresses and

displacements are presented for oval cylinders having major to minor

axis ratios of 1.10, 1.30, and 1.50 . it is shown that the maximum

stresses and displacements increase significantly as the major to

minor axis ratio is increased .

.I 766

.T

experimental investigation at mach number of 3. 0 of

effects of thermal stress and buckling on flutter characteristics

of flat single-bay panels of length-width ratio 0. 96 .

.A

dixon,s.c.

.B

nasa tn.d1485, 1962.

.W

experimental investigation at mach number of 3. 0 of

effects of thermal stress and buckling on flutter characteristics

of flat single-bay panels of length-width ratio 0. 96 .

flat, single-bay, skin stiffener

panels with length-width ratios of 0.96

were tested at a mach number of 3.0,

at dynamic pressures ranging from 1,500 to

stagnation temperatures from 300 f to

effects of thermal stress and buckling on the flutter of such panels . the panel supporting structure allowed partial thermal expansion of the skins in both the longitudinal and lateral directions . panel skin material and skin thickness were varied .

a boundary faired through the experimental flutter points consisted of a flat-panel portion, a buckled-panel portion, and a transition point, at the intersection of the two boundaries, where a panel is most susceptible to flutter . the flutter region consisted of two fairly distinct sections, a large-amplitude flutter region and a small-amplitude flutter region . the results show that an increase in panel skin temperature flutter . the flutter trend for buckled panels is reversed . use of a modified temperature parameter, which approximately accounts for the effects of differential pressure and variations in panel skin material and skin thickness, reduced the scatter in the data which resulted when these effects were neglected . the results are compared with an exact theory for clamped panels for the condition of zero midplane stress . in addition, a

two-mode /transtability/ solution for clamped

panels is compared with the

experimentally determined transition point .

.I 767

.T

mathematical techniques applying to the thermal fatigue behaviour

of high temperature alloys .

.A

.B

.W

mathematical techniques applying to the thermal fatigue behaviour

of high temperature alloys .

during thermal fatigue testing

of a specimen with a thin edge, or

during rapid temperature changes in the gas

flow past turbine blades, the thin

edges are deformed plastically in compression

during heating and subsequently

creep in tension as the bulk of the specimen

or blade heats up . the plastic

deformation is determined from temperature

distributions, which are calculated

by biot's variational method . the creep

deformation is determined as a function

of time by a differential equation, which

expresses the balance between increasing elastic stress and reduction of stress due

to creep relaxation, and which is solved

to a riccati equation soluble in

terms of bessel functions, or (iii) by transformation

to a second-order differential

equation with a periodic coefficient .

using the thermal stresses obtained from

the solution of the differential equation, the theoretical thermal

fatigue endurance

is determined from cyclic (mechanical) stress

endurance data . agreement between

theoretical and experimental thermal fatigue

endurances is obtained, over ranges

of temperature, strain, and strain rate, or equivalently, over ranges

of temperature-edge radius and heat transfer coefficient .

this agreement supports the use of

the theoretical methods in wider contexts .

the accuracy of the temperature

distributions is better than the accuracy of

other factors entering into the correlation

between theoretical and experimental endurances .

improvement in the

interpretation of experimental results requires

consideration of the alteration

of the stress cycles during the course of thermal

fatigue testing . this requirement

is catered for partially by the various solutions of the differential equation for thermal stress .

.I 768

.T

formulae for use with the fatigue load meter in the

assessment of wing fatigue life .

.A

phillips,j.

.B

rae tn. struct.279, 1960.

.W

formulae for use with the fatigue load meter in the

assessment of wing fatigue life .

this note gives a method for the derivation of suitable constants

which, when multiplied by the readings recorded at each appropriate

acceleration level on a fatigue load meter and then added together, give

directly the proportion of fatigue life used up in the wing . it is

suggested that when the estimated proportion is of order 80, then a more

detailed assessment of fatigue life should be made .

.I 769

.T

local circumferential buckling of thin circular cylindrical

shells .

.A

johns,d.j.

.B

nasa tn.d1510, 1962, 267.

.W

local circumferential buckling of thin circular cylindrical

shells .

the problem of circumferential

buckling of a thin circular

cylindrical shell due to compressive

hoop stresses which vary in the axial

direction is examined . for

extremely localised compressive hoop stress

distributions resulting from

thermal discontinuity effects, or from a

uniform, radial line loading,

the buckle pattern should also be

localised . simplified analyses

into these two types of problem are

considered which show that only

a limited number of buckle deflection

modes needs to be assumed .

.I 770

.T

the flow of a compressible fluid past a sphere .

.A

caplan,c.

.B

naca tn.762, 1940.

.W

the flow of a compressible fluid past a sphere .

the flow of a compressible fluid past a sphere fixed in a uniform stream is calculated to the third order of

approximation by means of the janzen-rayleigh method .

the velocity and the pressure distributions over the

surface of the sphere are computed and the terms involving

the fourth power of the mach number, neglected in rayleigh's

calculation, are shown to be of considerable importance as

the local velocity of sound is approached on the sphere .

the critical mach number, that is, the value of the mach

number at which the maximum velocity of the fluid past the

sphere is just equal to the local velocity of sound, is

calculated for both the second and the third

approximations and is found to be, respectively, and .

.I 771

.T

on the flow of a compressible fluid past a sphere .

.A

tamada,k.

.B

proc. physico-math. soc japan, 21, 1939, 743.

.W

on the flow of a compressible fluid past a sphere .

it was shown by raleigh (philos. mag. 32, 1 (1916))

that the velocity potential for the subsonic flow of a

compressible fluid past a sphere can be expressed as a power

series in terms of mach's number m (which is the ratio of

the undisturbed velocity u, divided by the velocity of sound

for the undisturbed flow) . the equation in question is and boundary conditions are prescribed for

raleigh himself computed the first two terms of this series,.

the author finds the third term . he gives some graphs

showing numerical differences between raleigh's and his

approximation .

.I 772

.T

an experimental study of jet-flap compressor blades .

.A

clark, e.l. /jnr./ and ordway, d.e.

.B

j. aero. sc. november 1959. p. 698-702, 738 .

.W

an experimental study of jet-flap compressor blades .

the results of a preliminary experimental investigation to determine the

feasibility of using the jet flap to improve the section

characteristics of an axial-flow compressor blade are presented and discussed

trailing edge . internal design of the blade is described and details

of the resulting jet flow are given . also included are wind-tunnel

design and test procedures for the two-dimensional cascade used in the

test .

test results are presented in the form of the measured turning angle,

pressure rise, and lift coefficient . they are examined with particular

reference to the prevention of rotating stall .

.I 773

.T

q . app . math . 7, 1950, 381 . experiments on porous-wall cooling and flow separation control in a supersonic

nozzle .

.A

green,l. and nall,k.l.

.B

j. ae. scs. 1959, 689.

.W

q . app . math . 7, 1950, 381 . experiments on porous-wall

cooling and flow separation control in a supersonic

nozzle .

control of flow separation by fluid injection at one diverging boundary of a two-dimensional, transparent-walled de laval nozzle was investigated by spark schlieren photography of dry nitrogen flows expanded from two stagnation temperatures injection conditions at the permeable boundary were varied by the use of three grades of porous stainless steel with nominal pore diameters of 10, 20, and 30 microns, through which nitrogen was forced by coolant reservoir pressures of 25, 50, and 100 psig, in addition to the case of no forced injection . pressure distribution measurements were made along the nonpermeable diverging boundary . it was found that flow separation at expansion ratios approaching the optimum value for maximum thrust coefficient could be induced at the porous wall by a local injection mass velocity of the order of a few per cent of the local main-stream mass velocity . separation at the solid boundary was not noticeably influenced by injection at the opposite wall, and the asymmetrical separation thus effected jet deflections of up to 10 degrees at the lower stagnation-pressure levels . variation of the

wall heat-transfer condition by changing the stagnation

temperature did not significantly influence separation behavior .

temperature measurements at the reservoir face of the porous

section, together with use of published correlations and of the

rube-sin analysis for estimation of stream-side stanton numbers under

noninjection and injection conditions, respectively, permitted

heat-transfer calculations which indicated that the effectiveness

of the transpiration technique in controlling nozzle wall

temperatures derives primarily from intimate fluid-solid contact in a

porous material of high specific surface .

.I 774

.T

general characteristics of the flow through nozzles

at near critical speeds .

.A

sauer,r.

.B

naca tm.1147, 1944.

.W

general characteristics of the flow through nozzles

at near critical speeds .

the characteristics of the position and form of

the transition surface through the critical velocity

are computed for flow through flat and round nozzles

from subsonic to supersonic velocity . corresponding

considerations were carried out for the flow about profiles in the vicinity of sonic velocity .

.I 775

.T

studies on two dimensional flows of compressible fluid.

.A

.B

.W

studies on two dimensional flows of compressible fluid.

it is well-known that when the flow is everywhere subsonic in a field of flow, the nature of the two-dimensional isentropic flow of a compressible perfect fluid differs only slightly from that of the corresponding flow of an incompressible perfect fluid . thus, in such a case, we can calculate the field of flow by any of the well-known methods of approximation . on the other hand, if the flow is supersonic throughout the field, we can determine the flow pattern by the method of characteristics .

.I 776

.T

force measurements on square and dodecagonal sectional cylinders at high reynolds numbers .

.A

cowdrey,c.f. and lawes,j.a.

.B

npl. aero.351.

.W

force measurements on square and dodecagonal sectional cylinders at high reynolds numbers .

results are given of measurements in the compressed air tunnel of the forces on two cylinders, one of square cross-section and the other dodecagonal . the tests were carried out at various reynolds numbers ranging from approximately 0.1 x 10 to 5.5 x 10, based on the distance between parallel faces .

.I 777

.T

a technique for rendering approximate solutions to physical problems uniformly valid .

.A

lighthill,m.j.

.B

phil. mag. 40, 1949, 1179.

.W

a technique for rendering approximate solutions to physical problems uniformly valid .

a method is described for treating some of the characteristically non-linear problems of physics, in particular those involving a non-linear partial differential equation for which an approximate linearization is permissible everywhere except in a limited region, such as the neighbourhood of (5) a singular characteristic of the approximate solution, or of approximation is valueless . the

method involves a transformation of
an independent variable, which is
determined progressively with successive
approximations to the solution ..
only one step being necessary if a
first approximation valid uniformly
be obtained . the method is most
easily understood in its application
to simple first order ordinary
differential equations, which are
studied in detail in 2 and 3 as a
preparation for the extension to
more complicated problems in 4, 5
and 6 . physically, the longest
section, 6, concerns the /spread/ of
a progressive wave at infinity,
an important and essentially non-linear
process .

.I 778

.T

an integral related to the radiation integrals .

.A

powell,e.o.

.B

phil. mag. 34, 1943, 600.

.W

an integral related to the radiation integrals .

the author points out the relation of the integral

to problems in astrophysics and quantum mechanics, and

because of its importance supplies a table of values . the

table gives rl(x) to seven places of decimals from x=0 to

x=6.00 at intervals of 0.01 . second order central differences

are tabulated to assist in interpolation .

.I 779

.T

calculation and compilation of the unsteady lift functions for a rigid

wing subjected to sinusoidal gusts and to sinusoidal sinking

oscillations .

.A

drischler, j.a.

.B

naca tn 3748, october 1956

.W

calculation and compilation of the unsteady lift functions for a rigid

wing subjected to sinusoidal gusts and to sinusoidal sinking

oscillations .

the total lift responses of wings to sinusoidal gusts and to sinusoidal

vertical oscillations are calculated from the response to gust

penetration and to a sudden change in sinking velocity through use of the

well-established reciprocal relations for unsteady flow . the cases

considered are two-dimensional wings in incompressible, subsonic

compressible, sonic, and supersonic flow,. elliptical and rectangular wings

in incompressible flow,. wide rectangular and delta wings in supersonic

flow,. and delta wings of vanishingly low aspect ratio in incompressible and compressible flow . for most of the cases considered,

closed-form expressions are given and the final results are presented in

the form of plots of the square of the modulus of the lift coefficient

for wings in a sinusoidally oscillating gust and in the form of the real

and imaginary parts of the lift component for wings undergoing

sinusoidal sinking oscillations . a summary table is presented as a guide to

the scope and results of this paper,. this table contains the figure

and equation numbers for the types of flow and plan forms considered .

.I 780

.T

the transonic characteristics of 38 cambered rectangular

wings of varying aspect ratio and thickness as determined

by transonic-bump technique .

.A

nelson,w.h. and krumm,w.j.

.B

naca tn.3502, 1955.

.W

the transonic characteristics of 38 cambered rectangular

wings of varying aspect ratio and thickness as determined

by transonic-bump technique .

an investigation to determine

the effects of camber on the

aerodynamic characteristics of a series

of rectangular wings having various

aspect ratios and thickness-to-chord

ratios was conducted in the ames

the transonic-bump method .

the mach number range of the

investigation was from 0.6 to 1.12, with

a corresponding reynolds number

range of 1.7 to 2.2 million . the lift,

drag, and pitching-moment data

are presented for wings having aspect

ratios of 4, 3, 2, 1.5, and 1,

and naca 63a2xx and 63a4xx sections with

thickness-to-chord ratios of

.I 781

.T

use of subsonic kernel function in an influence-coefficient method of

aeroelastic analysis and some comparisons with experiment .

.A

john l. sewall, robert w. herr, and charles e. watkins

.B

technical note d-515

.W

use of subsonic kernel function in an influence-coefficient method of

aeroelastic analysis and some comparisons with experiment .

this paper illustrates the development and application of an

influence-coefficient method of analysis for calculating the response

of a flexible wing in an airstream to an oscillating disturbing force

and for treating such aeroelastic instabilities as flutter and

divergence . aerodynamic coefficients are derived on the basis of

lifting-surface theory for subsonic compressible flow by use of the method presented in nasa technical report r-48 .

application of the analysis is made to a uniform cantilever

wing-tip tank configuration for which responses to a sinusoidal disturbing

force and flutter speeds were measured over a range of subsonic mach

numbers and densities . calculated responses and flutter speeds based

on flexibility influence coefficients measured at nine stations are in

good agreement with experiment, provided the aerodynamic load is

distributed over the wing so that local centers of pressure very nearly

coincide with these nine influence stations . the use of experimental

values of bending and torsional structural damping coefficients in the

analysis generally improved the agreement between calculated and

experimental responses . some calculations were made to study the

effects of density on responses near the flutter conditions, and linear

response trends were obtained over a wide range of densities .

.I 782

.T

calculated subsonic span loads and resulting stability derivatives of

unswept and 45degree sweptback tail surfaces in sideslip and steady roll .

.A

queiijo, m.j. and riley, d.r.

.B

naca, tn 3245, 1954 .

.W

calculated subsonic span loads and resulting stability derivatives of

unswept and 45degree sweptback tail surfaces in sideslip and steady roll .

subsonic span loads and the resulting stability derivatives have been

calculated for a systematic series of vertical- and horizontal-tail combinations in sideslip and in steady roll in order to provide information embracing a wide range of probable tail configurations . all calculations were made by application of the discrete-horseshoe-vortex method to the problem of estimating loads on intersecting surfaces . the investigation covered variations in vertical-tail aspect ratio, the ratio of horizontal-tail aspect ratio to vertical-tail aspect ratio, the effects of horizontal-tail dihedral angle /for the sideslip case/, and the effects of vertical position of the horizontal tail for surfaces having their quarter-chord lines swept back 0degrees and 45degrees . the results of the investigation are presented in charts from which the span loads for the various conditions can be obtained . the resulting stability derivatives are presented as vertical- and horizontal-tail contributions as well as total-tail-assembly derivatives .

the results of this investigation, which was made for a wider range of geometric variables than previous studies, showed trends which were in general agreement with the results of previous investigations . also presented in this paper and used in the computations is an extensive table of values of sidewash due to a rectangular vortex .

.I 783

.T

a method for calculating the subsonic steady-state loading on an airplane with a wing of arbitrary planform and stiffness .

.A

gray, w.l.

.B

naca tn 3030, 1953 .

.W

a method for calculating the subsonic steady-state loading on an

airplane with a wing of arbitrary planform and stiffness .

a method for computing the steady-state span load distribution on an

elastic airplane wing for specified airplane weights and load factors is

given . the method is based on a modification of the weissinger

l-method and applies at subcritical mach numbers . it includes the effects

of external stores and fuselage on the spanwise loading .

modifications are outlined for treating tail-boom and tailless airplane

configurations and for calculating the divergence dynamic pressure of a

swept wing with a large external store . a method is also outlined for

reducing wind-tunnel data to obtain effective aerodynamic coefficients

which are free of model flexibility effects . the effects of mach number

can readily be evaluated from the aerodynamic coefficients thus

obtained .

.I 784

.T

heat transfer through the laminar boundary layer on

a circular cylinder in axial incompressible flow .

.A

bourne,d.e. and davies,d.r.

.B

q. j. mech. app. math. 11, 1958, 52.

.W

heat transfer through the laminar boundary layer on

a circular cylinder in axial incompressible flow .

this paper presents a method of calculating

the distribution of rate of heat transfer

into a laminar incompressible boundary layer
from the exterior surface of a long thin
circular cylinder, when the surface of the
cylinder is maintained at a constant
temperature and the flow is parallel to the
cylinder axis,. the temperature difference
between the surface and the main stream
is taken to be small enough to neglect
buoyancy effects . a series solution, valid
for small downstream distances from the
nose, has been obtained already by seban,
bond, and kelly . this is now extended
by deriving an asymptotic series solution,
valid at large downstream distances, and
bridging the gap between these two series
solutions by an approximate solution,
based on the method used recently by davies
and bourne to calculate heat transfer
from a flat plate . the calculation is used to
demonstrate the effect of curvature
and of prandtl number on the local rate of heat
transfer at various downstream
distances by comparing with the corresponding flat plate results .

.I 785

.T

the flow of fluid along cylinders .

.A

cooke, j.c.

.B

quart. j. mech. app. math, vol.x /3/, 1957, p. 312-321 .

.W

the flow of fluid along cylinders .

the boundary layer equations for uniform flow parallel to the generators of any cylinder without corners are put into the form of a series of linear third-order differential equations . the first three of these are the same as those obtained by seban and bond /1/ for a circular cylinder and solved by kelly /2/ . the rest have additional terms depending on the radius of curvature of the cylinder and its derivatives . the problem is also attacked by a pohlhausen method as far as four terms of the series . for large distances from the front, rayleigh's method, as given by hasimoto /3/, gives the first two terms of an asymptotic expansion for the drag . explicit calculations are made of the drag of an elliptic cylinder of eccentricity 1/2 3 . there is evidence that the drag is everywhere less than that of a circular cylinder of the same perimeter .

.I 786

.T

the skin friction on infinite cylinders moving parallell to their length .

.A

batchelor, g.k.

.B

quart. j. mech. app. math. vol. vii, /2/, 1954, p. 179-192 .

.W

the skin friction on infinite cylinders moving parallell to their length .

the frictional force on a cylinder moving steadily parallel to its length through a viscous liquid which is initially at rest is determined with reasonable accuracy over the whole range of values of the duration of the motion and for a wide variety of shapes of the cylinder cross-section . when the time t is small, the first approximation gives a force per unit area which is the same as that for a flat plate of infinite width . the second approximation takes the shape of the cylinder into account and the force on unit length of cylinder is determined in terms of the number of corners, and their angles, in the cylinder cross linder is the same, to this approximation, as that on a circular cylinder of the same perimeter . for large values of t the determination of the frictional force is reducible to that of a potential problem, the solution of which is known for a number of different shapes . the approximations for small and large values of t for any one cylinder do not overlap but can be joined without much ambiguity . for no value of t do the forces on cylinders of different shape /excluding those whose curvature is not everywhere inwards/ differ by more than about 25 per cent .

.I 787

.T

rayleigh's problem for a cylinder of arbitrary shape .

.A

hasimoto, h.

.B

j. physical soc. of japan, vol. 9 /4/, 1954, p. 611-619 .

.W

rayleigh's problem for a cylinder of arbitrary shape .

the motion of an incompressible viscous fluid generated by a cylinder of arbitrary cross-sectional form which is started to move suddenly from rest with uniform velocity in the direction of its length is considered formulae in powers of are derived for the velocity distribution /valid in the vicinity of the cylinder/ and for the frictional drag on the cylinder, correct to the order of a, where a is the characteristic length of the cross section, v is the kinematic viscosity, and t is the time . these formulae are given in terms of only the analytic function which maps conformally the region outside the cross section of the cylinder onto the region outside the unit circle, and of certain integrals e which are common to any arbitrary cylinder . in particular, when a is sufficiently small, the total frictional drag on the cylinder per unit length is expressed as, irrespective of the cross-sectional form, where b 2 and y 0.5772.../euler's constant/ .

.I 788

.T

an approximate boundary layer theory for semi-infinite cylinders of arbitrary cross-section .

.A

varley, e.

.B

j. fluid mech. vol. 3. 1958, p. 601-614 .

.W

an approximate boundary layer theory for semi-infinite cylinders of arbitrary cross-section .

an estimate is given of the distribution of skin frictional force per unit length, and of displacement area, on the outside of a semi-infinite cylinder, of arbitrary cross-section, moving steadily in a direction parallel to its generators . a pohlhausen method is employed with a velocity distribution chosen to yield zero viscous retarding force on the boundary layer approximations . /the smallness of the fluid acceleration far from the leading edge has been pointed out by batchelor reasonable results atlarge distances from the leading edge . however, for a large class of cross-sections, which includes all convex cross-sections and locally concave cross-sections with re-entrant angles greater than 1/2, the method yields the expected square root growth of the boundary layer at the leading-edge, with a fairly close approximation to the coefficient, and it is supposed that the skin-frictional force and displacement area are given with reasonable accuracy along the whole length of the cylinder .

results for the elliptic cylinder and the finite flat plate are given in closed form, valid for the whole length of the cylinder, and are expected to be in error by at most 20 per cent . in addition, some estimate is given of the effect of corners on skin frictional force and displacement area .

.I 789

.T

a further note on the calculation of heat transfer

through the axisymmetrical laminar boundary layer on

a circular cylinder .

.A

bourne,d.e., davies,d.r. and wardle,s..

.B

q. j. mech. app. math. 12, 1959, 257.

.W

a further note on the calculation of heat transfer

through the axisymmetrical laminar boundary layer on

a circular cylinder .

by using a karman-pohlhausen method

the distribution of local rate of heat

transfer is ovaluated for the case of air flow

in an axisymmetrical laminar boundary

layer on a heated circular cylinder, the temperature

of the cylinder being independent

of downstream distance . this calculation

serves to link the numerical values

obtained by seban, bond, and kelly

for small downstream distances to those

obtained by bourne and davies for large

downstream distances .

.I 790

.T

a wind-tunnel test technique for measuring the dynamic rotary

stability derivatives at subsonic and supersonic speeds .

.A

report 1258

.B

benjamin h. beam

.W

a wind-tunnel test technique for measuring the dynamic rotary

stability derivatives at subsonic and supersonic speeds .

a method is described for measuring the dynamic stability derivatives of a model airplane in a wind tunnel . the characteristic features of this system are that single-degree-of-freedom oscillations were used to obtain combinations of rolling, yawing and pitching motions., that the oscillations were excited and controlled by velocity feedback which permitted operation under conditions unfavorable for more conventional types of oscillatory testing., and that data processing was greatly simplified by using analog computer elements in the strain-gage circuitry .

the system described is primarily for measurement of the damping derivatives damping

in roll damping in pitch, damping in yaw, and the cross derivatives rolling moment due to yawing and yawing moment due to rolling . the method of testing also permits measurement under oscillatory conditions of the static derivatives rolling moment due to sideslip, yawing moment due to sideslip, and pitching moment due to angle of attack . all these derivatives are of particular importance in estimating the short-period oscillatory motions of a rigid airplane .

a small number of experimental data are included to illustrate the general scope of results obtainable with this system .

.I 791

.T

measurements at mach numbers up to 2. 8 of the longitudinal characteristics of one plane and three cambered slender 'ogee' wings .

.A

taylor,c.r.

.B

rae r. aero.2658, 1961.

.W

measurements at mach numbers up to 2. 8 of the longitudinal

characteristics of one plane and three cambered slender

'ogee' wings .

measurements have been made of

the longitudinal characteristics of one

plane and three cambered slender ogee

wings (p = 0.45,) at two

subsonic and eight supersonic mach

numbers up to 2.8 . the tests also included

measurements of the zero-lift pressure

drag and support interference of the

plane wing . the results have been analysed

to give data for estimating the

performance of supersonic transport aircraft .

.I 792

.T

some low speed problems of high speed aircraft .

.A

spence, a. and clean, d.

.B

j. royal aero. soc. v. 66, april 1962, pp 211-225 .

.W

some low speed problems of high speed aircraft .

the first part of the paper deals with the low speed aerodynamics of

aircraft shapes suggested by kuchemann, at the second international congress in aeronautical sciences at zurich in 1960, as suitable for achieving a required range at supersonic speeds, namely wingbody arrangements with sweepback angles of 55degrees or 60degrees and streamwise thickness-chord ratio of about 5 per cent suitable for low supersonic speed, and slender near-triangular wings with sharp leading edges suitable for mach numbers of about 2 or more .
no attention is given to /slewed/ wings, powered lift or variable geometry .
in dealing briefly with swept wings, the need for avoiding separation of flow from the leading edge is demonstrated, with the conclusion that it is desirable to use leading edge flaps with blowing or suction at the knee together with blown trailing edge flaps . wind tunnel tests are described on a simplified model with these boundary layer control methods applied . mention is made of the possibility of adverse ground effect on maximum lift .
more attention is given to the case of slender wings because their use involves a new type of flow with separation from all edges . this flow and its steadiness are therefore discussed from the point of view of the possibility of buffeting,. the effect of plan form on static longitudinal stability and pitch-up is analysed,. and a short summary of available results on damping in pitch is given . large rolling moments due to sideslip are shown to give rise to serious problems of control, and the present state of knowledge of static lateral and directional stability and rolling and yawing rotary derivatives is discussed .
finally the effects of proximity to the ground are summarised .
the second part of the paper is concerned with work aimed at clarifying

some of the requirements for handling qualities of future aircraft . it
is not so much concerned with forecasts of the dynamic behaviour of
these future aircraft as with determining what the pilot wants .
two aspects of control in the vertical plane are discussed in some
detail namely speed control and glide path holding . flight tests on an
avro 707a aircraft, with artificially worsened characteristics, are
described, and it is shown that substantially constant performance in the
piloting task can be achieved at the expense of increased pilot effort .
some tentative conclusions on desirable levels of speed stability and
phugoid damping are, nevertheless, drawn .
a brief review of the present status of lateral/directional handling
requirements, using mainly american data, is also included .

.I 793

.T

the flow pattern on a tapered sweptback wing at mach numbers between 0.6
and 1.6 .

.A

hall, i.m. and rogers, e.w.e.

.B

a.r.c. r + m 3271, part 1, july 1960 .

.W

the flow pattern on a tapered sweptback wing at mach numbers between 0.6
and 1.6 .

the development of the flow pattern on a swept wing with incidence and
stream mach number is described . the wing, of aspect ratio 2 828,
taper ratio 0 333 and leading-edge sweep 53 5 deg, was tested at mach
numbers between 0 6 and 1 6 at incidences up to about 12 deg . the test

reynolds number varied with mach number, being typically 2 3 x 10 at m

leading edge .

it is shown that the flow pattern at moderate incidences develops smoothly from a subsonic type involving leading-edge separation to a supersonic type where the flow is attached near the leading edge and with shock-induced separation further aft . the formation and movement of the shock-wave system and the vortices near the wing surface are briefly discussed .

.I 794

.T

experiments with a tapered swept-back wing of warren 12 planform at mach numbers between 0.6 and 1.6 .

.A

hall, i.m. and rogers, e.w.e.

.B

a.r.c. r + m 3271, part ii, july 1960 .

.W

experiments with a tapered swept-back wing of warren 12 planform at mach numbers between 0.6 and 1.6 .

the development of the flow pattern on a wing of aspect ratio 2 828, taper ratio 0 333, leading-edge sweepback 53 5 deg and 6 per cent thickness/chord ratio in the streamwise direction has been described in part 1, which discussed oil-flow patterns obtained on the surface of the wing . the complete programme of tests also included pressure plotting at four spanwise stations and force measurements . these are discussed in relation to the flow development in this part of the report .

the wing was tested at mach numbers between 0 6 and 1 6 for incidences up to about 14 deg . the tunnel stagnation pressure was held constant at a value near atmospheric pressure during the tests, so that the reynolds number varied with mach number .. at m 1 0 it was 2 3 x 10 based on the mean aerodynamic chord . boundary-layer transition was fixed by a roughness band at the leading edge .

a detailed analysis has been made of the pressure distributions on the surface of the wing and the chordwise distributions integrated to determine the spanwise loading . the overall lift and pitching moment of the wing were also obtained from these data, as well as from direct measurements using a strain-gauge balance, by means of which the wing drag was also determined . these results are considered in some detail to illustrate the effects of mach number and incidence on the flow about the model . a preliminary analysis is also made of the conditions for boundary-layer separation due to shock waves on the wing surface . the principal factor appears to be the component of mach number normal to the shock front .

.I 795

.T

the operation of the npl 18in x 14in. wind tunnel in the transonic speed range .

.A

hall, i.m.

.B

a.r.c. c.p. 338, january 1957 .

.W

the operation of the npl 18in x 14in. wind tunnel in the transonic speed range .

a brief description of the slotted liners used is given together with

the power requirements and some flow surveys .

some observations are made on wall interference on a half-model of a

swept wing tested in the wind tunnel .

.I 796

.T

an investigation at transonic speeds of the performance of various

distributed roughness bands used to cause boundary layer transition

near the leading edge of a cropped delta half-wing .

.A

rogers, e.w.e. and hall, i.m.

.B

a.r.c., c.p. 481, may 1959 .

.W

an investigation at transonic speeds of the performance of various

distributed roughness bands used to cause boundary layer transition

near the leading edge of a cropped delta half-wing .

distributed roughness bands of no.320 and no.500 carborundum were found

to be effective in causing boundary-layer transition if they extended

over the first 5(and 10(respectively of the local chord . use of

larger grain sizes, or increases in the band width for a given grain

size resulted in a drag penalty . with very large particle sizes /about

between the particles . the drag penalty was constant over the test mach

number range /0.80 to 1.15/ and decreased slowly with incidence . the

wing lift and pitching moment were only slightly modified by the

presence of any of the roughness bands tested, but this result would not

of course necessarily apply to wings of other planforms or section shapes . the test reynolds number was about 2.7 million .

in the appendix, the structure of the roughness bands is discussed, as

well as the details of the materials used and the techniques used to

apply the band .

.I 797

.T

a study of the effect of leading-edge modifications on the flow over

a 50degree sweptback wing at transonic speeds .

.A

rogers, e.w e., townsend, j.e.g. and berry, c.j.

.B

a.r.c., r + m 3270, may 1960 .

.W

a study of the effect of leading-edge modifications on the flow over

a 50degree sweptback wing at transonic speeds .

summary . an investigation has been made in the n.p.l. 18 in. x 14 in .

tunnel of the effects of leading-edge modifications on the flow and

forces on an untapered wing of 50 deg leading-edge sweep, at stream mach

numbers between 0 60 and 1 20 . seven leading-edge profiles were

tested, ranging from a drooped extension of 18 per cent of the chord of

the basic sharp-nosed section to a round-nosed section with a

leading-edge radius of 1 0 per cent of the basic chord .

leading-edge droop was found to increase the wing drag near zero lift

but to reduce appreciably the lift-dependent drag component, except at

the highest test mach numbers . droop also increased the lift

coefficient at which leading-edge separation occurred on the upper surface

at moderate subsonic speeds, but in addition reduced the mach number for transonic flow attachment . the appearance of the forward shock /but not the rear shock/ is considerably delayed when the leading edge is drooped .

with the undrooped sections an increase in leading-edge radius was accompanied by successively earlier appearances of the forward shock, and hence the outboard shock with its attendant separation . the conditions at which the rear shock first appeared changed only slowly as the section was changed .

the variations in wing flow pattern as the leading edge is modified are discussed and related to measured changes in the wing lift and drag .

an attempt is also made to estimate the local mach numbers on some parts of the wing from the oil-flow patterns,. this material is used to assess the flow conditions appropriate to shock-induced separation .

the main section of the report concludes with a tentative discussion of the significance of the present results to the design of swept wings .

in an appendix results obtained with the wing in a sweptforward configuration are briefly considered .

.I 798

.T

interaction between shock waves and boundary layers, with a note on the effects of the interaction of the performance of supersonic intakes .

.A

holder, d.w., pearcey, h.h. and gadd, g.e.

.B

a.r.c., c.p. 180, february 1954 .

.W

interaction between shock waves and boundary layers, with a note on the effects of the interaction of the performance of supersonic intakes .

the interaction between shock waves and boundary layers has important effects in many problems of high-speed flow . this paper has been written as a guide to the literature on the subject, and as a critical review of the present state of knowledge concerning both the underlying physical processes and the practical applications . it will be clear to the reader that, although substantial progress has been made, our knowledge is still far from complete and that more work both of a fundamental nature and on specific applications is needed before the problem is understood sufficiently well for design purposes .

part i of the paper describes experiments on comparatively simple types of flow designed to provide fundamental information and to assist in the development of the theory . these experiments show that the interaction depends mainly on the mach and reynolds numbers and on the strength of the shock wave . in particular, the interaction of a shock wave with a laminar boundary layer is shown to produce much larger effects than if the boundary layer is turbulent . for most cases where the effects of the interaction are large enough to have serious practical consequences it is found that the boundary layer separates from the surface, and the difference between the interaction with laminar and turbulent layers arises mainly because the laminar layer separates much more readily in an adverse pressure gradient . the details of the interaction downstream of the separation point thus depend critically on the behaviour of the separated layer, and on the conditions under which it reattaches to the surface .

many of the features found in the fundamental experiments appear also in practical applications and these are considered in parts ii and iii of the paper . although the emphasis hero is on the performance of aerfoils and wings moving at high subsonic speeds, the importance of the interaction in other examples such as at supersonic trailing edges and in supersonic intakes is also discussed briefly . the differences between the interaction with laminar and turbulent boundary layers are often a source of serious discrepancy between model experiments and full-scale conditions . for small-scale models it is, therefore, frequently essential to make the boundary layer turbulent by artificial means . some of the difficulties involved in doing this, and certain of the more promising methods are briefly discussed . it is shown that experiments on models with transition fixed can be used to explain a number of aerodynamic effects encountered in transonic flight, and connected with the occurrence of shock-induced separation of the turbulent boundary layers . for both two-dimensional aerofoils and straight and sweptback wings, turbulent separation occurs for shocks above a certain strength which applies for both model and full-scale conditions full-scale conditions,. differences in magnitude would be expected if the pressure recovery along the separated layer between the shock and the trailing edge is affected by reynolds number, but little information is at present available on this point .

most of the repercussions of turbulent separation on the steady-motion characteristics of aerofoils and wings can be traced to the associated reduction in the pressure recovery over the roar of the surface . this is because the pressure at the trailing edge controls the inter-relation between the two surfaces /so long as the flow at the trailing edge remains subsonic/, and in particular the relative movements of the shock waves and the extents of the local regions of supersonic flow . certain unsteady-flow characteristics such as buffeting and control surface separation .

some evidence is presented on the influence of section shape on the occurrence and effects of separation, but in this, as in many other respects, information relevant to turbulent boundary layers is scarce .

some notes on the further work which is required are given in part iv of the paper .

.I 799

.T

some effects of wind-tunnel interference observed in tests on two-dimensional aerofoils at high subsonic and transonic speeds .

.A

pearcey, h.h., sinnott, c.s. and osborne, j.

.B

n.p.l. aero. 373. february 1959 .

.W

some effects of wind-tunnel interference observed in tests on two-dimensional aerofoils at high subsonic and transonic speeds .

in the high-speed research on two-dimensional aerofoils at the national physical laboratory the need to keep model size above a certain minimum, in order to reproduce correctly the boundary layer separation effects experienced at full scale, has been considered paramount even at the risk of incurring significant tunnel interference effects .

this report discusses the interference effects for the slotted working sections now in use . the magnitudes of the blockage and lift effect corrections are deduced for the ratio of model chord to tunnel height normally used . it is shown that a simple adaptation to reduce the open area of the walls would reduce both corrections to insignificant

proportions simultaneously, but would give a reduced choking mach number

separated flows . the observed trends in the variation of the blockage

effects for other ratios of model chord to tunnel height differ from

those predicted theoretically, and so the results cannot be applied more

generally until these trends have been checked by further

investigations .

it is suggested that wake interference effects can significantly

influence the manner in which separated flows develop with increasing

incidence or mach number, particularly for walls of small open area .

examples are also given of effects of distortions in the local

supersonic flow, which are most noticeable for walls with relatively large

open areas .

.I 800

.T

wall interference at transonic speeds on a hemisphere cylinder model .

.A

rogers, e.w.e. and hall, i.m.

.B

a.r.c., c.p. 510, september 1959 .

.W

wall interference at transonic speeds on a hemisphere cylinder model .

tests have been made in three n.p.l. wind tunnels on a pressure-plotting

model consisting of a long cylinder with a hemispherical nose . the

surface pressure distributions were measured for stream mach numbers

between 0.7 and 1.1 at zero model incidence, and schlieren photographs

were taken . the blockage ratios were 0.211(, 0.117(and 0.120(.

the principal feature of the flow is the effect of working section size on the rate at which the terminal shock wave moves back along the model with increasing stream mach number . this is thought to depend mainly on the distance from the model to the slotted walls of the tunnel, and not necessarily on the blockage ratio . the distance of the solid sidewall is important in influencing the local mach number ahead of the terminal shock, by reflecting the expansion-wave system originating near the model nose .

.I 801

.T

experimental study of the equivalence of transonic flow about slender cone-cylinders of circular and elliptic cross section .

.A

page,w.a.

.B

naca tn.4233, 1958.

.W

experimental study of the equivalence of transonic flow about slender cone-cylinders of circular and elliptic cross section .

this report describes an experimental investigation of the equivalence relationship and the related theory for lifting forces proposed by transonic slender-body theory . the models chosen for this study are a flat, winglike, elliptic cone-cylinder and its equivalent body of revolution, a circular cone-cylinder . it is determined that the flows about the two models are closely related in the manner predicted by the theory, the relationship persisting over a mach number range of 0.92 to

cone-cylinder vary linearly only over the small angle-of-attack range of

approximately 1 and that the aerodynamic loading at sonic speed compares

favorably with jones' slender-wing theory .

the results of the investigation suggest that at transonic speeds and

at small angles of attack the calculation of all aerodynamic

characteristics of slender, three-dimensional shapes can be made by use

of transonic slender-body theory when the pressures on the equivalent

body of revolution are known, either by experiment, or by an adequate

nonlinear theory . from transonic slender-body theory it is deduced

that the slenderness required for this application is the same as that

required for the successful application of the transonic area rule .

.I 802

.T

the behaviour of supersonic flow past a body of revolution

far from the axis .

.A

whitham,g.b.

.B

proc. roy. soc. a, 201, 1950, 89.

.W

the behaviour of supersonic flow past a body of revolution

far from the axis .

a theory is developed of the supersonic flow past a body of revolution

at large distances from the axis, where a linearized approximation is

valueless owing to the divergence of the characteristics at infinity .

it is used to find the asymptotic forms of the equations of the shocks which are formed from the neighbourhoods of the nose and tail . in the special case of a slender pointed body, the general theory at large distances is used to modify the linearized approximation to give a theory which is uniformly valid at all distances from the axis . the results which are of physical importance are summarized in the conclusion (9) and compared with the results of experimental observations .

.I 803

.T

the shock pattern of a wing-body combination far from the flight path .

.A

walkden,f.

.B

aero. quart. 9, 1958, 164.

.W

the shock pattern of a wing-body combination far from the flight path .

the position and strength of the front shock wave at large distances from a wing-body combination, are deduced from the linear theory for the combination, using a method developed by whitham . the combination consists of a body of revolution and a wing which has thickness and is lifting . the effects of interference between the flow over the body and the flow over the wing are included . in any direction the flow far from the wing-body combination is equivalent to the flow past a body of revolution determined from the configuration of the combination . the modified formulae for unsteady flow are given and some results are evaluated for the combination of a body of revolution

and a delta wing with subsonic leading edges .

.I 804

.T

a flight test investigation of the sonic boom .

.A

mullens,m.e.

.B

afftc-tn-56-20, air res. and dev. command, u.s.a.f., 1956.

.W

a flight test investigation of the sonic boom .

the /sonic boom/ as it is now popularly called, has become the

center of considerable interest during the past few years because of

widespread public disturbance and possible damage that can result

from it . in the hopes of minimizing this disturbance and to extend

the general knowledge of the shock waves which produce the booming

noise, the aeronautical research laboratory, wright air development

center, has initiated an extensive research program to study the sonic

boom phenomenon .

this report presents the results of flight tests undertaken as one

phase of this program . the tests had as their objective the

determination and measurement of the shock wave pressure pattern surrounding

an f-100 aircraft in level supersonic flight .

the flight tests were conducted at the air force flight test

center, edwards air force base, california, under the authority of air

research and development command test directive no. 5524-f1 .

.I 805

.T

ground measurements of the shock wave noise from airplanes

in level flight at mach numbers to 1. 4 and at altitudes

to 45,000 feet .

.A

maglieri,d.j., hubbard,h.h. and lansing,d.l.

.B

nasa tn.d48, 1959.

.W

ground measurements of the shock wave noise from airplanes

in level flight at mach numbers to 1. 4 and at altitudes

to 45,000 feet .

time histories of noise pressures near ground level were measured during flight tests of fighter-type airplanes over fairly flat, partly wooded terrain in the mach number range between 1.13 and 1.4 and at altitudes from 25,000 to 45,000 feet . atmospheric soundings and radar-tracking studies were made for correlation with the measured noise data .

the measured and calculated values of the pressure rise across the shock wave were generally in good agreement . there is a tendency for the theory to overestimate the pressure at locations remote from the track and to underestimate the pressures for conditions of high tailwind at altitude . the measured values of ground-reflection factor averaged about 1.8 for the surfaces tested as compared to a theoretical value of 2.0 . two booms were measured in all cases . the observers also generally reported two booms,. although, in some cases, only one boom was reported . the shock-wave noise associated with some of the flight tests was judged to be objectionable by ground observers, and in one

case the cracking of a plate-glass store window was correlated in time

with the passage of the airplane at an altitude of 25,000 feet .

.I 806

.T

ground measurements of airplane shock wave noise at

mach numbers to 2, and at altitudes of 60,000 feet .

.A

lina,l.j. and maglieri,d.j.

.B

nasa tn.d235, 1960.

.W

ground measurements of airplane shock wave noise at

mach numbers to 2, and at altitudes of 60,000 feet .

the intensity of shock-wave noise at the ground resulting from

flights at mach numbers to 2.0 and altitudes to 60,000 feet was

measured . measurements near the ground track for flights of a

supersonic fighter and one flight of a supersonic bomber are

presented .

level cruising flight at an altitude of 60,000 feet and a mach

number of 2.0 produced sonic booms which were considered to be

tolerable, and it is reasonable to expect that cruising flight at higher

altitudes will produce booms of tolerable intensity for airplanes of the

size and weight of the test airplanes . the measured variation of

sonic-boom intensity with altitude was in good agreement with the

variation calculated by an equation given in nasa technical note d-48 .

the effect of mach number on the ground overpressure is small between

mach numbers of 1.4 and 2.0, a result in agreement with the theory . no
amplification of the shock-wave overpressures due to refraction effects
was apparent near the cutoff mach number .

a method for estimating the effect of flight-path angle on cutoff
mach number is shown . experimental results indicate agreement with the
method, since a climb maneuver produced booms of a much decreased
intensity as compared with the intensity of those measured in level flight at
about the same altitude and mach number .

comparison of sound pressure levels for the fighter and bomber
airplanes indicated little effect of either airplane size or weight at an
altitude of 40,000 feet .

.I 807

.T

ground measurements of the shock wave noise from
supersonic bomber airplanes in the altitude range from 30,000
to 50,000 feet .

.A

maglieri,d.j. and hubbard,h.h.

.B

nasa tn.d880, 1961.

.W

ground measurements of the shock wave noise from
supersonic bomber airplanes in the altitude range from 30,000
to 50,000 feet .

shock-wave ground-pressure measurements
have been made for
supersonic bomber airplanes in the mach number

range from 1.24 to 1.52, for

altitudes from about 30,000 to 50,000 feet,

and for a gross-weight range

from about 83,000 to 120,000 pounds . the

measured overpressures were

generally higher than would be predicted

by the theory which accounts

only for volume effects . there is thus

a suggestion that lift effects

on sonic-boom intensity may be significant

for this type of airplane

for the altitude range of the present tests .

.I 808

.T

an investigation of some aspects of the sonic boom

by means of wind tunnel measurements of pressures about

several bodies at a mach number of 2. 01 .

.A

carlson,h.w.

.B

nasa tn.d161, 1959.

.W

an investigation of some aspects of the sonic boom

by means of wind tunnel measurements of pressures about

several bodies at a mach number of 2. 01 .

an investigation of some aspects

of the sonic boom has been made

with the aid of wind-tunnel measurements

of the pressure distributions

about bodies of various shapes . the

tests were made in the langley

at a mach number of 2.01 and

at a reynolds number per foot of 2.5 x 10 .

measurements of the pressure

field were made at orifices in the surface

of a boundary-layer bypass

plate . the models which represented both

fuselage and wing types of

thickness distributions were small enough

to allow measurements as far

away as 8 body lengths or 64 chords . the

results are compared with

estimates made using existing theory .

to the first order, the boom-producing

pressure rise across the bow

shock is dependent on the longitudinal

development of body area and not

on local details . nonaxisymmetrical shapes

may be replaced by

equivalent bodies of revolution to obtain satisfactory

theoretical estimates

of the far-field pressures .

.I 809

.T

an investigation of the influence of lift on sonic-boom intensity by means of wind tunnel measurements of the pressure fields of several wing-body combinations at a mach number of 2. 01 .

.A

carlson,h.w.

.B

nasa tn.d881, 1961.

.W

an investigation of the influence of lift on sonic-boom intensity by means of wind tunnel measurements of the pressure fields of several wing-body combinations at a mach number of 2. 01 .

an investigation of the effect of lift on sonic-boom intensity has been performed by means of wind-tunnel measurements of the pressure fields surrounding small wing-body combinations . the tests were conducted in the langley 4- by 4-foot supersonic pressure tunnel at a mach number of per foot . effects of lift were found to be real and significant . measured bow-shock intensities agreed fairly well with, but were consistently less than, shock intensities estimated by theoretical methods .

available flight data were examined

for correlation with wind-tunnel test

results .

.I 810

.T

the shock wave noise problem of supersonic aircraft

in steady flight .

.A

maglieri,d.j. and carlson,h.w.

.B

nasa memo 3-4-59l, 1959.

.W

the shock wave noise problem of supersonic aircraft

in steady flight .

data are presented which provide

an insight into the nature of the

shock-wave noise problem, the significant

variables involved, and the

manner in which airplane operation

may be affected . flight-test data

are also given, and a comparison with

the available theory is made . an

attempt is also made to correlate the

subjective reactions of observers

and some associated physical phenomena

with the pressure amplitudes

during full-scale flight .

it is indicated that for the proposed
supersonic transport airplanes
of the future, booms on the ground will
most probably be experienced
during the major portion of the flight
plan . the boom pressures will be
most severe during the climb and descent
phases of the flight plan .
during the cruise phase of the flight,
the boom pressures are of much
lesser intensity but are spread laterally
for many miles . the manner
in which the airplane is operated appears
to be significant,. for example,
the boom pressures during the climb,
cruise, and descent phases can be
minimized by operating the airplane at
its maximum altitude consistent
with its performance capabilities .

.I 811

.T

an investigation of lifting effects on the intensity
of sonic booms .

.A

morris,j.

.B

j. roy. aero. soc. 64, 1960, 610.

.W

an investigation of lifting effects on the intensity of sonic booms .

this paper is a brief summary of an investigation made to check the effect of lift on the shock noise of aircraft flying at supersonic speeds . the method of hayes has been combined with the theory of whitham to predict the asymptotic shock strength of wings carrying lift and of combinations of bodies and lifting wings . (a similar, but not quite as general, method was derived by walkden in ref. 6 .) whitham's formula, including only the volume term, has been used extensively to predict the boom intensity of aeroplane type bodies and the agreement with experiment has, so far, been quite reasonable . the test data obtained to date extends only up to about 40,000 ft. altitude and the calculations of this paper show that under those conditions the shock noise of the aircraft tested so far will, in most cases, be dominated by the volume term . it is shown that at higher altitudes lifting effects will dominate for even the small fighter and they will dominate over most of the altitude range for large bomber and supersonic transport aircraft . the boom intensity due to lift decreases with altitude as which compares to

in the volume case (=pressure at

altitude h) . it is insensitive to mach number,

wing loading, wing plan shape and

lift distribution . a simple rule for calculating

the shock noise due to combined volume and

lifting effects is proposed which is applicable to

configurations with wings located towards the

rear of the fuselage . the rule states that the shock

noise of an aircraft carrying lift is equal

to the shock noise due to volume (neglecting lift)

or the shock noise due to lift (neglecting

volume), whichever is the greater . a chart is

presented from which rapid estimates can be

made of the shock noise of lifting

wing-body combinations .

.I 812

.T

investigation of two bluff shapes in axial free flight

over a mach number range from 0. 35 to 2. 15 .

.A

coltrane,l.c.

.B

naca rm l58a16, 1958.

.W

investigation of two bluff shapes in axial free flight

over a mach number range from 0. 35 to 2. 15 .

a fineness-ratio-2.71 right circular cylinder and a fineness-
ratio-been tested in free flight over a mach number range of 0.35 to 2.15 and
a reynolds number range of 1 x 10 to 12 x 10 . time histories, cross
plots of force coefficients, rolling velocity, and longitudinal-force
coefficient are presented for both cylinders . in addition, cross plots
of moment coefficients and plots of the normal-force curve slope and
the aerodynamic center are presented for the fineness-ratio-2.71
cylinder . the average aerodynamic center of the right circular
cylinder moved rearward with decreasing speeds until at the subsonic
mach numbers it remained approximately constant and comparisons of the
drag data of this test with wind tunnel and other free-flight data show
good agreement . an appreciable decrease in drag was observed when the
data of the present test of the rounded nose cylinder were compared with
data of a right circular cylinder of a similar configuration .

.I 813

.T

the motion of rolling symmetrical missiles referred
to a body-axis system .

.A

nelson,r.l.

.B

naca tn.3737, 1956.

.W

the motion of rolling symmetrical missiles referred
to a body-axis system .

the linearized equations of motion have been derived for a rolling
missile having slight aerodynamic asymmetries . time histories of

rolling-missile motions referred to a body-axis system have been

prepared to show the types of missile motions that can be encountered .

the motions resulting from a trim change and a pulse-rocket disturbance

are shown to be determined mainly by the ratio of rolling velocity to

pitching frequency .

finally, the derived equations are used in establishing a technique

for the reduction of rolling-missile oscillation data . it is shown

that the aerodynamic derivatives can be obtained from flight data if

four accelerations are measured . the method is applied to the results

obtained from a flight test of a missile configuration .

.I 814

.T

stability derivatives of cones at supersonic speeds .

.A

tobak,m. and wehrend,w.r.

.B

naca tn.3788, 1956.

.W

stability derivatives of cones at supersonic speeds .

the aerodynamic stability derivatives due to pitching velocity and

vertical acceleration are calculated by use of potential theory for

circular cones traveling at supersonic speeds . the analysis is based on

two theoretical techniques used successfully previously in application

to the case of uniform axial and inclined flow . in the first,

potential solutions for axial flow and crossflow are derived from the

first-order wave equation but in application to calculations for the

forces no approximations are made either to the tangency condition or to the isentropic pressure relation . the second method consists in combining the first-order crossflow potential with an axial-flow potential correct to second order . closed-form solutions by both methods are found for a cone, and numerical results for the stability derivatives are presented as a function of mach number for cones having semivertex angles of 10 and 20 .

in addition, expressions for the forces, moments, and stability derivatives of arbitrary bodies of revolution are obtained using newtonian impact theory . numerical results for cones compare well with those obtained from the combined first- and second-order potential theory at the highest mach number for which the latter theory is applicable .

.I 815

.T

investigation of several blunt bodies to determine trans- onic aerodynamic characteristics including effects of spinning and of extendible afterbody flaps and some measurements of unsteady base pressures .

.A

fisher,l.r. and dicamillo,j.r.

.B

nasa memo 1-21-59l, 1959.

.W

investigation of several blunt bodies to determine trans- onic aerodynamic characteristics including effects of spinning and of extendible afterbody flaps and some measurements of unsteady base pressures .

several blunt bodies having shapes that may be suitable for atmospheric reentry vehicles were tested to determine the aerodynamic characteristics of such shapes for angles of attack up to 34 . the tests were conducted through the transonic mach number range and at reynolds numbers from 1.74 x 10 to 2.78 x 10, based on body diameter .

a full-skirted rather than a short-skirted type of shape developed the greatest amount of static stability and the largest lift-curve slopes . the angle of attack for maximum lift for such bodies appears to be subject to mach number effects . spinning a full-skirted body about its longitudinal axis generally increased the lift and reduced the pitching moment at angles of attack and reduced the aerodynamic static stability parameter through the transonic mach number range . the extension of segmented clamshell-shaped flaps from the afterbody of a short-skirted model served to increase the lift and static stability only if the flaps extended into the airstream .

some evidence was found of oscillatory base pressures on two dissimilar shapes at certain high angles of attack and the highest mach number in these tests . there is doubt, however, that these pressures can induce any significant oscillatory motion for a reentry vehicle because of their small amplitude and phasing .

.I 816

.T

experimental investigation at a mach number of 3. 11

of the lift, drag and pitching moment characteristics

of a number of blunt low-fineness-ratio bodies .

.A

letko,w.

.B

nasa memo 1-18-159l, 1959.

.W

experimental investigation at a mach number of 3. 11

of the lift, drag and pitching moment characteristics

of a number of blunt low-fineness-ratio bodies .

a number of blunt bodies having shapes that may be suitable for

atmospheric reentry were tested to determine the lift, drag, and

pitching-moment characteristics at a mach number of 3.11 and a reynolds

number of 6 x 10 based on maximum body diameter of 2 inches .

the results of the tests showed that all the bodies were statically

stable about a point located one-third of the body length from the

nose . the results also showed that high-drag bodies which have a large

portion of their afterbodies negatively sloped (decrease in

cross-sectional area from nose to base) may have a negative lift-curve slope .

this negative slope results from the large negative lift component of

the axial force obtained with those bodies and the fact that with

negatively sloped afterbodies only small normal forces are developed .

.I 817

.T

loading paths and the incremental stress law .

.A

hankelman,g.h. and warner,w.h.

.B

j. math. and phys. 33, 1954, 157.

.W

loading paths and the incremental stress law .

this paper will be concerned with some properties of the stress-strain law for work hardening materials introduced by w. prager incremental strain or plastic flow law by which it is meant that the differentials of strain are expressed as functions of the stresses, strains, and differentials of the stress . we shall also have occasion to refer to total strain or plastic deformation laws, in which the strains are given directly as functions of the stress .

.I 818

.T

a quantitative comparison of flow and deformation theories of plasticity .

.A

hodge,p.g. and white,g.n.

.B

j. app. mech. 17, 1950.

.W

a quantitative comparison of flow and deformation theories of plasticity .

the stresses and displacements in a partly plastic, infinitely long, hollow cylinder are obtained according to the flow type of stress-strain law of prandtl-reuss and to the deformation law of hencky . in both cases the mises yield condition is used and the compressibility of the material is taken into account . it is shown that under these assumptions the two theories yield substantially the same results for this particular problem, but that one theory or the other may be preferable for computing purposes in certain cases . the results are compared with those of

other investigations in which different combinations of

stress-strain law, yield condition, compressibility, and end

loading were assumed .

.I 819

.T

stresses in the plastic range around a normally loaded

circular hole in an infinite sheet .

.A

magasarian,o.l.

.B

j. app. mech. 27, 1960, 65.

.W

stresses in the plastic range around a normally loaded

circular hole in an infinite sheet .

the stresses in the plastic range around a normally loaded circular hole

in an infinite sheet are found numerically on the basis of both

deformation and incremental theories . the results of deformation

theory are quantitatively assessed in the light of a criterion, recently

developed by budiansky, for the acceptability of deformation theories .

the criterion is completely satisfied . moreover, the results obtained

by using these two different theories of plasticity do not differ

greatly despite the fact that the stress paths are far from being

radial .

.I 820

.T

theories of plastic buckling .

.A

batdorf,s.b.

.B

j. ae. scs. 16, 1949, 405.

.W

theories of plastic buckling .

the theory for the plastic buckling of columns which appears finally to have achieved a satisfactory form, rests upon the well-established uniaxial stress-strain relation . the development of a correspondingly satisfactory theory for the plastic buckling of plates has been hampered by the nonexistence of an established polyaxial stress-strain relation in the plastic range .

present theories for the polyaxial stress-strain relation beyond the elastic range can be divided into two types, often called flow and deformation theories . theories of plastic buckling based on deformation theories are in better agreement with experiment than those based on flow theories . on the other hand, tests in which a material is compressed into the plastic range and then subjected to shear at constant compressive stress are in better agreement with flow than with deformation theories . legitimate doubt therefore has existed as to the validity of any theory for the plastic buckling of plates .

as a result of studying these apparent contradictions, a new theory of plasticity has been developed which is of neither the flow nor the deformation type . it is based upon the concept of slip, and its formulation was guided more by physical, and less by mathematical, considerations than previous theories .

experimental evidence of limited scope but of crucial character is in

better agreement with the new theory than with either flow or

deformation theories . the new theory accounts for the apparent

contradictions previously alluded to and justifies the use of

deformation theory in the analysis of the plastic buckling of plates .

.I 821

.T

inelastic column theory .

.A

shanley,f.r.

.B

j. ae. scs. 14, 1949, 261.

.W

inelastic column theory .

the action of a column in the plastic range is analyzed on the

basis that bending may proceed simultaneously with increasing

axial load . this leads to a new column formula that includes

both the tangent-modulus (engesser) and the reduced-modulus

the tangent-modulus load and that the column load increases

with increasing lateral deflection, approaching the

reduced-modulus load as a limit if the tangent modulus is assumed to

remain constant .

.I 822

.T

effects of imperfections on buckling of thin cylinders

and columns under axial compression .

.A

donnell,l.h. and wan,c.c.

.B

j. app. mech. 1950.

.W

effects of imperfections on buckling of thin cylinders

and columns under axial compression .

von karman and tsien have shown that under elastic

conditions the resistance of perfect thin cylinders

subjected to axial compression drops precipitously after

buckling . it is considered that this indicates that this type of

buckling is very sensitive to imperfections or disturbances .

in this paper the effects of certain imperfections of shape

turbances combined) are studied by the large-deflection

shell theory developed in a previous paper (2) .

it is found that two types of buckling failure may occur .

one is of a purely elastic type which occurs when the peak

of the average stress versus average strain curve is reached,

while the other type is precipitated by yielding, which for

thicker cylinders or lower-yield material may occur before

such a peak is reached . curves are derived giving the

dependence of each type of failure upon the dimensions and

elastic and yield properties of the specimen and also upon

an /unevenness factor/ u which determines the

magnitude of the initial imperfections and is assumed to depend

on the method of fabrication . the relations derived are in

line with test results, and similar studies of the buckling

of struts indicate that the magnitude of the initial imperfections which have to be assumed to explain test strengths

are reasonable .

.I 823

.T

plastic torsional buckling strength of cylinders including

the effects of imperfections .

.A

lee,l.h.n. and ades,c.s.

.B

j. ae. scs. 24, 1957, 241.

.W

plastic torsional buckling strength of cylinders including

the effects of imperfections .

the torsional buckling strength of a cylinder in the plastic

range has been determined . an energy solution and a more exact

solution, both based on a plastic stress-strain relationship given

by the simple deformation theory, are presented . close

agreement between the two solutions is found . the effects of large

deflections and imperfections on buckling strength are analyzed .

for two groups of experimental results used for comparison, the

effects of geometrical imperfections in the plastic range are

negligible . the theoretical results are found to be in good

agreement with the experimental results .

.I 824

.T

on the concept of stability of inelastic systems .

.A

drucker,d.c. and onat,e.t.

.B

j.ae.scs., 21, 1954, 543.

.W

on the concept of stability of inelastic systems .

simple models are employed to bring out the large and important differences between buckling in the plastic range and classical elastic instability . static and kinetic criteria are compared and their interrelation discussed . nonlinear behavior in particular is often found to be the key to the physically valid solution . the nonconservative nature of plastic deformation in itself or in combination with the nonlinearity requires concepts not found in classical approaches . conversely, the classical linearized condition of neutral equilibrium is really not relevant in inelastic buckling . plastic buckling loads are not uniquely defined but cover a range of values and are often more properly thought of as maximum loads for some reasonable initial imperfection in geometry or dynamic disturbance .

the models indicate that basically the same information is obtained from essentially static systems by assuming initial imperfection in geometric forms as by assuming dynamic disturbances . one approach complements the other and both are helpful in obtaining an understanding of the physical phenomena .

.I 825

.T

inelastic instability and incremental theories of plasticity .

.A

onat,e.t. and drucker,d.c.

.B

j. ae. scs. 20, 1953, 181.

.W

inelastic instability and incremental theories of plasticity .

a most troublesome paradox has existed for a number of years with respect to buckling in the plastic range . theoretical considerations and all direct experimental evidence show conclusively that an incremental or flow type of mathematical theory of plasticity is valid . however, the results of plastic buckling tests are well correlated by a simple total or deformation theory and bear no resemblance to published predictions of incremental theory .

the suggestion was made that initial imperfections of shape or loading might well explain this most peculiar result . however, subsequent investigations by several authors seem to have given the impression that excessively large imperfections would be needed and that the answer would be overly sensitive to the magnitude of such imperfections .

it is the purpose of this paper to demonstrate that extremely small, and therefore unavoidable, imperfections of shape do account for the paradox in a simple manner . the buckling load is shown to be extremely insensitive to the amount of imperfection . the example chosen is a simplified version of the long rectangular plate hinged along one edge and free on the other under uniform compressive stress at the ends . this is the equivalent of the case of the cruciform column, which has been so disturbing in the past because incremental theory applied to a perfect

cruciform column did lead to an entirely incorrect result .

.I 826

.T

small bending and stretching of sandwich type shells .

.A

reissner,e.

.B

naca tn.1832, 1949.

.W

small bending and stretching of sandwich type shells .

a theory has been developed for small bending and stretching of sandwich-type shells . this theory is an extension of the known theory of homogeneous thin elastic shells . it was found that two effects are important in the present problem, which have not been considered previously in the theory of curved shells .. (1) the effect of transverse shear deformation and (2) the effect of transverse normal stress deformation . the first of these two effects has been known to be of importance in the theory of plates and beams . the second effect was found to occur in a manner which is typical for shells and has no counterpart in flat-plate theory .

the general results of this report have been applied to the solution of problems concerning flat plates, circular rings, circular cylindrical shells, and spherical shells . in each case numerical examples have been given, illustrating the magnitude of the effects of transverse shear and normal stress deformation .

the results of this investigation indicate the necessity of taking account of transverse shear and normal stress in sandwich-type shells, as soon as there is an order-of-magnitude difference between the elastic constants of the core layer and of the face layers of the composite shell . it was found that the changes due to transverse shear and normal stress deformation in the core may be so large as to be no mere corrections to the results of the theory without transverse core flexibility .

the actual magnitude of the changes is greatly dependent on the geometry and loading condition of the structure under consideration so that no general rules may be given which indicate for which elastic modulus ratio the changes begin to be significant .

solutions of problems in the present theory may in general be obtained by mathematical methods which are similar to those employed in the theory of plates and shells without the effect of transverse shear and normal stress deformation included . the present work does not include consideration of buckling and finite deflection effects .

.I 827

.T

a nonlinear theory of bending and buckling of thin elastic shallow spherical shells .

.A

kaplan,a. and fung,y.c.

.B

naca tn.3212, 1954.

.W

a nonlinear theory of bending and buckling of thin elastic shallow spherical shells .

a shallow spherical dome subjected to lateral pressure is a structure for which the deformation departs appreciably from the linear theory at relatively small values of the deflection amplitude . it is also one for which the buckling process is characterized by a rapid decrease in the equilibrium load once the buckling load has been surpassed . for structures having this type of buckling characteristics the question arises as to whether the proper buckling criterion to apply is the classical criterion, which considers equilibrium with respect to infinitesimal displacements, or the finite-displacement /energy criterion/ proposed by tsien .

in this paper the problem of the finite displacement and buckling of a shallow spherical dome is investigated both theoretically and experimentally . in the theoretical approach the nonlinear equations are converted into a sequence of linear equations by expanding all of the variables in powers of the center deflection and then equating the coefficients of equal powers . the basic parameter for the shallow dome is proportional to the ratio of the central height of the dome h to its thickness t . for small values of this ratio the expansions converge rapidly and enough terms are computed to determine the buckling load according to the classical criterion . for higher values of h t, convergence deteriorates rapidly and it was not possible to determine the buckling load with the number of terms which were computed . however even for these higher values of h t the deflection shapes are determined for deflection amplitudes below the amplitude at which buckling occurs . these deflection shapes are characterized by their rapid change as h t increases and by the fact that, over most of the range of h t studied, the maximum deflection does not occur at the center of the dome .

experimental results seem to indicate that the classical criterion of buckling is applicable to very shallow spherical domes for which the theoretical calculation was made . a transition to energy criterion for higher domes is also indicated .

.I 828

.T

stresses and small displacements of shallow spherical shells .

.A

reissner,e.

.B

j. math. phys. 25, 1946, 80.

.W

stresses and small displacements of shallow spherical shells .

the purpose of the present paper is to derive a system of equations which can be used for the analysis of shallow segments of thin, elastic, spherical shells . a segment will be called shallow if the ratio of its height to base diameter is less than, say . the results obtained on the basis of this assumption will often also be applicable to shells which are not shallow, namely then, when the loads are such that the stresses are effectively restricted to shallow zones .

the problem of the spherical elastic shell has been the subject of numerous researches . for the rotationally symmetric case the fundamental results were obtained in 1912 (1) and have been the starting point of many applications . while it is possible to deduce from these results approximate equations equivalent to part of what follows, it is believed that the present approach to the problem of the shallow shell may be of some interest even for rotationally symmetric cases .

a number of investigations have been concerned with the shell loaded in a non-rotationally symmetric manner (2,3,4) . in its general form this problem is quite difficult and the results so far obtained are not easy to apply . restricting attention to the shallow shell in the manner of the present paper brings with it a very considerable simplification of the analysis .

.I 829

.T

stability of thin-walled tubes under torsion .

.A

donnel,l.h.

.B

naca r.479, 1933, 12.

.W

stability of thin-walled tubes under torsion .

in this paper a theoretical solution is developed for the torsion on a round thin-walled tube for which the walls become unstable . the results of this theory are given by a few simple formulas and curves which cover all cases . the differential equations of equilibrium are derived in a simpler form than previously found, it being shown that many items can be neglected . the solution obtained is length ratio is zero and infinite, and is a good approximation for intermediate cases . the theory is compared with all available experiments, including about 50 tests made by the author . the experimental-failure torque is

always smaller than the theoretical-buckling torque,

averaging about 75 percent of it, with a minimum of 60

percent . as the form of the deflection checks closely with

that predicted by theory and the experiments cover a great

range of shapes and materials, this discrepancy can

reasonably be ascribed largely to initial eccentricities in

actual tubes .

.I 830

.T

nonlinear deflections of shallow spherical shells .

.A

reiss,e.l., greenberg,h.j. and keller,h.b.

.B

j. ae. scs. 24, 1957, 533.

.W

nonlinear deflections of shallow spherical shells .

the equations obtained by chien for the nonlinear deflection

of shallow spherical shells under uniform external pressure are

solved by means of power series expansions, following procedures

introduced by friedrichs and stoker in their treatment of

buckling of circular plates . these equations depend upon two

parameters . one of these parameters is related to the external

pressure, while the other depends upon the dimensions of the shell .

the equations are solved for several ranges of the parameters

under boundary conditions corresponding to a fixed edge .

the solution, carried out numerically on the aec univac

at new york university, yields a complete description of the
stresses and deflections as functions of the polar angle over a
wide range of values of the loading parameter and the
dimensional parameter . prediction of the upper buckling load is then
made by means of a numerical criterion based on the load vs.
deflection curve . for some cases, the postbuckling behavior
is investigated . the results agree well with existing
experimental and theoretical studies and cover a wide range of cases
not previously treated .

.I 831

.T

buckling of shallow shells under external pressure .

.A

reiss,e.l.

.B

j. app. mech. 25, 1958,556.

.W

buckling of shallow shells under external pressure .
a formula for the initial buckling loads for clamped, shallow spherical
shells under uniform external pressure is obtained by combining the
solutions of two linearized versions of the original nonlinear problem .
one of these versions is a linear eigenvalue problem while the other is
the bending problem for a shallow cap in the linear theory of
elasticity . the formula, which is obtained in a simple manner, yields
buckling loads that are in better agreement with experiments than
previous approximate solutions to the nonlinear problem .

.I 832

.T

accelerating convergence of iteration processes .

.A

wegstein,j.h.

.B

comm. of assoc. for computing mach. 1958.

.W

accelerating convergence of iteration processes .

a technique is discussed which, when applied to an iterative procedure

for the solution of an equation, accelerates the rate of convergence if

the iteration converges and induces convergence if the iteration

diverges . an illustrative example is given .

.I 833

.T

a simple method of matric structural analysis, part

iv, non-linear problems .

.A

klein,b.

.B

j. ae. scs. 26, 1959.

.W

a simple method of matric structural analysis, part

iv, non-linear problems .

the method presented in the previous parts is employed to

solve various kinds of nonlinear problems, such as problems

concerning large deflections or buckling, or thermal creep, or

inelastic stress redistribution involving thermal gradients, or design . the procedure used in each case is one of direct

iteration--i. e., after one assumes a starting point all subsequent cycles are

self-generating . simple numerical examples are worked out .

.I 834

.T

limit design for economical missile structures .

.A

l. a. riedinger

.B

assistant manager-rand d structures dept., missile systems div.,

lockheed aircraft corp.

.W

limit design for economical missile structures .

a special safety factor alone won't do the trick in the design of

lightweight, high temperature missile structures . if you really want

to end up with the most efficient structure you can get, an entirely

new design approach is needed .

.I 835

.T

the problem of strain accumulation under thermal cycling .

.A

b. e. gatewood

.B

research coordinator, air force institute of technology,

wright-patterson air force base, ohio

.W

the problem of strain accumulation under thermal cycling .

parkes and sprague and huang have shown that it is possible for strain

growth to occur in a beam structure under temperature-load cycling .

the various aspects of this problem as to criteria for convergence and

divergence of the strain accumulation can be simply demonstrated by

thermal cycling one element of a two-element structure .

.I 836

.T

analytical and experimental investigation of stress distributions in

long flat plates subjected to lingitudinal loads and transverse

temperature gradients .

.A

sprague, g.h. and huang, p.c.

.B

wadc tr 55-350, 1955 .

.W

analytical and experimental investigation of stress distributions in

long flat plates subjected to lingitudinal loads and transverse

temperature gradients .

stress and strain distributions were studied in long flat plates in

order to develop practical analytical procedures for the design

analysis of aircraft structures at elevated temperatures .

various load-temperature conditions are presented . these include ..

methods of analysis for calculation of stress distributions under

and plastic range,.

ture .

experimental verification of the analytical procedures is shown with

comparisons between the use of constant room temperature or temperature dependent values of modulus of elasticity and coefficient of thermal

expansion .

the test specimen, equipment, instrumentation, and experimental program

are discussed in detail . experimental data obtained from the specimen

and associated material control coupon tests are presented .

.I 837

.T

inelastic behaviour of structures subjected to cyclic

thermal and mechanical stressing conditions .

.A

padlog,j., huff,r.d. and holloway,g.f.

.B

wadc tr 60-271, 1960.

.W

inelastic behaviour of structures subjected to cyclic

thermal and mechanical stressing conditions .

a general analytical procedure is outlined for structures subjected to

varying thermal and mechanical stressing conditions . consideration is

given to the accumulation of time-independent plastic strains and creep

strains . stress-strain-temperature-time relations for uniaxial and

multiaxial stresses are defined, based on various material behavior

assumptions . several of the assumptions are compared with a limited

number of time-varying temperature and uniaxial stress tests .

the procedure is illustrated by its application to uniaxial stress

problems in which /planes originally plane remain plane/ and to plane

stress plate problems . a solution, based on the influence coefficient

approach to the plane stress plate problem, is obtained which is

applicable to all plate plan forms, edge boundary conditions, and
inplane thermal and mechanical loading conditions .
from the predicted inelastic behavior of a three-bar structure
subjected to cyclic thermal and mechanical loading conditions, it is
shown that eventual failure could result from large permanent
deformation accumulations, tensile rupture, or thermal-stress-fatigue .
a sample plate with a centrally located hole was analyzed for two
cycles of a time-varying temperature and edge stress condition . both
plastic strain reversals and plastic strain growths were predicted at
the hole . however, a test-theory comparison indicated failure by
creep-rupture .

.I 838

.T

bending and compression tests of pressurised ring-stiffened
cylinders .

.A

dow,m.b. and peterson,j.p.

.B

nasa tn.d360, 1960.

.W

bending and compression tests of pressurised ring-stiffened
cylinders .
the results of tests on pressurized ring-stiffened cylinders
subjected to compression and bending are presented and discussed . the
results obtained at high values of internal pressure differ from those
obtained by previous investigators in that the theoretical
small-deflection compressive buckling coefficient of 0.6 was nearly achieved

in each test . small amounts of internal pressure had a greater

stabilizing effect in the bending tests than in the compression tests .

.I 839

.T

the bending stability of thin-- walled unstiffened

circular cylinders including the effects of internal

pressure .

.A

suer,h.s., harris,l.a., skene,w.t. and benjamin,r.j.

.B

j. ae. scs. 25, 1958.

.W

the bending stability of thin-- walled unstiffened

circular cylinders including the effects of internal

pressure .

in a recent paper, the authors presented a statistical,

semiempirical design procedure for the determination of the buckling

strength of unpressurized and pressurized cylinders under axial

compression . this procedure has been extended in the present

paper to the bending of unpressurized and pressurized cylindrical

shells and allows the calculation of the critical bending stress with

a knowledge of the cylinder geometry and the internal pressure

only .

because no published data could be found, an extensive series

of bending tests of pressurized cylinders has been performed .

these new data for pressurized cylinders are treated

semiempirically together with all of the other known test data for unpressurized cylinders . best-fit curves are presented using applicable

theoretical parameters . design curves for determining the

critical buckling stress for unpressurized and pressurized cylinders

in bending are then developed as 90 per cent probability curves

from the test data .

.I 840

.T

analysis of partly wrinkled membrane .

.A

stein,m. and hedgepeth,j.m.

.B

nasa tn.d813, 1961.

.W

analysis of partly wrinkled membrane .

a theory is derived to predict the stresses and deformations of

stretched-membrane structural components for loads under which part of

the membrane wrinkles . rather than studying in detail the deformations

in the wrinkled region, the present theory studies average displacements

of the wrinkled material . specific solutions of problems in flat and

curved membranes are presented . the results of these solutions show

that membrane structures retain much of their stiffness at loads

substantially above the load at which wrinkling first occurs .

.I 841

.T

on the bending of circular cylindrical shells under

pure bending .

.A

seide,p. and weingarten,v.i.

.B

j. app. mech. 28, 1961.

.W

on the bending of circular cylindrical shells under

pure bending .

the stability of circular cylindrical shells under pure bending is

investigated by means of batdorf's modified donnell's equation and the

galerkin method . the results of this investigation have shown that,

contrary to the commonly accepted value, the maximum critical bending

stress is for all practical purposes equal to the critical compressive

stress .

.I 842

.T

an improvement on donnell's approximation for thin-walled

circular cylinders .

.A

morley,l.s.d.

.B

q. j. mech. app. math. 12, 1959.

.W

an improvement on donnell's approximation for thin-walled

circular cylinders .

donnell's equation for thin-walled circular cylinders is replaced by

where w is a non-dimensional form of the radial displacement and q is

the distributed radial loading . this equation retains the essential

simplicity of the original but, unlike donnell's equation, the accuracy does not decrease as the wavelength of circumferential distortion

increases .

.I 843

.T

a simplified method of elastic stability analysis for

thin cylindrical shells .

.A

batdorf,s.b.

.B

i. donnell's equation. naca tn.1341, 1947.

.W

a simplified method of elastic stability analysis for

thin cylindrical shells .

the equation for the equilibrium of cylindrical

shells introduced by donnell in naca report no. 479 to

find the critical stresses of cylinders in torsion is

applied to find critical stresses for cylinders with

simply supported edges under other loading conditions .

it is shown that by this method solutions may be obtained

very easily and the results in each case may be expressed

in terms of two nondimensional parameters, one dependent

on the critical stress and the other essentially

determined by the geometry of the cylinder . the influence of

boundary conditions related to edge displacements in the

shell median surface is discussed . the accuracy of the

solutions found is established by comparing them with

previous theoretical solutions and with test results .

the solutions to a number of problems concerned with buckling of cylinders with simply supported edges on the basis of a unified viewpoint are presented in a convenient form for practical use .

.I 844

.T

flexural vibrations of the walls of thin cylindrical shells having freely supported ends .

.A

arnold,r.n. and warburton,g.b.

.B

proc. roy. soc. a, 197, 1949, 238.

.W

flexural vibrations of the walls of thin cylindrical shells having freely supported ends .

the paper deals with the general equations for the vibration of thin cylinders and a theoretical and experimental investigation is made of the type of vibration usually associated with bells . the cylinders are supported in such a manner that the ends remain circular without directional restraint being imposed . it is found that the complexity of the mode of vibration bears little relation to the natural frequency,. for example, cylinders of very small thickness-diameter ratio, with length about equal to or less than the diameter, may have many of their higher frequencies associated with the simpler modes of vibration . the frequency equation which is derived by the energy method is based on strain relations given by timoshenko . in this approach, displacement equations are evolved which are comparable to those of love and flugge, though differences are evident due to the strain expressions used by each author . results are given for cylinders of various lengths, each with the same thickness-diameter ratio, and also for a very thin cylinder in which the simpler modes of vibration occur in the higher frequency range . it is shown that there are three possible natural frequencies for a particular nodal pattern, two of these normally occurring beyond the aural range .

.I 845

.T

the flexural vibrations of thin cylinders .

.A

arnold,r.n. and warburton,g.b.

.B

j. proc. i. mech. e. 167, 1953, 62.

.W

the flexural vibrations of thin cylinders .

the flexural vibrations of the walls of thin cylinders are considered . in this type of vibration many forms of nodal pattern may exist owing to the combination of circumferential and axial nodes . theoretical expressions are developed for the natural frequencies of cylinders with freely-supported and fixed ends and a comparison is made with the frequencies obtained experimentally .

in practice, the ends of cylinders are subjected to a certain degree of fixing by end-plates, flanges, etc., and the natural frequencies thus lie between the corresponding values for freely-supported and fixed ends . to make possible the estimation of such frequencies, a method is devised in which an equivalent wavelength factor is used . this factor represents the wavelength of the freely-supported cylinder that would

have the same frequency as the cylinder under consideration when

vibrating in the same mode . the results of experimental investigations

with various end thicknesses and flange dimensions are recorded, and

from these the equivalent factors are derived .

  sets of curves calculated for cylinders with freely-supported ends and

covering a range of cylinder thicknesses are given . from these it is

possible to obtain close approximation to the frequencies of cylinders

under other end conditions by the use of an appropriate factor . an

example is given of frequency calculations for a large air-receiver for

which two frequencies were identified by experiment .

.I 846

.T

on the vibration of thin cylindrical shells under internal

pressure .

.A

fung,y.c., sechler,e.e. and kaplan,a.

.B

j. ae. scs. 27, 1957, 650.

.W

on the vibration of thin cylindrical shells under internal

pressure .

  the frequency spectra and vibration modes of thin-walled

circular cylinders subjected to internal pressure are considered .

it is shown that for very thin cylinders the internal pressure

has a significant effect on the natural vibration characteristics .

for these cylinders, particularly those having smaller length to diameter ratios, the mode associated with the lowest frequency is in general not the simplest mode . the exact number of circumferential nodes, n, which occur in the mode associated with the lowest frequency, depends on the internal pressure p . if this number n is large, it decreases rapidly with increasing p when p is small, and the /fundamental/ frequency--the lowest frequency at each p--increases rapidly with increasing internal pressure . at higher values of internal pressure the frequency spectrum tends to be arranged in the regular manner, the frequency increases with the increasing number of circumferential nodes, and the lowest frequency rises slowly with the internal pressure .

experimental results on the frequency spectra, vibration modes, and structural damping of a series of thin-walled cylinders subjected to internal pressure are briefly described . these results show agreement with the features predicted by reissner's

the effect of slight deviation of the cylinder from perfect circular symmetry is discussed .

.I 847

.T

experimental study of the vibrations of a circular cylindrical shell .

.A

gottenberg,w.g.

.B

j. acoust. soc. america, 32, 1960, 1002.

.W

experimental study of the vibrations of a circular

cylindrical shell .

an apparatus is described which permits the mode

shape of a vibrating circular cylindrical shell to be

obtained quite easily . these measurements are made

without contacting the cylinder and can be converted

to actual lineal values . a representative number of

results obtained with such a system are shown to

illustrate the relationship between the nodal pattern and

frequency in a cylinder as well as the effect of internal

pressure on these frequencies . finally, comparisons

are made between these results and timoshenko theory

and an appropriate shell theory .

.I 848

.T

the effect of an internal compressible fluid column

on the breathing vibrations of a thin pressurised cylindrical

shell .

.A

berry,j.g. and reissner,e.

.B

j. ae. scs. 25, 1958,288.

.W

the effect of an internal compressible fluid column

on the breathing vibrations of a thin pressurised cylindrical

shell .

the free oscillations of a thin pressurized cylindrical shell containing a compressible fluid are studied here . the use of an

approximate set of shell equations (shallow shell theory) leads to

a relatively simple formula for the natural frequencies of the

coupled fluid-cylinder system . the results of some

computations are presented .

.I 849

.T

a theory of imperfection for the vibrations of elastic bodies of

revolution .

.A

s. a. tobias, ph.d.

.B

.W

a theory of imperfection for the vibrations of elastic bodies of

revolution .

various observations and preliminary experiments

have shown that the effect of imperfections upon the

vibrations of bodies of revolution cannot be neglected .

owing to the possibility of applying the lagrange

equation, the influence of the imperfections could be

traced through the kinetic energy, the potential energy

and the dissipation function . although the fundamental

difficulty of the uncertainty of certain variables

was not eliminated, this procedure permitted at least

the making of general qualitative statements as to the

behaviour of the system if imperfections are present .

.I 850

.T

remarks on donnell's equations .

.A

kempner,j.

.B

j. app. mech. 22, 1955.

.W

remarks on donnell's equations .

flugge's set of differential equations of equilibrium for

circular cylindrical shells is expressed in a form analogous

to the donnell equations . the results of solutions of the

two sets of equations for a simply supported cylinder under

a centrally applied, uniformly distributed radial line load

over a generator segment, as well as under sinusoidally

applied line loads, are in very good agreement for the

particular geometry investigated .

.I 851

.T

energy expressions and differential equations for stress

and displacement analysis of arbitrary cylindrical

shells .

.A

kempner,j.

.B

j. ship res. 1958, 8.

.W

energy expressions and differential equations for stress and displacement analysis of arbitrary cylindrical

shells .

energy expressions and the related equilibrium equations and natural boundary conditions for the determination of the stresses in and displacements of uniform, thin-walled cylinders of arbitrary cross section loaded in an arbitrary manner by surface and edge forces and moments are presented . the derivations are based upon the kirchhoff-love assumptions of the classical theory of shells and are performed to within a degree of accuracy employed by flugge in his derivation of the equilibrium equations applicable to circular cylindrical shells,. hence, in terms of stress resultants, the exact, small-deflection equilibrium equations are obtained . methods of simplification of the relations derived and of solution of the differential equations presented are indicated .

.I 852

.T

stress and displacement analysis of simply supported

non- circular cylindrical shell under lateral pressure .

.A

romano,f. and kempner,j.

.B

pibal r.415, 1958.

.W

stress and displacement analysis of simply supported

non- circular cylindrical shell under lateral pressure .

this paper presents an analysis of the deflections of and stresses in a

short noncircular cylindrical shell of uniform wall thickness whose median-surface cross section is described analytically by a simple expression corresponding to a family of doubly symmetric ovals . the cylinder is under a uniform lateral load and is simply supported at its edges . the small deflection analysis considered is based upon a series solution of appropriate differential equations of shell theory which leads ultimately to infinite sets of algebraic equations, truncated forms of which are considered . numerical values of the significant stresses and displacements for points of the oval cylinder, which are 5 percent of the axial length and 2.5 percent of the circumferential length apart, have been calculated for an oval cross section with a major-minor axis ratio of 1.10 .

.I 853

.T

the accuracy of donnell's equations .

.A

hoff,n.j.

.B

j. app. mech. 22, 1955.

.W

the accuracy of donnell's equations .

solutions of donnell's equations of the small deformations of the perfectly elastic thin-walled circular cylindrical shell are compared with those obtainable from flugge's equations . the range of the basic parameters is found within which the two solutions are approximately equal .

.I 854

.T

boundary-value problems of the thin-walled circular

cylinder .

.A

hoff,n.j.

.B

j. app. mech. 21, 1954.

.W

boundary-value problems of the thin-walled circular

cylinder .

the homogeneous differential equations of donnell's

theory of thin cylindrical shells are integrated .

expressions are obtained in closed form for the displacements,

membrane stresses, moments, and shear forces .

.I 855

.T

simplified formulas for boundary value problems of

the thin-- walled circular cylinder .

.A

pohle,f.v. and nardo,s.w.

.B

j. app. mech. 22, 1955.

.W

simplified formulas for boundary value problems of

the thin-- walled circular cylinder .

n. j. hoff has presented formulas which can be used in

the solution of boundary-value problems of circular

cylinders . the purpose of this note is to express these results in exact simplified form,. a more detailed

investigation appears elsewhere . the notation will be that of

hoff unless otherwise stated .

.I 856

.T

some experimental studies of panel flutter at mach

1 .3.

.A

sylvester,m.a. and baker,j.e.

.B

naca tn.3914, 1957.

.W

some experimental studies of panel flutter at mach

1 .3.

experimental studies of panel flutter were conducted at a mach number of 1.3 to verify the existence of this phenomenon and to study the effects of some structural parameters on the flutter characteristics . thin rectangular metal plates were used in these studies and were mounted as a section of the tunnel wall . most of the data were obtained by using aluminum-alloy panels, although a few steel, magnesium, and brass panels were also used . different materials with various thicknesses and lengths were used to determine the effect of these parameters on panel flutter . the experimental program consisted of three phases .. panels clamped front and rear, and (3) buckled panels clamped on all four edges .

panel flutter was obtained under controlled laboratory conditions and it was found that, at the flow conditions of these tests, increasing tensile forces were effective in eliminating flutter, as were

shortening the panels or increasing the bending stiffness . no apparent

systematic trends in the flutter modes or frequencies could be observed,

and it is significant that the panel flutter sometimes involved higher

modes and frequencies . the presence of a pressure differential between

the two surfaces of a panel was observed to have a stabilizing effect .

initially buckled panels were more susceptible to flutter than panels

without buckling . buckled panels with all four edges clamped were much

less prone to flutter than buckled panels clamped front and rear .

.I 857

.T

experimental studies of flutter of buckled rectangular

panels at mach numbers from 1. 2 to 3. 0 including

effects of pressure differential and of panel

width-length ratio .

.A

sylvester,m.a.

.B

nasa tn.d833, 1961.

.W

experimental studies of flutter of buckled rectangular

panels at mach numbers from 1. 2 to 3. 0 including

effects of pressure differential and of panel

width-length ratio .

experimental panel flutter data have been obtained at mach numbers

from 1.2 to 3.0 for buckled rectangular panels and the effect of a

pressure differential has been determined . increasing the pressure differential was effective in eliminating flutter on most of the panels

tested . the effects of the variables in the panel flutter parameter

sure, e is young's modulus, and t and l are the panel thickness and

length, respectively) were investigated for buckled panels clamped on

the front and rear edges and a critical value of this parameter of 0.44

is indicated at zero pressure differential when the panel width-length

ratio is 0.69 . an estimated flutter boundary is presented for buckled

panels clamped on four edges, with width-length ratios of 0.21 to 4.0 .

this boundary shows that the panel width is more significant than the

panel length when the ratio of width to length is less than

approximately 0.5 . panels clamped on four edges and buckled in two half waves

in the direction of flow were found to be particularly susceptible to

flutter . the results of limited tests on panels with applied damping,

curvature, and lengthwise stiffeners are also presented and discussed .

.I 858

.T

experimental investigation at mach numbers 3. 0 of

the effects of thermal stress and buckling on the flutter

of four-bay aluminium alloy panels with length-width

ratios of 10 .

.A

dixon,s.c., griffith,g.e. and bohon,h.k.

.B

nasa tn.d921, 1961.

.W

experimental investigation at mach numbers 3. 0 of

the effects of thermal stress and buckling on the flutter of four-bay aluminium alloy panels with length-width

ratios of 10 .

skin-stiffener aluminum alloy panels consisting of four bays, each

bay having a length-width ratio of 10, were tested at a mach number

of 3.0 at dynamic pressures ranging from 1,500 psf to 5,000 psf and at

stagnation temperatures from 300 f to 655 f . the panels were

restrained by the supporting structure in such a manner that partial thermal

expansion of the skins could occur in both the longitudinal and lateral

directions .

a boundary faired through the experimental flutter points

consisted of a flat-panel portion, a buckled-panel portion, and a

transition point at the intersection of the two boundaries . in the region

where a panel must be flat when flutter occurs, an increase in panel

skin temperature (or midplane compressive stress) makes the panel more

susceptible to flutter . in the region where a panel must be buckled

when flutter occurs, the flutter trend is reversed . this reversal in

trend is attributed to the panel postbuckling behavior .

.I 859

.T

flutter of aerodynamically heated aluminium-alloy and

stainless steel panels with length-width ratio of 10

at mach 3. 0.

.A

guy,l.d. and bohon,h.l.

.B

nasa tn.d1353, 1962.

.W

flutter of aerodynamically heated aluminium-alloy and stainless steel panels with length-width ratio of 10 at mach 3. 0.

an investigation of the effects of aerodynamic heating on the flutter of multibay external-skin panels has been carried out at a mach number of 3.0 in the langley 9- by 6-foot thermal structures tunnel . both aluminum-alloy and 17-7 ph stainless-steel panels with a length-width ratio of 10 for each bay were tested at dynamic pressures between addition, a few tests were made on the lower vertical stabilizer of the x-15 airplane which has external-skin panels unsupported for a length

all panels showed flutter boundaries characterized by an increase in panel thickness required to prevent flutter with increasing thermally induced stress prior to buckling . after buckling the panels showed flutter boundaries characterized by a decrease in thickness required to prevent flutter with further increases in thermal stress . the largest thickness required to prevent flutter in the presence of aerodynamic heating occurred at the transition between the flat-panel boundary and the buckled-panel boundary . this peak value (for aluminum-alloy panel) was as much as 60 percent greater than the extrapolated value for an unheated, unloaded panel .

values of the modified-thickness-ratio flutter parameter for the unstressed panels (obtained by extrapolation) were in fair agreement for the aluminum, steel, and x-15 stabilizer panels . peak values at transition, however, showed large differences due to apparently minor changes in panel-support construction and or changes in panel-skin material .

.I 860

.T

test of an aerodynamically heated multi-- web wing

structure (mw-1) in a free jet at mach number 2.

.A

heldenfels,r.s., rosecrans,r. and grifith,g.e.

.B

naca rm l53e27, 1953.

.W

test of an aerodynamically heated multi-- web wing

structure (mw-1) in a free jet at mach number 2.

a multiweb wing structure, representing an airplane or missile wing,

was tested under simulated supersonic flight conditions to determine the

transient temperature distribution . the aerodynamic loads played an

important and unanticipated role, however, in that the model experienced

a dynamic failure near the end of the test . the test is discussed and

the conclusion reached that the model failed as a result of the combined

action of aerodynamic heating and loading . the temperature data

collected are analyzed and are shown to be in reasonable agreement with

calculated values .

.I 861

.T

charts adapted from van driest's turbulent flat-plate

theory for determining values of turbulent aerodynamic

friction and heat transfer coefficients .

.A

lee,d.b. and faget,m.a.

.B

naca tn.3811, 1956.

.W

charts adapted from van driest's turbulent flat-plate

theory for determining values of turbulent aerodynamic

friction and heat transfer coefficients .

a modified method of van driest's flat-plate theory for turbulent

boundary layer has been found to simplify the calculation of local

skin-friction coefficients which, in turn, have made it possible to obtain

through reynolds analogy theoretical turbulent heat-transfer

coefficients in the form of stanton number . a general formula is given and

charts are presented from which the modified method can be solved for

mach numbers 1.0 to 12.0, temperature ratios 0.2 to 6.0, and reynolds

numbers 0.2 x 10 to 200 x 10 .

.I 862

.T

the phenomenon of change in buckle pattern in elastic

structures .

.A

stein,m.

.B

nasa tr r39, 1959.

.W

the phenomenon of change in buckle pattern in elastic

structures .

a model is analyzed which exhibits the important properties

associated with change in buckle pattern of plates . the analysis includes

a rigorous study of stability in its various modes . a discussion of how the present results may be applied to plates and other elastic

structures is given .

.I 863

.T

loads and deformations of buckled rectangular plates .

.A

stein,m.

.B

nasa tr r40, 1959.

.W

loads and deformations of buckled rectangular plates .

the nonlinear large deflection equations of von karman for plates

are converted into a set of linear equations by expanding the

displacements into a power series in terms of an arbitrary parameter . the

post-buckling behavior of simply supported rectangular plates subjected

to longitudinal compression and to a uniform temperature rise is

investigated in detail by solving the first few of the equations .

experimental data are presented for the compression problem .

comparisons are made for total shortening and local strains and deflections

which indicate good agreement between experimental and theoretical

results .

.I 864

.T

status of flutter of flat and curved panels .

.A

national advisory committee for aeronautics

.B

.W

status of flutter of flat and curved panels .

representative results are presented to show the current status of

the panel flutter problem . the discussion includes flat panels with

and without midplane stresses, buckled panels, and both unstiffened and

stiffened infinitely long circular cylinders .

.I 865

.T

a study of the thermal fatigue behaviour of metals .

 the effect of test conditions on nickel-base high

temperature alloys .

.A

glenny,e. and taylor,t.a.

.B

j.inst.metals, 88, 1960, 449.

.W

a study of the thermal fatigue behaviour of metals .

 the effect of test conditions on nickel-base high

temperature alloys .

an attempt has been made to identify the significant factors governing

the thermal-fatigue behaviour of nickel-base high-temperature alloys,

mainly by using a laboratory technique with hot and cold fluidized beds

as the heating and cooling media . a succession of heating shocks is

generally more damaging than a succession of cooling shocks between the

same temperature limits . the duration of the heating shock and the

upper temperature of the cycle are dominant factors . the

thermal-fatigue cracks are initiated at the surface and are intercrystalline in origin and propagation . surface oxidation, which is intergranular in

nature for nickel-base alloys, has a significant effect on

thermal-fatigue life .

.I 866

.T

regularities in creep and hot fatigue data .

.A

graham,a. and walles,k.f.a.

.B

arc cp.379, 1958.

.W

regularities in creep and hot fatigue data .

published experimental results are assembled to support a

previously-given theory of uniaxial deformation, and the theory is then used to

analyse published data on the creep-rupture and hot-fatigue of

engineering materials . the theory enables data for different times and

temperatures to be classed together, thereby providing information over

a much greater range of times than could practicably be covered by

experiments at a single temperature . an underlying numerical pattern

common to all the widely different group 8 materials considered then

shows through the experimental scatter . data for further engineering

materials is considered in these terms in part 2 .

.I 867

.T

low frequency fatigue of nimonic 90 . low frequency

fatigue - a rheological approach .

.A

landau,c.s.

.B

engineer, 213, 1962, 911.

.W

low frequency fatigue of nimonic 90 . low frequency

fatigue - a rheological approach .

an alloy of nimonic 90 type has been tested under cyclic loads at

temperatures of 800 deg., 850 deg. and 900 deg. cent . graham's

descriptive theory of deformation has been developed in detail for the

tests reported here, and shown to provide a satisfactory description of

the experimental results . the variation of cumulative strain with

number of cycles can be described by the sum of powers of cycle number,

n, n, n, and n . there is qualitative agreement between the observed

and the calculated stress-strain loops . the effect of variation of

maximum stress per cycle can be described by the sum of power terms with

simple exponents . within the scatter of observation, the total time to

fracture is independent of the frequency .

.I 868

.T

design and operation of the n. g. t. e. thermal shock

analogue .

.A

stanworth,c.g. and paine,d.s.c.

.B

arc cp.557, 1960.

.W

design and operation of the n. g. t. e. thermal shock analogue .

a description is given of the n.g.t.e. thermal shock analogue which

is suitable for estimating the temperature in a turbine blade section as

a function of position and time when the blade is subjected to a step

change in gas temperature . the method of operating the analogue and

obtaining results has also been described .

the limitations of the analogue have been stated, but they are

considered a small penalty in view of the essential simplicity of the

design .

.I 869

.T

the calculation of transient temperature in turbine

blades and tapered discs using biot's variational method .

.A

howe,p.w.h.

.B

arc cp.617, 1961.

.W

the calculation of transient temperature in turbine

blades and tapered discs using biot's variational method .

transient temperatures in aerofoil

sections and tapered discs are

calculated taking advantage of simplifications

in heat flow analysis

achieved in biot's variational method .

cross-sections are represented by

a line of adjacent squares of various

sizes suitable for the local dimensions, e.g. small squares near the leading and trailing edges . the potential, dissipation and surface dissipation functions of biot's method are set up, and the lagrange equations lead, by automatic procedures, to an eigenvalue formulation in matrix form for the temperatures and their first time derivatives . solutions are sums of exponentials in time, and are evaluated by digital computer, requiring about five minutes for each cross-section and heat transfer coefficient . transient temperatures in a particular aerofoil section for variation of heat transfer coefficient and for external temperature depending exponentially on time agree with results obtained on an analogue computer . maximum transient temperature differences are evaluated for tapered discs by a simple electrical analogue) with variation of edge radius and heat transfer coefficient . peculiarities in the solution for cyclic temperature external to an aerofoil over a range of frequencies indicate

limitations in the mathematical formulation .

a successful solution for cyclic

external temperature might enable

eigenvalues to be separated out in

experimental measurements using

electronic equipment, and this might be

extended to exponential external

temperature if a relationship between

cyclic and exponential external

temperature could be established .

eigenvalues and eigenvectors as discrete

values arise fictitiously from the

sub-division into squares and the

possibility of an integral formulation

is mentioned . there is a possible,

but not immediate, extension to

cooled blades, whose cross-sections

are multiply-connected regions .

transient stresses due to creep, and

viscoelasticity might be included .

.I 870

.T

effect of rheological behaviour on thermal stresses .

.A

freudenthal,a.m.

.B

j.app.phys. 25, 1954.

.W

effect of rheological behaviour on thermal stresses .

since the conventional elastic analysis of thermal stress problems coupled with limiting creep rates and time-dependent fracture stresses as (inelastic) design criteria, results in design procedures for thermal stresses (in heat exchangers, nuclear reactors, flight structures at supersonic speeds, etc.) of considerable unreality, the effect of various types of rheological behavior (viscoelastic, plastic, work hardening) on the level of thermal stresses is analyzed under simplified assumptions, such as uniaxial stress and polar or cylindrical symmetry . the effect on the thermal stress intensity of the rheological behavior of the material is shown to be very significant, particularly with respect to stress relaxation and the development of residual stresses .

.I 871

.T

steady-state creep through dislocation climb .

.A

weertman,j.

.B

j.app. phys. 28, 1957, 362.

.W

steady-state creep through dislocation climb .

a dislocation climb creep model is considered which does not require the production of immobile dislocations . the creep equation that results from the analysis is

where a and b are constants, is the stress, q is the activation energy of creep and kt has its usual meaning . this equation is quite similar

to one previously proposed .

.I 872

.T

fundamentals of boundary layer heat transfer with streamwise

temperature variations .

.A

biot,m.a.

.B

j. ae. scs. 1962, 558.

.W

fundamentals of boundary layer heat transfer with streamwise

temperature variations .

boundary-layer heat transfer is analyzed for the case of a

sinusoidal distribution of temperature in the direction of flow . it is

shown that for both laminar and turbulent flow the spatial

distribution of heat transfer is generally out of phase with the wall

temperature by an angle of 30 to 45 . this leads to the

conclusion that in some areas the heat flow is opposite to the

temperature difference as used in the definition of the heat-transfer

coefficient, and points to the basic shortcomings of this concept .

the physical explanation for this behavior is found to be the

temperature-field distortion by the fluid motion . the distortion

is measured by the peclet number . approximate equations

representing a /conduction analogy/ were used in this analysis

and the validity of these equations for unsteady flow is examined

with reference to limitations in frequency and wavelength . a solution of these equations is given for the case of a velocity

profile which is not a straight line . the use of previously developed

variational principles for the evaluation of convective heat

transfer including cases of three-dimensional unsteady flow,

turbulence, and nonparallel streamlines is also discussed .

.I 873

.T

lagrangian thermodynamics of heat transfer in systems

including fluid motion .

.A

biot,m.a.

.B

j. ae. scs. 29, 1962, 568.

.W

lagrangian thermodynamics of heat transfer in systems

including fluid motion .

the lagrangian thermodynamic equations of irreversible

processes are extended to convective heat transfer . this

generalization provides equations for the unified analysis of transient

heat flow in complex systems comprising solid structures and

moving fluids in either laminar or turbulent flow . the concept

of a surface-heat-transfer coefficient is eliminated from the

formulation . the theory is developed along two different lines . in

one approach a new concept referred to as the /trailing function/

is introduced . it represents the surface-heat-transfer properties

and may be evaluated by quite simple but remarkably accurate

variational procedures . the method of /associated fields/ is also

generalized to convective phenomena . the second line of
approach extends to convective heat transfer the thermodynamic
concept of entropy production for both laminar and turbulent
flow . the theory amounts to an extension of the
thermodynamics of irreversible processes to systems for which onsager's
relations are not valid .

.I 874

.T

the use of models for the determination of critical flutter speeds .

.A

duncan, w.j.

.B

r + m 1425, july 1931 .

.W

the use of models for the determination of critical flutter speeds .
the use of model tests in the prediction of full-scale critical flutter
speeds is now well established, and the technique of such tests is
therefore worthy of discussion . in order to obtain critical speeds for
the model within the speed range of ordinary wind tunnels it is
necessary that the model should differ in some respect from a mere small
suggested by mckinnon wood the modification of the model consists in a
reduction of its effective stiffnesses . this method has the defect /in
most cases probably not serious/ that the model experiment is conducted
at a reynolds number much below that for full-scale . in the present
paper it is pointed out that an alternative method of reducing the
critical speed is to increase the mass loading of the model and to make
the flutter tests in compressed air . * it is then quite feasible to

reach the full-scale reynolds number . this method of reducing the critical speeds by a proportionate increase of all effective densities may also be combined with a reduction of the elasticity of the model . the relation of model and full-scale stresses at the critical flutter speeds is considered . where the reduction in critical speed is effected by increase of density only, the model and full-scale stresses are equal . in a model of reduced elasticity the stresses in the wires are the same as for full-scale, whereas, the stresses in the spars are less than for full-scale . this is in accord with the usual experience that the wires of such a model are the first parts of the structure to fail in a flutter .

lastly, the influence of gravity on flutter is considered . this is negligibly small for full-scale, but not necessarily so for the model . gravitational effects can sometimes be corrected by suitable orientation of the model .

.I 875

.T

models for aeroelastic investigation .

.A

templeton,h.

.B

arc cp.255, 1955.

.W

models for aeroelastic investigation .

this addendum provides a short note on two aspects omitted from the original paper, viz. gravitational effects and structural damping . a short list of references to earlier papers dealing with the subject is also added .

.I 876

.T

on flutter testing in high speed wind tunnels .

.A

lambourne,n.c. and scruton,c.

.B

arc r + m.3054, 1956.

.W

on flutter testing in high speed wind tunnels .

the requirements for simulating in a wind tunnel flutter conditions appropriate to high-speed flight are discussed, and an assessment is made of the desirable features of a wind tunnel suitable for flutter testing at transonic and supersonic speeds .

it is concluded that such a tunnel should have either the mach number or the stagnation pressure variable during the tunnel run, and that it is of considerable advantage, and for some purposes essential, for high stagnation pressures to be available . the stagnation pressure required to allow flight conditions to be simulated with a flutter model is considered to range from at least 2 atmospheres for transonic speeds to about 15 atmospheres for m = 4 . no attempt to simulate kinetic heating is envisaged, although its effect on stiffness should be allowed for in the design of the model . to minimise uncertainties due to the variation of the model stiffness with temperature it is desirable that means for controlling the stagnation temperature should be incorporated in the tunnel .

.I 877

.T

the influence of aerodynamic heating on the flexural

rigidity of a thin wing .

.A

mansfield,e.h.

.B

arc r + m.3115, 1957.

.W

the influence of aerodynamic heating on the flexural

rigidity of a thin wing .

this report considers the loss of flexural rigidity of a thin wing due to the presence of middle-surface stresses resulting from aerodynamic heating . the spanwise properties of the wing are assumed constant but the wing section is arbitrary . the loss of flexural rigidity is comparable with the corresponding loss of torsional rigidity .

.I 878

.T

experimental model techniques and equipment for flutter

investigations .

.A

molyneux,w.g.

.B

air. eng. 1958.

.W

experimental model techniques and equipment for flutter

investigations .

an outline is given of the uses of flutter models as an aid to the designer in the avoidance of flutter . details

are given of the different types and methods of

construction that are used for flutter models and of the

various test facilities that are available for high speed

and low speed tests .

the procedure followed in the u.k. for flutter

clearance of the full scale aircraft is described, and the value

of the electronic flutter simulator in this field is

discussed .

.I 879

.T

flutter model testing at transonic speeds .

.A

targoff, w.p. and white, r.p.

.B

inst. aero. sc. perp. 706, may 1957 .

.W

flutter model testing at transonic speeds .

flutter research on reflection plane models of straight, swept, and

delta wings in a 3 x 4 foot transonic test facility . techniques of

model construction and testing developed .

.I 880

.T

the design and testing of supersonic flutter models .

.A

mccarthy, j.f. and halfman, r.l.

.B

j. aero. sc. june, 1956 .

.W

the design and testing of supersonic flutter models .

the basic problems of flutter testing in the low supersonic speed range

simulate full-scale airplanes when mach number is included as a

parameter are reviewed and are compared with those where velocity is scaled

when the leading edges become transonic,. these are compared with

results given elsewhere and serve as a check on the results of tables

i and ii .

.I 881

.T

cumulative damage in fatigue .

.A

miner,m.a.

.B

j. app. mech. 12, 1945, a159.

.W

cumulative damage in fatigue .

the phenomenon of cumulative damage under repeated

loads was assumed to be related to the net work absorbed

by a specimen . the number of loading cycles applied

expressed as a percentage of the number to failure at a

given stress level would be the proportion of useful life

expended . when the total damage, as defined by this

concept, reached 100 per cent, the fatigue specimen should

fail . experimental verification of this concept for an

aluminum alloy, using different types of specimens, various stress ratios, and various combinations of loading

cycles is presented . these data are also analyzed to

provide information on different stress ratios when an s-n

curve for any one ratio is known . results of a sample

analysis based on experiments are given . it is

concluded that a simple and conservative analysis is possible

using the concept of cumulative fatigue damage .

.I 882

.T

the variation of gust frequency with gust velocity

and altitude .

.A

bullen,n.i.

.B

arc cp.324, 1956.

.W

the variation of gust frequency with gust velocity

and altitude .

  information on atmospheric turbulence obtained from counting

accelerometer records is examined and relations giving the variation of

gust frequency with gust velocity and altitude are obtained . the

results are summarized in a form convenient for use in estimating the

fatigue life of an aircraft .

.I 883

.T

correlated fatigue data for aircraft structural joints .

.A

heywood,r.b.

.B

arc cp.227, 1955.

.W

correlated fatigue data for aircraft structural joints .

results of fatigue tests carried out at r.a.e. on typical aircraft wing structural joints are correlated to give an indication of general fatigue behaviour . the results are plotted in the form of s - log n curves, and these indicate that the mode of behaviour cannot be attributed to any single factor, such as the type of aluminium alloy, the ultimate tensile strength, or the mean stress of the fatigue cycle . the detailed method of design undoubtedly has a predominant influence on behaviour, but this quality is not revealed by a broad classification according to the proportion of load transmitted at holes .

.I 884

.T

the estimation of fatigue damage on structural elements .

.A

chilver,a.h.

.B

rae tn.struct.125, 1954.

.W

the estimation of fatigue damage on structural elements .

a method is presented for the estimation of fatigue damage to aircraft structural elements . the gust spectrum to which the aircraft is subjected is analysed in terms of infinitesimal loading intervals . gust data supplied by j. taylor for flying below 15,000 ft are used to study fatigue damage in a number of aircraft structural joints and one

whole structure .

.I 885

.T

buckling of thin cylindrical shells under hoop stresses

varying in axial direction .

.A

hoff,n.j.

.B

j. app. mech. 24, 1957, 405.

.W

buckling of thin cylindrical shells under hoop stresses

varying in axial direction .

the buckling of a thin cylindrical shell simply supported

along the perimeter of its end sections is analyzed under

hoop compressive stresses varying in the axial direction .

the thermal stresses arising from a uniform increase in

the temperature of the cylinder are determined . it is

found that such thermal stresses are not likely to cause

elastic buckling . simple approximate formulas are

developed for buckling stress and thermal stress .

.I 886

.T

thermal buckling of clamped cylindrical shells .

.A

zuk,w.

.B

j. ae. scs. 24, 1957, 359.

.W

thermal buckling of clamped cylindrical shells .

the problem of thermal buckling of shells arises in

connection with air-frame bodies subject to aerodynamic heating at

supersonic speeds . the case of the shell with clamped edges is

presented, as this case typifies all structures with a tubular shell

stiffened at intervals with stiffening rings . the shell is assumed

to be unrestrained longitudinally and fully restrained laterally at

the edges .

.I 887

.T

buckling due to thermal stress of cylindrical shells

subjected to axial temperature distributions .

.A

johns,d.j.

.B

coa r.147, 1961.

.W

buckling due to thermal stress of cylindrical shells

subjected to axial temperature distributions .

thermal stress distributions in uniform circular cylindrical shells

due to axial temperature distributions are investigated . the

discontinuity effect due to the presence of a cooler stiffening bulkhead is

considered, and the possibility of thermal buckling of the shell due to

the circumferential discontinuity stress is examined . the buckling

analysis is based on donnell's shell equation, and particular attention is given to shells having clamped edges .

an experimental investigation of this buckling problem is discussed,

and the results obtained are seen to agree reasonably well with theory .

.I 888

.T

combinations of temperature and axial compression required

for buckling of a ring-stiffened cylinder .

.A

anderson,m.s.

.B

nasa tn.d1224, 1962.

.W

combinations of temperature and axial compression required

for buckling of a ring-stiffened cylinder .

a theory is presented to predict the buckling temperature of an

axially compressed, uniformly heated ring-stiffened cylinder . the

cylinder buckles because of the interaction of the axial stress due to

applied compressive loads and the circumferential stress resulting

from restraint of thermal expansion by the rings . buckling charts

covering a wide range of cylinder proportions are presented for both

clamped and simply supported cylinders . the buckling temperature for a

given axial loading is determined from a simple equation involving a

coefficient given in the buckling charts and the radius-thickness ratio

of the cylinder .

.I 889

.T

a simplified method of elastic stability analysis for thin cylindrical shells .

.A

batdorf,s.b.

.B

naca r.874, 1947.

.W

a simplified method of elastic stability analysis for

thin cylindrical shells .

this paper develops a new method for determining the buckling stresses of cylindrical shells under various loading conditions . for convenience of exposition, it is divided into two parts .

in part 1, the equation for the equilibrium of cylindrical shells introduced by donnell in naca report no. 479 to find the critical stresses of cylinders in torsion is applied to find critical stresses for cylinders with simply supported edges under other loading conditions . it is shown that by this method solutions may be obtained very easily and the results in each case may be expressed in terms of two nondimensional parameters, one dependent on the critical stress and the other essentially determined by the geometry of the cylinder . the influence of boundary conditions related to edge displacements in the shell median surface is discussed . the accuracy of the solutions found is established by comparing them with previous theoretical solutions and with test results . the solutions to a number of problems concerned with buckling of cylinders with simply supported edges on the basis of a unified viewpoint are presented in a convenient form for practical use .

in part 2, a modified form of donnell's equation for the equilibrium of thin cylindrical shells is derived which is equivalent to donnell's equation but has certain advantages in physical interpretation and in ease of solution, particularly in the case of shells having clamped edges . the solution of this modified equation by means of trigonometric series and its application to a number of problems concerned with the shear buckling stresses of cylindrical shells are discussed . the question of implicit boundary conditions also is considered .

.I 890

.T

comments on 'thermal buckling of clamped cylindrical shells' .

.A

david j. johns

.B

lecturer, college of aeronautics, cranfield, england

.W

comments on 'thermal buckling of clamped cylindrical shells' .
in the recent paper by zuk, an expression was presented for the critical buckling temperature of a clamped cylindrical shell in terms of the material and geometrical properties of the shell .
restraint at the edges of the shell was assumed to be provided by rigid frames experiencing no temperature rise .
the circumferential stress induced in the shell when it experienced a temperature rise, t, may be approximated by the function .
in other words, there is a compressive circumferential stress along the entire length, l, of the shell .

it is well known, however, that the discontinuity stresses introduced at

the junction of a shell and a rigid frame (or bulk-head) are extremely

localized, and the circumferential stresses induced in the shell

decrease rapidly away from the joint .

.I 891

.T

buckling of a finite length cylindrical shell under

a circumferential band of pressure .

.A

almroth,b.o. and bruch,d.o.

.B

j. ae. scs. 28, 1961, 573.

.W

buckling of a finite length cylindrical shell under

a circumferential band of pressure .

this paper is concerned with buckling of a circular cylinder of

finite length subjected to a symmetrical band of external pressure .

both experimental and theoretical results are presented . the

experimental data were obtained from tests of three thin-walled

steel cylinders subjected to external pressure by a pneumatic

tube encircling the test cylinder at mid-length . the theory is

based on the principle of minimum potential energy, and the

rayleigh-ritz procedure is used to expand the displacement

components in trigonometric series .

theoretical results are given in the form of graphs which show

buckling pressure as a function of the following ratios ..

cylinder radius thickness

cylinder length radius

pressure bandwidth cylinder length

theoretical results are in close agreement with existing solutions to special cases in which (1) the pressure is applied over the entire lateral surface, and (2) the pressure is concentrated along a circumferential line . the theoretical results are also in agreement with the test results .

.I 892

.T

research on unsteady flow .

.A

jones,w.p.

.B

j. ae. scs. 29, 1962, 249.

.W

research on unsteady flow .

this is a survey of certain recent advances made in the study of aerodynamic unsteady flow and of some of the new problems arising which require further investigation . no attempt is made to reproduce classical theory, but emphasis rather is laid on validity and general usefulness, particular attention being given to unsteady boundary-layer effects, especially when there is flow separation . coverage is broad and author thus provides a useful review for those interested in this field .

.I 893

.T

a new design of pitot-static tube with a discussion of pitot-static tubes and their calibration factors .

.A

salter,c.

.B

npl. aero. r.1013, 1962.

.W

a new design of pitot-static tube with a discussion

of pitot-static tubes and their calibration factors .

the report describes experiments devised to investigate some of the previously unexplained peculiarities of normal types of pitot-static tube . in the process use was made of what was as nearly as could be a standard static pressure tube .

the experiments led to a new alternative design of instrument having a nose of modified ellipsoidal shape and for which the main characteristics have been investigated .

it has been found to be necessary as well as desirable to include a discussion of the implications of the term /calibration factor/ and (although not in this respect comprehensive) of the special features and limitations of various pitot-static tubes .

.I 894

.T

flutter of a two dimensional simply supported buckled

panel with elastic restraint against edge displacement .

.A

smith,g.e.

.B

rae r. struct.269, 1961.

.W

flutter of a two dimensional simply supported buckled

panel with elastic restraint against edge displacement .

the critical flutter speed is evaluated for a two-dimensional thin

buckled panel with one surface exposed to a supersonic airstream and the

other to still air at the same static pressure . the panel is simply

supported along the leading and trailing edges by rigid edge members

separated by an elastic member represented by a compression spring .

the whole system is acted upon by a constant compressive force uniformly

distributed along the edge members . the aerodynamic forces acting on

the deflected panel are found from two-dimensional /quasi-steady/

theory, valid for slow oscillations where the downwash velocity is small

compared with the speed of flow and provided that the mach number is

sufficiently greater than . the elastic behaviour of the panel is given

by von karman's large deflection equations modified to cover initially

curved plates . the solution of the equations is carried out by means

of galerkin's method, which has been shown to give valid results for

a panel with a non-zero bending rigidity .

the influence of the midplane compressive force carried by the panel

itself, the initial buckle amplitude and the elastic restraint against

edge displacements is investigated, and curves are presented giving the

critical dynamic pressure ratio as a function of these variables .

.I 895

.T

the airforces on the low aspect ratio rectangular wing

oscillating in sonic flow .

.A

davies,d.e.

.B

rae tn. struct.311, 1962.

.W

the airforces on the low aspect ratio rectangular wing

oscillating in sonic flow .

approximate expressions for the

generalised airforces acting on a

rectangular wing of low aspect ratio

oscillating harmonically in sonic flow

at low frequencies are derived in this

paper . the modes of oscillation

considered are rigid modes and a small

selection of flexible modes . results

are presented as the first few terms of

infinite expansions .

a brief description of the modes

of oscillation and of the generalised

airforces is given towards the end of

the paper so that the results may be

used without the main text of the paper

having to be read .

.I 896

.T

the calculation of loads on a supersonic weapon in

the steady circling case .

.A

seal,d.m.

.B

rae tn.struct.275, 1960.

.W

the calculation of loads on a supersonic weapon in

the steady circling case .

the economy of a design depends

on the accuracy with which it is possible

to solve the various structural problems

as this has a direct bearing on

structure weight . this paper describes

the calculation of the bending moments

on a specific weapon for the high-g steady

circling case . a hybrid method is

used to obtain the aerodynamic loads .

the results presented show the effect

of a number of parameters, such

as - altitude, weight, acceleration and mach

number - on the magnitude of the

maximum bending moment .

.I 897

.T

some results on buckling and postbuckling of cylindrical

shells .

.A

kempner,j.

.B

nasa tn.d1510, 1962, 173.

.W

some results on buckling and postbuckling of cylindrical

shells .

in this summary paper, the

effects of initial deformations on the

buckling and postbuckling characteristics

of circular cylindrical

shells under hydrostatic pressure

is determined in an approximate

manner . the influence of initial

axisymmetric deformations is stressed .

also, the classical buckling of

an axially compressed, noncircular

the results show that the

major-minor axis ratio of the cross section

has a marked effect on the

critical load, and that use of the

maximum radius of curvature in the

formula for the classical buckling

stress of a circular cylindrical

shell leads to good results for moderate

eccentricities .

.I 898

.T

a survey of buckling theory and experiment for circular

conical shells of constant thickness .

.A

seide,p.

.B

nasa tn.d1510, 1962, 401.

.W

a survey of buckling theory and experiment for circular

conical shells of constant thickness .

a survey of the state-of-the-art

for the stability of thin-walled

conical shells is presented . known

theoretical results are summarized

and compared with experiment . the

shortcomings of present knowledge

and recommended work for the future

are discussed .

.I 899

.T

aerodynamic effects on boundary layer unsteadiness .

.A

moore,f.k.

.B

6th a.a.aero.conf. 1957.

.W

aerodynamic effects on boundary layer unsteadiness .

with a view to the study of

aerodynamic problems, a review is made

of boundary layer theory for a flat

plate moving with a time-dependent velocity . unsteady effects are shown to enter according to the magnitude of the ratio of time for diffusion to act throughout the boundary layer to the characteristic time of the imposed unsteadiness .

it is concluded that a boundary layer may be considered quasi-steady even during extreme flight manocuvres . generation of acoustic noise purely by boundary layer unsteadiness is generally small . thermal and heat-transfer effects are cited .

unsteady boundary layer considerations are important in damping or amplifying certain instabilities, such as flutter of panels and stalling flutter of aerofoils . in connection with the aerofoil problem, laminar separation concepts and the stagnation-point boundary layer are described for unsteady flow .

an analysis of aerofoil lift hysteresis is described, using unsteady laminar boundary layer considerations, which leads to a prediction of counter-clockwise hysteresis at maximum lift .

.I 900

.T

some measurements in the vortex flow generated by a

sharp leading edge having 65 sweep .

.A

lambourne,n.c. and bryer,d.w.

.B

arc cp.477, 1959.

.W

some measurements in the vortex flow generated by a

sharp leading edge having 65 sweep .

  the report is concerned with the vortex flow which arises

when separation occurs at a highly swept leading edge . measurements

were made in the flow over flat plates at 15 incidence each having

a sharp leading edge of 65 sweep . the pressure and velocity

distributions both along the axis of the vortex and for one

cross section of the flow are presented together with a preliminary

discussion of their significance .

.I 901

.T

long slender delta wings with leading edge separation .

.A

brown,c.e. and michael,w.h.

.B

naca tn.3430.

.W

long slender delta wings with leading edge separation .

the slender-body approximation of linearized compressible flow is applied to the problem of a delta wing in which flow separation occurs at the leading edges . the vortex sheets found in the real flow are approximated by concentrated vortices with feeding lattices, and a plausible adaptation of kelvin's theorem is applied to simulate the force-free nature of the vortex sheet .

the computations show that leading-edge separation produces an increase in lift over that given by the jones slender-wing theory and that the lift does not vary linearly with angle of attack . computed pressure distributions and span loadings are presented and the theoretical lift results are compared with the results of simple force tests made at a mach number of 1.9 .

.I 902

.T

some current and proposed investigations into the flow for slender delta and other wings in unsteady motion .

.A

lambourne,n.c.

.B

arc 21,844, 1960.

.W

some current and proposed investigations into the flow for slender delta and other wings in unsteady motion .

the practical need for research into the aerodynamics of slender delta wings in unsteady motion has been emphasized in a recent paper by zbrozek . two important aspects are .. - formation and presence of

leading-edge vortices .

with oscillatory or transient modes of longitudinal (or chordwise) bending .

the first of the aspects above, has already been briefly discussed in ref. 2 . one feature of the flow with leading-edge vortices which seems to be of particular significance to the dynamic behaviour of a wing is the shedding of vorticity at the leading edge as well as at the trailing edge . any time-dependent motion, or distortion, of the wing leads to a change in the rate at which vorticity is shed . with more conventional types of flow, the free vorticity being shed only from the trailing edge has diminishing influence on the wing, but when the free vorticity is shed from the leading edge, in passing downstream, it remains close to the upper surface of the wing . it might be expected then, that, although the magnitudes of the unsteady forces may not be

greatly affected for a slender delta, the

time delays associated with

the forces may be significantly different

for the attached and

separated regimes of leading-edge flow .

.I 903

.T

two dimensional transonic unsteady flow with shock

waves .

.A

eckhaus,w.

.B

m.i.t. fluid dynamics res. group r.59-3, 1959.

.W

two dimensional transonic unsteady flow with shock

waves .

a study is made of the unsteady flow around an

airfoil at transonic mach numbers, the situation being

such that local supersonic regions terminated by

shock-waves are present in the vicinity of the airfoil . for

the unsteady part of the flow, small perturbations

technique is employed and the interaction with the shock

wave is taken into account . the case of an oscillating

aileron is considered first, and a solution is derived

for the pressure distribution on the aileron . it is

found that the solution has a simple form when the

shock-wave is well ahead of the hinge axis of the
aileron . as the shock approaches the hinge-axis a
correction must be added to the solution . an
interpretation of these results is given . the results are
compared with results of a theory which neglects the
presence of the shock and it is found that both agree
for m = 1 . for m 1, however, neglecting the presence
of the shock waves introduces errors of the order of
magnitude (1 - m), where m is the local mach number
behind the shock .

the theory is finally extended to include the case
in which the whole airfoil oscillates, but only the
solution for the subsonic region behind the shock is
treated . the role of the unsteady shock-boundary layer
interaction is discussed and it is shown that this
mechanism can be included in the results of the present
theory .

.I 904

.T

calibration of the standard pitot-static head used
in the rae low speed wind tunnels .

.A

kettle,d.j.

.B

rae tech.memo 222, 1951.

.W

calibration of the standard pitot-static head used in the rae low speed wind tunnels .

recent results of tests in the r.a.e. wind tunnels concerned with the measurement of pressure distributions have shown slight discrepancies between the readings of various static pressure tubes and calculated pressure distributions . as a consequence some doubt was felt concerning the calibrations of tunnel static pressure and upon the validity of the reading given by the standard pitot-static head .

it was therefore decided to check the standard pitot-static head used in the r.a.e. wind tunnels, against an instrument similar to the measurements of static pressure were also made using a long tube where the interference from head and support is calculated to be small .

this note gives the results of tests made in the 5 ft open jet wind tunnel and the no. 1 11 ft wind tunnel in order to determine the necessary correction to the reading of static pressure given by the r.a.e.

pitot-static head . the tests were made

during september and october, 1951 .

.I 905

.T

comparative tests of pitot-static tubes .

.A

merriam,k.g. and spaulding,e.r.

.B

naca tn.546.

.W

comparative tests of pitot-static tubes .

comparative tests were made on seven conventional

pitot-static tubes to determine their static, dynamic, and

resultant errors . the effect of varying the dynamic

opening, static openings, wall thickness, and inner-tube

diameter was investigated . pressure-distribution measurements

showing stem and tip effects were also made . a tentative

design for a standard pitot-static tube for use in

measuring air velocity is submitted .

this report covers an investigation conducted under

the auspices of the national research council .

.I 906

.T

review of the pitot tube .

.A

folson,r.g.

.B

asme trans. 70, 1956.

.W

review of the pitot tube .

this paper is an attempt to bring together the

important information regarding pitot tubes and their use,.

to summarize the available data on the application of

various types of impact and velocity probes for the

guidance of engineers and research workers,. and to aid

them in the design of flow instruments for specific

applications .

.I 907

.T

cavitation and pressure distribution

head forms at zero angle of yaw .

.A

hunter rouse and john s. mcnown

.B

iowa institute of hydraulic research, state university of iowa

iowa city

.W

cavitation and pressure distribution

head forms at zero angle of yaw .

early in the fall of 1943 the iowa institute of hydraulic research

undertook the design and fabrication of a variable-pressure water

tunnel . as the tunnel neared completion, however, its immediate use

for the study of the pressure distribution around various body forms

was requested .

the original request for this investigation was a natural out-growth

of the need for systematic data on the distribution of pressure

in flow around various bodies, particularly under conditions leading to

cavitation, information which is desirable for the design of a wide

variety of navy equipment . ultimately the study is to include data

for two- and three-dimensional head and tail forms at various angles of

yaw . the first phase of the study, namely the investigation of

three-dimensional head forms at zero angle of yaw, is described herein .

three general geometric series have been studied.dash rounded,

ellipsoidal, and conical.dash together with other related forms .

the data obtained have been systematized to yield information for

a wide variety of geometrical forms either directly or by

interpolation . whenever possible, analytical methods have been used to

corroborate the experimental data and to provide a reliable means

of generalizing the results .

.I 908

.T

random vibration .

.A

crandal,s.

.B

app. mech. rev. 12, 1959, 739.

.W

random vibration .

random vibration is vibration which results

from an excitation which is not well represented by any simple

function (sinusoid, step, etc.) or any simple combination of

such functions but which is satisfactorily modeled by a
stochastic process . it is perhaps not too much of an exaggeration
to say that /all vibration is random vibration ./ every
vibration record contains /hash/ at some level . nevertheless,
until recently, engineering vibration theory has been able to
get along without including the consideration of random
excitations .

now in several fields simultaneously there has occurred a
burst of activity in the application of random processes . the
response of aircraft to buffeting from atmospheric turbulence
and the response of ships to confused seas have been put on
reasonably firm footing . possibly the most dramatic problems
have been posed by the development of large jet and rocket
engines which produce spectacular amounts of random
vibrational energy . the high level of random vibration in a jet
plane or a missile provides a severe environment with respect
to fatigue failure of structural members and with respect to
malfunctions of sensitive equipment .

.I 909

.T

the effect of jet noise on aircraft structures .

.A

clarkson,b.l.

.B

aero. quart. 10, 1959,103.

.W

the effect of jet noise on aircraft structures .

the present state of knowledge on the

problem of fatigue failure due

to vibrations excited by jet noise is reviewed .

it is concluded that it should

currently be possible to make reasonable

estimates of the stress levels set up

in a structure by jet noise but, in general,

the resultant fatigue life of the

components cannot be estimated

with any confidence .

.I 910

.T

natural frequencies of continuous beams of uniform

span length .

.A

ayre,r.s. and jacobson,l.s.

.B

j. app. mech. 17, 1950, 391.

.W

natural frequencies of continuous beams of uniform

span length .

a simple graphical network is used to determine the

natural frequencies of flexural vibration of continuous

beams having any number of spans of uniform length .

the network is based upon a relatively few calculated

values .

.I 911

.T

experimental study of the random vibrations of an aircraft

structure excited by jet noise .

.A

clarkson,b.l. and ford,r.d.

.B

u.s.a.a. r.128, univ. of southampton, 1960.

.W

experimental study of the random vibrations of an aircraft

structure excited by jet noise .

recordings have been made of the strains induced in a

full scale rear fuselage test structure of the caravelle

air-liner when one jet engine is running at maximum

take-off thrust . the analysis has been concentrated on the

strains in the centres of panels . correlation

measurements indicate that the larger panel strains occur above

resonance peak in each panel has been identified with the

fundamental stringer-twisting mode but the mode-shapes for

the two smaller peaks have not been completely determined .

an attempt has been made to calculate the panel resonant

frequencies theoretically .

.I 912

.T

the axisymmetric free-convection temperature field

along a vertical thin cylinder .

.A

hama,p.r., recesso,j.v. and christiaens,j.

.B

j. ae. scs. 1959, 335.

.W

the axisymmetric free-convection temperature field

along a vertical thin cylinder .

with a view to studying the effect of strong transverse

curvature on boundary-layer problems, the axisymmetric free-

convection problem along a vertical thin cylinder is investigated

theoretically as well as experimentally . a theory is developed

as an extension of the pohlhausen solution of a thick

axisymmetric laminar boundary layer by mark and by glauert and

lighthill . experiments consist of a thermocouple survey of the

temperature field over an electrically-heated brass cylinder of

diameter and 10 ft. height and an interferometric study

of the density field over a bare tungsten wire of 0.02-in. diameter

and 5 ft. height . the thermal-layer thicknesses are about five

and fifty times the radii of the cylinders, respectively .

experimental results of the local heat-transfer coefficient are in excellent

agreement with the theory . this, in turn, justifies the theories

of laminar boundary layer along a thin cylinder, at least indirectly .

.I 913

.T

vibrations of beams on many supports .

.A

miles,j.w.

.B

proc. a.s.c.e., 82, 1956.

.W

vibrations of beams on many supports .

the natural frequencies of a continuous

beam resting on an arbitrary

number of uniformly spaced supports are

determined from a difference equation

formulation . these frequencies fall in

periodically spaced groups that are

separated by spectral gaps of widths

equal to approximately half the interval

between the natural frequencies of a

single beam on a square root frequency

scale . these groups tend to uniform

spectra as the number of supports tends

to infinity, but the gaps remain, giving

a band-pass character to the entire

spectrum . wave propagation along an

infinite, periodically supported beam

is discussed and the phase and group

velocities evaluated as functions of

frequency .

.I 914

.T

transtability flutter of supersonic aircraft panels .

.A

r. p. isaacs

.B

rand corp.

.W

transtability flutter of supersonic aircraft panels .

for certain aero-elastic configurations it is possible to ascertain

critical flutter conditions from static considerations alone . the

idea is simply one of negation.. when the air speed exceeds a certain

value statically stable equilibrium - and sometimes equilibrium itself

take place . there are times when the dynamics of a situation are

complex enough to defy a tractable analysis . the value of being able

to indicate a flutter criterion from the simpler statics is clear .

we will suppose flutter begins when some critical value of the air

speed (or some parameter simply related the to) is exceeded . here

we will show that there is a critical value which, when exceeded,

precludes static equilibrium . underlying our work is the premise

that these two critical values are the same . this assumption begs

discussion .

we will call the lowest value of our air speed parameter to preclude

statically stable equilibrium of the system the transtability value .

in some cases, excess of this value will ban all possibility of

static equilibrium - stable or not., we will then call it a strong

transtability value .

.I 915

.T

a buckled plate in a supersonic stream .

.A

hayes, w.

.B

naaa inc. r al1029

(abstract by e.m.keen)

.W

a buckled plate in a supersonic stream .

the forcible buckling of an

external skin plate from a guided missile, with a pinned end,

is considered in a supersonic air flow . conidtions of dynamic stability

orthogonality and expansion of the buckling mode, the degree of

freedom and cases of small deflection are considered .

without a 50 half angle conical afterbody

in a pressurized ballistic

of range at nominal mach numbers of 3.5 and

of 90000 and 220000, respectively . it

.I 916

.T

the flow around oscillating low aspect ratio wings at transonic

speeds .

.A

landahl, m.t.

.B

kth aero. t.n.40, 1954 .

.W

the flow around oscillating low aspect ratio wings at transonic

speeds .

when certain conditions are fulfilled for thickness ratio, aspect ratio,

and reduced frequency for a three-dimensional wing, it can be shown

that the partial differential equation for the non-steady perturbation potential can be reduced to a comparatively simple linear equation .

the solution is then obtained by applying a fourier transformation in

the free-stream direction and then using an iterative process developed

by adams and sears for steady flow . the method gives solutions valid

for low combinations of aspect ratio and reduced frequency .

the method is applied to a delta wing oscillating in some selected rigid

and elastic modes . from the results it can be seen that the special

non-steady forces in the potential equation, which are neglected in

slender-body theory, are very important . stability derivatives can also

be obtained by the method and it is seen that the damping in pitch may

be negative at m 1 for delta wings of too high aspect ratio .

.I 917

.T

a method of calculating the short period longitudinal

stability derivatives of a wing in linearised unsteady

compressible flow .

.A

mangler,k.w.

.B

rae r. aero.2468.

.W

a method of calculating the short period longitudinal

stability derivatives of a wing in linearised unsteady

compressible flow .

a method is developed for the

calculation of the pressure

distribution and the aerodynamic forces and moments on a wing performing harmonic pitching and heaving oscillations . the calculation is based on the assumption of inviscid potential flow without shock waves and is restricted to small incidence, so that the linearized theory is valid .

in contrast to other work in the field the theory applies to all mach numbers . it is restricted to small values of the reduced frequency and should be valid for the usual range of short periods occurring at present in flight . the formal solution yields two integral equations for the parts of the load, which are in phase and go out of phase with the oscillation,. these are of the same form as the corresponding equation in steady flow .

the way is thus opened for solutions over the whole mach number range at small frequencies, if the corresponding steady solutions can be found . the calculation is in fact easiest for m = 1 and has been done here for delta-wings to supplement a previous supersonic calculation,

made on different frequency assumptions,

which broke down near m = 1 .

it appears from the two sets of results

that the short period oscillation

will be unstable near m = 1, if the apex

angle of the delta wing is

greater than about 60 . this confirms a

now generally recognised trend .

such results near m = 1 must of

course be invalidated to an

unknown extent by thickness viscosity and

shock waves at their maximum

effect . nevertheless it is unlikely that

these factors will remove

the critical nature of the transonic damping

as calculated by this method .

with all its obvious limitations this method,

when extended to other

planforms, should provide a useful tool in

studying the effect of

geometrical parameters on the stability of

an aircraft at transonic

speeds .

.I 918

.T

on the low aspect ratio oscillating rectangular wing in supersonic flow .

.A

miles,j.w.

.B

aero. quart. 4, 1953, 231.

.W

on the low aspect ratio oscillating rectangular wing

in supersonic flow .

the laplace transform of

the lift distribution on an oscillating

rectangular wing in a supersonic flow is obtained

by separating the linearised equation

for the velocity potential in elliptic (cylindrical)

co-ordinates . the results for the case

of no spanwise distortion are expanded in ascending

powers of the aspect ratio in order

to compare with the slender body theory, and the

longitudinal stability derivatives are

calculated . it is found that at either supersonic

or transonic speeds single-degree-

of-freedom instability in pitch is impossible insofar

as the fourth power of the aspect

ratio is neglected .

.I 919

.T

theoretical studies of unsteady transonic flow . part iii . the

oscillating low aspect ratio rectangular wing .

.A

landahl, m.t.

.B

aero. res. inst. of sweden /f.f.a./ rep. 79, 1958 .

.W

theoretical studies of unsteady transonic flow . part iii . the

oscillating low aspect ratio rectangular wing .

by expanding the velocity potential in an asymptotic series, the

aerodynamic forces on an oscillating low aspect ratio rectangular wing are

calculated . the approximate theory is valid for small values of ko /o

semi-span-to-chord ratio,. k reduced frequency/ and complements an

earlier low-aspect-ratio-wing theory by the author valid only for

pointed wings like delta wings . the present report gives formulas for

the calculation of generalized forces for any smooth, flexible or rigid

mode of oscillation with spanwise symmetry .

comparisons with the slender-wing theory show that, except for wings of

very low aspect ratio, unsteady-flow effects are appreciable even at

fairly low reduced frequencies . near the upper limit in ko for the

applicability of the present theory good agreement is obtained with a

recent theory for high aspect ratios .

.I 920

.T

supersonic flow over an inclined wing of zero aspect ratio .

.A

stewartson, k.

.B

proc. camb. phil. soc. v. 46, pt. 2, p307, april 1950 .

.W

supersonic flow over an inclined wing of zero aspect ratio .

an asymptotic expression is found for the lift distribution on a long,

narrow, laminar wing, at incidence in a supersonic stream . the

approximations of the linearized potential theory are used .

.I 921

.T

slender-body theory-review and extension .

.A

adams, m.c. and sears, w.r.

.B

j. aero. sc. v. 20, february 1953 .

.W

slender-body theory-review and extension .

the approximate theory of flow about slender bodies and wings originated
by munk and jones is reviewed . it is presented here in a form that
emphasizes the relation to the source-sink methods of von karman and
others . the extension to noncircular bodies is made for subsonic flow,
paralleling ward's extension for supersonic flow . the calculation of
pressures and forces and the extension of the theory to unsteady flows
are reviewed, and some discrepancies in the published literature are
explained .

finally, interpreting the jones slender-wing result as the first term of
an expansion in powers of a breadth parameter /e.g., aspect ratio/, it
is shown how a more accurate theory can be developed by carrying
additional terms for both subsonic and supersonic speeds . this theory of
not-so-slender wings is applied to some practical wing problems,

including direct problems of flow past given wings and problems of wing design for minimum drag . the accuracy of the new results is assessed by comparison with linearized supersonic-airfoil theory for the special case of a flat delta wing .

.I 922

.T

supersonic flow past slender bodies of revolution, slope of whose median section is discontinuous .

.A

lighthill,m.j.

.B

q. j. mech. app. math. 1948, 90.

.W

supersonic flow past slender bodies of revolution, slope of whose median section is discontinuous .

the theory of supersonic flow around slender bodies of revolution, yawed or unyawed, with pointed or open bows, based on the linearized equation, is extended to the case when the meridian section of the outer surface has discontinuities in slope . expressions for the pressure distribution on the surface are obtained . it is found that the drag coefficient is no longer independent of mach number, and tends to zero more slowly than the square of the thickness of the body . the large pressure change behind a discontinuity is made up

remarkably rapidly . the first

approximation to the lift coefficient is unchanged .

.I 923

.T

methods for estimating lift interference of wing-body

combinations at supersonic speeds .

.A

nielsen,j.n. and kaattari,g.e.

.B

naca til.2950.

.W

methods for estimating lift interference of wing-body

combinations at supersonic speeds .

the modified slender-body method

used by nielsen, katzen, and tang

in rm a50f06, 1950, to predict the

lift and moment interference of

triangular wing-body combinations has

been adapted to combinations with

other than triangular wings . that

part of the method for predicting the

effect of the body on the wing has

been retained, but a new method for

predicting the effect of the wing on

the body has been presented . these

methods have been applied to the

prediction of the lift-curve slopes of

nearly 100 triangular, rectangular,

and trapezoidal wing-body

configurations . the estimated and experimental

values for the lift-curve slopes

agree for most of the cases within 10

percent . some of the

higher-order effects that must be taken into

account in a theory that is to

give greater accuracy than the present

one are discussed . a numerical

example illustrating the method is included .

.I 924

.T

a method for calculating the lift and centre of pressure

of wing-body-tail combinations at subsonic, transonic

speeds .

.A

nielsen,j.n., kaattari,g.e. and anastasio,r.f.

.B

naca til.3959.

.W

a method for calculating the lift and centre of pressure

of wing-body-tail combinations at subsonic, transonic

speeds .

a method is presented for

calculating the lift and pitching-moment characteristics of circular cylindrical bodies in combination with triangular, rectangular, or trapezoidal wings or tails through the subsonic, transonic, and supersonic speed ranges . the method covers unbanked wings, sweptback leading edges or sweptforward trailing edges, low angles of attack, and the effects of wing and tail incidence . the wing-body interference is handled by the method presented in naca rm's a51j04 and a52b06, and the wing-tail interference is treated by assuming one completely rolled-up vortex per wing panel and evaluating the tail load by strip theory . a computing table and set of design charts are presented which reduce the calculations to routine operations . comparison is made between the estimated and experimental characteristics for a large number of wing-body and wing-body-tail combinations . generally speaking, the lifts were estimated to within 10 percent and the centers of pressure were estimated to within effect of wing deflection on wing-tail

interference at supersonic speeds

was not correctly predicted for triangular

wings with supersonic leading

edges .

.I 925

.T

factors affecting loads at hypersonic speeds .

.A

henderson,a. and bertram,m.h.

.B

naca til.5576.

.W

factors affecting loads at hypersonic speeds .

this paper gives a brief summary of current loads information at

hypersonic speeds . several methods which the designer can employ in

estimating the loads on various aircraft components are discussed . the

paper deals with the characteristics of both slender and blunt

configurations and touches upon the effects of boundary-layer and

aerodynamic interference .

.I 926

.T

post buckling behaviour of circular cylinderical shells

under hydrostatic pressure .

.A

kempner,j.

.B

j. ae. scs. 24, 1957, 253.

.W

post buckling behaviour of circular cylinderical shells

under hydrostatic pressure .

the postbuckling behavior of initially perfect, thin-walled,

circular cylindrical shells under hydrostatic pressure is

investigated with the aid of the principle of stationary potential energy

together with appropriate approximate deflection functions .

calculations show that postbuckling equilibrium configurations

exist for loads greater than as well as loads slightly less than the

critical load calculated from small-deflection theory . loads

less than the critical load are obtained only for a finite range of a

parameter indicative of shell geometry . for loads corresponding

to radial displacements of the order of the shell thickness, it is

found that the number of circumferential waves remain

essentially constant with increasing deflection and equal to the

number of waves developed at buckling .

.I 927

.T

investigation of normal force distributions and wake

vortex characteristics of bodies of revolution at supersonic

speeds .

.A

mello,j.f.

.B

j. ae. scs. 1959, 155.

.W

investigation of normal force distributions and wake vortex characteristics of bodies of revolution at supersonic speeds .

the supersonic aerodynamic characteristics of inclined bodies of revolution at high angles of attack have been investigated in order to provide a more basic understanding of the body vortex wake flow and its relation to the problem of body-wing interference . the results of wind-tunnel tests, whereby the normal force, pitching moment, normal force distributions, and the local flow properties in the vicinity of the body were determined, are discussed and analyzed .

comparisons of experimental normal force coefficient and center of pressure data with values calculated in accordance with theories which include methods for estimating the effects of viscosity show that the accuracy of these estimates is strongly dependent on the body fineness ratio and the angle of attack . further comparisons of the distributions of theoretical and experimentally derived cross-flow drag coefficients clearly show that, in general, the disagreement between experiment and existing theories is due to the inadequate prediction of the magnitude and distribution of the forces resulting from flow separation .

the circulation strengths of the concentrated vortices and the circulation strengths of the vortex feeding sheets in the body vortex wake are determined by closed-contour velocity-perimeter integrations for paths enclosing the vortex or the feeding sheet . the values of vortex strength calculated in this manner are in close agreement with the values predicted by vortex strength formulas written for a simple theoretical model for which it is assumed that the cross-flow in any plane along the cylindrical portion of the body is represented by the steady incompressible potential flow about a cylinder, two symmetrical vortices of equal strength, and the attendant image vortices . however, in computing these strengths it is necessary to use the vortex locations and the viscous normal force distributions determined from experiment .

the experimentally determined values of vortex strength are, in turn, used to calculate--by means of the aforementioned incompressible cross-flow potential--the local flow inclination angles which are in good agreement with the measured values, except in the vortex core, in the vicinity of the feeding sheet, and in regions for which transonic cross-flow velocities are expected . a consideration of these various regions with simple methods which account for the observed phenomena leads to substantial improvement in the agreement between theory and experiment .

it is indicated that the complete vortex wake flow may be adequately predicted for a body of revolution (for conditions represented by the theoretical flow model), provided that the distribution of the viscous normal force and the vortex locations are accurately known .

.I 928

.T

a new theory for the buckling of thin cylinders under

axial compression and bending .

.A

donnell,l.h.

.B

asme trans. 56, 1934, 795.

.W

a new theory for the buckling of thin cylinders under

axial compression and bending .

the results of experiments on axial loading of cylindrical

shells (thin enough to buckle below the elastic limit and

too short to buckle as euler columns) are not in good

agreement with previous theories, which have been based on the

assumptions of perfect initial shape and infinitesimal

deflections . experimental failure stresses range from 0.6

to 0.15 of the theoretical . the discrepancy is apparently

considerably greater for brass and mild-steel specimens

than for duralumin and increases with the radius-

thickness ratio . there is an equally great discrepancy between

observed and predicted shapes of buckling deflections .

in this paper an approximate large-deflection theory is

developed, which permits initial eccentricities or

deviations from cylindrical shape to be considered . true

instability is, of course, impossible under such conditions,.

the stress distribution is no longer uniform, and it is

assumed that final failure takes place when the maximum

stress reaches the yield point . the effect of initial

eccentricities and of large deflections is much greater than for

the case of simple struts . measurements of initial eccentricities in actual cylinders have not been made,. however, it is shown that most of these discrepancies can be explained if the initial deviations from cylindrical form are assumed to be resolved into a double harmonic series, and if certain reasonable assumptions are made as to the magnitudes of these components of the deviations . with these assumptions the failing stress is found to be a function of the yield point as well as of the modulus of elasticity and the radius-thickness ratio . on the basis of this a tentative design formula (5) is proposel, which involves relations suggested by the theory but is based on experimental data .

it is shown that similar discrepancies between experiments and previous theories on the buckling of thin cylinders in pure bending can be reasonably explained on the same basis, and that the maximum bending stress can be taken as about 1.4 times the values given by equation buckling problems can probably be explained by similar considerations, and it is hoped that this discussion may help to open a new field in the study of buckling problems . the large-deflection theory developed in the paper should be useful in exploring this field, and may be used in other applications as well .

the paper presents the results of about a hundred new tests of thin cylinders in axial compression and bending, which, together with numerous tests by lundquist, form the experimental evidence for the conclusions arrived at .

.I 929

.T

stability of the cylindrical shell of variable curvature .

.A

marguerre,k.

.B

naca tm.1302,1951.

.W

stability of the cylindrical shell of variable curvature .

the report is a first attempt to devise a calculation method

for representing the buckling behavior of cylindrical shells

of variable curvature . the problem occurs, for instance, in

dimensioning wing noses, the stability behavior of which is

decisively influenced by the variability of curvature . the

calculation is made possible by simplifying the stability

equations (permissible for the shell of small curvature) and

by assuming that the curvature as a function of the arc

length s can be represented by a very few fourier terms .

we evaluated the formulas for the special case of an

ellipse-like half oval with an axis ratio under

compression in longitudinal direction, shear, and a combination of

shear and compression . however, the results can also be

applied approximately to an unsymmetrical oval-shell segment

under compression, shear, and bending so that the numerical

values contained in the diagrams 10 to 12 represent directly

dimensioning data for the wing nose .

.I 930

.T

general theory of large deflections of thin shells

with special applications to conical shells .

.A

dill,e.h.

.B

nasa tn.d826, 1961.

.W

general theory of large deflections of thin shells

with special applications to conical shells .

a general theory is developed for the case of large deflections but with rotations of the elements negligible compared to unity . the derivation is carried out in tensor form and therefore any coordinate system on the surface of the shell can be used . the effect of initial imperfections is included . it is shown that for shells of negligible gaussian curvature (shallow shells and developable surfaces), the problem can be reduced to the solution of two fourth-order partial differential equations in a stress function and the deflection normal to the shell . for shells forming a surface of revolution the results are indicated in terms of the equation of the generating curve . the differential equations for the conical shell are then listed .

.I 931

.T

stability equations for conical shells .

.A

kempner,j.

.B

j. ae. scs. 25, 1958, 137.

.W

stability equations for conical shells .

the author rewrites v. s. vlasov's equations for (linear)

stability theory of shells (prikl. mat. meh. 8(1944),

109-placements . the result is a single eighth-order equation

for normal deflection, and two fourth-order equations

relating the displacement components in the shell middle

surface to the normal displacement .

.I 932

.T

buckling of circular cones under axial compression .

.A

lackman,l. and penzien,j.

.B

j. app. mech. 27, 1960, 458.

.W

buckling of circular cones under axial compression .

presented are the results of an experimental

investigation to determine the buckling

strength of right circular cones under axial

compression . correlation of these data is

made with existing theory and with previously

published experimental data on circular

cylinders,. thus a recommended procedure

for predicting the buckling load of right

circular cones under the foregoing loading condition is presented .

.I 933

.T

the characteristics of roughness from insects as observed

for two-dimensional, incompressible flow past airfoils .

.A

coleman,w.s.

.B

j. ae. scs. 1959, 264.

.W

the characteristics of roughness from insects as observed

for two-dimensional, incompressible flow past airfoils .

advances in the practical development of boundary-layer control for the maintenance of extensive laminar flow have drawn attention to the problem of surface roughness, due not only to artificial irregularities such as rivet heads, lap joints, window panels, etc., but also to the kind generated in flight from impact with insects . this natural form of roughening, the effects of which have been noted, though not investigated previously, is the subject of the present paper .

the phenomenon may be divided into two parts--namely, and (2) its effect upon the stability of the laminar boundary layer . wind-tunnel experiments with the fruit fly, drosophila, and the common housefly for the investigation of both (1) and airfoils are fully described . the former problem has also been treated mathematically in a separate paper, not yet published, agreement between theory and experiment being satisfactory in all essentials .

the characteristics of the roughness profile consist principally of a pronounced peak near the leading edge, followed by an extensive area of surface over which there is a much reduced and gradually diminishing value of the excrescence height . further, it is shown that, if the severe leading-edge roughness, or its effect upon the boundary layer, can be eliminated, then the down-stream roughness causes no disturbance to the passage of a laminar layer--i.e., the surface, though roughened, is aerodynamically smooth . moreover, it appears that the conditions defining the upstream boundary to this region of insignificant roughness are fundamentally the same as those which determine the critical state for transition at an artificial disturbance of a three-dimensional character .

.I 934

.T

stability of cylindrical and conical shells of circular cross section, with simultaneous action of axial compression and external normal pressure .

.A

mushtari,k.m. and sachenkov,a.v.

.B

naca tm.1433, 1958.

.W

stability of cylindrical and conical shells of circular cross section, with simultaneous action of axial compression and external normal pressure .

we consider in this report the

determination of the upper limit of

critical loads in the case of simultaneous

action of a compressive force,

uniformly distributed over plane cross

sections, and of isotropic external

normal pressure on cylindrical or

conical shells of circular cross

section . as a starting point we use

the differential equations for neutral

equilibrium of conical shells (ref. 1)

which have been used for the

solution of the problem of stability of conical

shells under torsion and under

axial compression (ref. 2),. upon solution

of the problem it is possible

to satisfy all boundary conditions, in

contrast to the report (ref. 3)

where no attention is paid to the fulfillment

of the boundary conditions

and to the report (ref. 4) where only part

of the boundary conditions are

satisfied by solution of the problem

according to galerkin's method .

approximate formulas are used for the

determination of the critical

external normal pressure with simultaneous action of longitudinal

compression . let us note that the formulas

suggested in reference 5 are

not well founded and may lead, in a number

of cases, to a substantial

mistake in the magnitude of the critical load .

.I 935

.T

buckling of thin single- and multi-layer conical and cylindrical

shells with rotationally symmetric stresses .

.A

p. p. radkowski

.B

avco manufacturing corporation

.W

buckling of thin single- and multi-layer conical and cylindrical

shells with rotationally symmetric stresses .

the buckling of simply supported, thin, single- and multi-layer conical

shells under axially symmetrical loading is analyzed in this paper .

the results are presented in a compact manner so that they may be easily

used for design and/or experimental purposes . the results are compared

with known experimental values .

.I 936

.T

a donnell-type theory for asymmetrical bending and

buckling of thin conical shells .

.A

seide,p.

.B

j. app. mech. 24, 1957, 547.

.W

a donnell-type theory for asymmetrical bending and

buckling of thin conical shells .

equations, somewhat more accurate than those recently

presented by n. j. hoff, are derived for bending and

buckling of thin circular conical shells under arbitrary loading .

these equations reduce to donnell's equations for thin

cylindrical shells when the cone semivertex angle becomes

very small and the minimum radius of curvature of the

median surface approaches a constant value . at the

other end of the scale the equations reduce to the

well-known equations for flat circular plates when the cone

semivertex angle approaches a right angle . in addition,

for the entire range of cone semivertex angles the

equations reduce to the known equations for axisymmetrical

bending when variations of the displacements around the

circumference vanish . the problem of bending is

reduced to the solution of a single fourth-order partial

differential equation with variable coefficients .

.I 937

.T

on the buckling of truncated conical shells in torsion .

.A

seide,p.

.B

j.app.mech.28, 1962.

.W

on the buckling of truncated conical shells in torsion .

the problem of the buckling of thin circular

conical frustums in pure torsion is solved in

a manner similar to that employed previously

by the author for buckling under uniform

hydrostatic pressure . synthesis of the numerical

results indicates that the critical torsion

of a truncated cone is equal to that of an

equivalent cylinder whose length and thickness

are the axial length and wall thickness of

the cone and whose radius is a function of the

semivertex angle and the taper ratio of the

cone . curves and equations to aid in the

analysis of conical frustums are given .

it is shown that a previous recommendation for

the analysis of truncated cones in torsion

may be seriously unconservative in some cases .

.I 938

.T

calculations for the stability of thin conical frustums

subjected to external uniform hydrostatic pressure

and axial loads .

.A

seide,p.

.B

j. ae. scs. 29, 1962, 951.

.W

calculations for the stability of thin conical frustums

subjected to external uniform hydrostatic pressure

and axial loads .

calculations are presented for the problem of the stability of

conical shells subjected to combined external uniform

hydrostatic pressure and axial tension or compression . stress

interaction curves are found to vary only slightly as a function of the

ratio of the end radii of the cone .

.I 939

.T

some explicit solutions for constant-temp . magnetogas

dynamic channel flow .

.A

berlot,r.r.

.B

j. ae. scs. 29, 1962, 748.

.W

some explicit solutions for constant-temp . magnetogas

dynamic channel flow .

in order to simplify the process of estimating the aerodynamic

loading on the after portions of slender vehicles, it is

frequently assumed that there is no nose-tail interaction . it is

the purpose of this note to show that, aside from boundary-layer

effects, this assumption is not warranted when the nose hypersonic-similarity parameter, tan, is of the order of unity, or

greater . physically speaking, the entropy change associated

with a strong bow wave reduces the stagnation pressure

down-stream of the shock, and hence, lowers the dynamic pressure in

the vicinity of the tail .

.I 940

.T

of a turbulent free shear layer .

.A

nash,j.f.

.B

npl. aero. r1019, 1962.

.W

of a turbulent free shear layer .

  the problem of predicting the mean velocity on streamlines

through the pre-asymptotic turbulent free shear layer in two-dimensional

incompressible flow is resolved into two parts . the linearized

momentum equation in terms of a generalized axial co-ordinate is

solved in the usual way . a relation between and the distance from

the separation point is then established analytically in contrast to the

previous use of empirical expressions .

  it is shown that except in the region close to separation the

velocity on the streamlines can be predicted by the simple approximation

proposed by kirk .

.I 941

.T

viscous compressible and incompressible flow in slender channels .

.A

williams,j.c.

.B

a.i.a.a. j. 1963, 186.

.W

viscous compressible and incompressible flow in slender

channels .

an analytical study is made of viscous flow in slender

channels . similar solutions to the approximate equations of motion,

valid for flow at moderate or high reynolds numbers in slender

channels, are found for incompressible two-dimensional and

axisymmetric flows and for compressible flows through

two-dimensional channels with adiabatic walls . a study of

compressible flows in convergent-divergent channels yields results

regarding the effect of viscosity on the location of the sonic line,

on the pressure ratio at the geometric throat and on the discharge

coefficient for such channels .

.I 942

.T

secondary gas injection in a conical rocket nozzle .

.A

walker,r.e., stone,a.r. and shandor,m.

.B

aiaa jnl. u, 1963, 334.

.W

secondary gas injection in a conical rocket nozzle .

data are presented on side forces generated
by secondary gas injection in a 15 conical
rocket exhaust nozzle . the side force was measured
directly with a force transducer and the
data examined in terms of an amplification
factor, where is the measured effective
specific impulse of injectant, and is the
specific impulse of injectant for sonic flow into a
vacuum . injection was normal to the axis
of the nozzle through a single circular orifice at a
fixed point in the diverging portion of the nozzle .
a variety of ambient temperature gaseous
injectants and orifice diameters were carefully studied .
injectant flow rate was varied for each configuration .
the main propellant was hot gas (
catalytically decomposed), and motor conditions
were held essentially constant .

.I 943

.T

compressible free shear layer with finite initial thickness .

.A

denison,m.r. and baum,e.

.B

aiaa jnl. 1, 1963, 342.

.W

compressible free shear layer with finite initial thickness .

the momentum equation was uncoupled from the other conservation equations for the case

of a finite initial profile in a laminar free shear layer .

the equation was solved numerically, in

the crocco coordinate system, using an implicit

finite difference method . profiles of velocity

and shear function were obtained as a function of

streamwise distance . the initial profiles as

the flow separates from the rear of the body

correspond to the blasius profile in transformed

coordinates . for large distances downstream,

the profiles approach the chapman

distribution, corresponding to the case of zero initial

free shear layer thickness . the effect of these

results on calculations of base pressure and wake

angle is discussed . a method for the

calculation of finite chemical kinetic effects on the

profiles of temperature and chemical

composition in the free shear layer with finite initial

thickness is outlined .

.I 944

.T

one dimensional heat conduction through the skin of

a vehicle upon entering a planetary atmosphere at constant

velocity and entry angle .

.A

wells,w.r. and mclellan,c.h.

.B

nasa tn.d1476, 1962.

.W

one dimensional heat conduction through the skin of

a vehicle upon entering a planetary atmosphere at constant

velocity and entry angle .

closed-form solutions of the

one-dimensional heat-conduction

equations for the flow of heat into a

plate with a laminar boundary layer

have been obtained for a configuration

entering a planetary atmosphere

with constant velocity and negative

entry angle . the atmospheric density

was assumed to obey an exponential law

and the temperature was assumed

constant initially . the solution is in

the form of a fourier series

expansion which, for most practical

applications, can be approximated by

retaining only one term of the expression .

the solution applies to the

initial part of the entry before the maximum

heating conditions are

encountered .

.I 945

.T

method for design of pump impellers using a high speed digital computer .

.A

stockman,n.o. and kramer,j.l.

.B

nasa tn.d1562, 1963.

.W

method for design of pump impellers using a high speed

digital computer .

a method of designing pump impellers

is derived from the equations of motion

and continuity for incompressible nonviscous

relative flow . the flow is assumed

to follow a known stream surface (representing

blade shape) that extends from hub

to shroud . equations are also derived for

approximate blade-surface velocities

and pressures . a detailed numerical procedure

and block diagram are given for

use on a digital computer . a numerical example

that illustrates limited use of

the method is presented and further uses are

indicated .

.I 946

.T

exploratory investigation of the effect of a forward

facing jet on the bow shock of a blunt body in a mach

number 6 free stream .

.A

romeo,d.j. and sterrett,j.r.

.B

nasa tn.d1605, 1963.

.W

exploratory investigation of the effect of a forward

facing jet on the bow shock of a blunt body in a mach

number 6 free stream .

the effect of a forward-facing jet on

the bow shock of a blunt body in a

mach 6 free stream was investigated

experimentally . the models tested had

forward-facing jets using air and helium exhausting

at mach numbers from 1 to 10.3 and

were run through a range of the ratio of jet

total pressure to free-stream total

pressure of 0.03 (jet off) to 2.5 . the ratio

of body diameter to jet-exit

diameter varied from 1.12 to 55.6 and the angle

of attack was varied from 0 to 35 .

the experimental results show that the

main-stream shock can be affected by

the jet in two significantly different ways .

one way is simply to move the strong

shock away from the body without altering its

shape . the second and perhaps more

interesting case occurs when the jet causes a large displacement of the main shock and considerably changes its shape . it was found that the ratio of jet total pressure to free-stream total pressure necessary to obtain the large displacements of the main-stream shock depended on the ratio of body diameter to jet-exit diameter and also on the jet-exit mach number . the maximum amount the shock could be displaced in percent of body diameter was seen to increase with increasing jet-exit mach number and also with decreasing ratio of body diameter to jet-exit diameter . for the models that were investigated through an angle-of-attack range, the displacement became very unsteady and fell off sharply as the angle of attack was increased .

simplified theoretical considerations applied to the shock-displacement phenomena provide a possible explanation for the two different types of main-stream shock displacement . theoretical curves show the regions where these types of displacement would occur for different exit mach numbers and pressure ratios for a forward-facing jet in a mach 6 stream .

.I 947

.T

static aerodynamic characteristics of a short blunt

10 semi-vertex angle cone at a mach number of 15 in

helium .

.A

fohrman,m.j.

.B

nasa tn.d1648, 1963.

.W

static aerodynamic characteristics of a short blunt

10 semi-vertex angle cone at a mach number of 15 in

helium .

axial force, normal force, pitching

moment, and shock-wave shape were

determined for a body of revolution

consisting of a short blunt 10 semivertex

angle cone with a flat base and also with

a conical afterbody having a

semi-vertex angle of 50 . measurements were

made in helium at a free-stream mach

number of 15 and a free-stream reynolds

number of 2.25x10 based on maximum body

diameter over an angle-of-attack range from

the configuration with the conical

afterbody was statically stable in the

nose-forward attitude only, whereas the

configuration with no afterbody was

statically stable in both the nose-forward

and base-forward attitudes . the force

and moment data of both shapes were predicted

reasonably well by modified

newtonian theory at all angles of attack,

except the pitching-moment coefficient

for the model without afterbody near 180

angle of attack . in this region,

measurements indicated static stability, whereas

theory indicated static instability .

the helium data agreed reasonably well with

a limited amount of force and moment

data obtained in a ballistic range at small

angles of attack in air at a mach

number of 15 and also with force and moment

data obtained in air over a complete

angle-of-attack range at a mach number of 5.5 .

the value of axial-force

coefficient and the shape of the bow shock wave at

zero angle of attack for both models

obtained from a numerical flow field calculation

agreed very well with the data .

the value of the axial force coefficient at 180

angle of attack for the model

with afterbody agreed reasonably well with the

theoretical value for a cone . the

position and shape of the shock envelope near

the stagnation point also could be

predicted accurately by an approximate method

over an angle-of-attack range from

.I 948

.T

panel flutter tests on full scale x-15 lower vertical

stabilizer at mach number of 3. 0.

.A

bohon,h.l.

.B

nasa tn.d1385, 1962.

.W

panel flutter tests on full scale x-15 lower vertical

stabilizer at mach number of 3. 0.

panel flutter tests were conducted

on two full-scale vertical

stabilizers of the x-15 airplane at a

mach number of 3.0 in the langley

at dynamic pressures from 1,500 psf

to 5,000 psf and stagnation temperatures

from 300 f to 660 f . flutter

boundaries were obtained for four of

the five distinct types of panels

which make up the vertical sides of

the stabilizers . the boundaries

consisted of a flat-panel boundary

and a thermally buckled-panel

boundary . the flat-panel boundaries were

characterized by a reduction in

dynamic pressure with increasing skin

temperature,. whereas, after thermal

buckling the trend was reversed . the

minimum dynamic pressure for

flutter occurred at the intersection of

the flat-panel and buckled-panel

boundaries and represented a large

reduction in the dynamic pressure

over the extrapolated, unstressed value .

as a result of panel flutter,

three of the five distinct types of

panels were modified to provide the

required flutter margin on the design

flight dynamic pressure of the

aircraft .

.I 949

.T

charts for equilibrium flow properties of air in

hyper-velocity nozzles .

.A

jorgensen,l.h. and baum,g.m.

.B

nasa tn.d1333, 1962.

.W

charts for equilibrium flow properties of air in
hyper-velocity nozzles .
for initial stagnation pressures
up to 1,000 atmospheres and
stagnation enthalpies up to 10,000 btu per
pound, nozzle-flow properties for
equilibrium air have been computed and
plotted on charts . the work of
nasa tn d-693 has been extended to
include flow properties for closer
intervals of specified stagnation
enthalpies . properties which have been
charted as a function of mach number
are as follows .. temperature,
pressure, density, velocity, area ratio,
dynamic pressure, reynolds number,
isentropic exponent, and molecular weight
ratio . ratios of temperature,
pressure, and density across normal shock
waves are also charted, and
weight-flow rate is plotted as a function
of stagnation enthalpy .

.I 950

.T

comparison of theoretical and experimental creep buckling
times of initially straight, centrally loaded columns .

.A

jahsman,w.e. and field,f.a.

.B

j. ae. scs. 29, 1962, 431.

.W

comparison of theoretical and experimental creep buckling

times of initially straight, centrally loaded columns .

the creep-buckling times of initially straight, centrally loaded

columns as predicted by the hypotheses of shanley, gerard,

and rabotnov and shesterikov are compared with appropriate

experimental data . it is found that the theoretical predictions

are generally conservative, due possibly to the fact that the

hypotheses predict initial instability times while the experiments

normally record final collapse times . of the three hypotheses,

that of gerard generally gives predictions which agree best with

the experimental data .

.I 951

.T

a unified theory of creep buckling under normal loads .

.A

stowell,e.z. and thein wah.

.B

j. ae. scs. 29, 1962, 658.

.W

a unified theory of creep buckling under normal loads .

a general theory of creep buckling, with the initial

imperfection as a parameter, is developed for the case of normal loading .

a hyperbolic-sine law is used to describe the process of creep . the

theory is believed to be applicable to, among other structures,
columns, tubes, and possibly conical shells . the wall of the
structure is idealized as a sandwich in order to simplify the
integration of the equations .

experimental data on columns and tubes, from two different
sources, are compared with the predictions of the theory .

.I 952

.T

study of creep collapse of a long circular cylindrical
shell under various distributed force systems .

.A

serpico,j.c.

.B

j. ae. scs. 29, 1962, 1316.

.W

study of creep collapse of a long circular cylindrical
shell under various distributed force systems .

an analysis is presented for determining the collapse of circular
rings and long cylinders subjected to primary and secondary creep
conditions at elevated temperatures . the types of loading
considered for the present investigation are dead loading and
hydrostatic pressure-type forces . the method of solution is based on
an application of the variational theorem for creep described in
ref. 1 with some additional terms being introduced for the
pressure-type loading case . the general results are reduced to a
relatively simple form for the theoretical predictions of collapse
time and are graphically illustrated for a typical sample material .

.I 953

.T

vibrations of infinitely long cylindrical shells under

initial stress .

.A

armenakas,a.e.

.B

aiaa jnl. 1, 1963, 100.

.W

vibrations of infinitely long cylindrical shells under

initial stress .

the general bending theory of shells under the influence of

initial stress presented recently by herrmann and armenakas

is applied in this investigation to study the effect of initial

uniform circumferential stress, uniform bending moment and

uniform radial shear on the dynamic response of an infinitely long

cylindrical shell .

.I 954

.T

analysis of stress at several junctions in pressurized

shells .

.A

johns,r.h., morgan,w.c. and spera,d.a.

.B

aiaa jnl. 1, 1963, 455.

.W

analysis of stress at several junctions in pressurized shells .

theoretical and experimental results are

presented for the discontinuity stresses arising at a

change of wall thickness in a cylinder, a

cylinder-hemisphere junction, and a cone-spherical torus

junction in pressure vessels . the effect of mismatch

of nonconcurrence of the middle surfaces of two

joined cylinders is considered . in addition, a

cylinder with a special closure which has considerably

reduced stresses is described, and curves with

theoretical and experimental stresses are presented .

.I 955

.T

the membrane approach to bending instability of pressureized

cylindrical shells .

.A

mccomb,h.g., zender,g.w. and mikulas,m.m.

.B

nasa tn.d1510, 1962, 229.

.W

the membrane approach to bending instability of pressureized

cylindrical shells .

recent theoretical and experimental

research is briefly described

to trace the development of deformation

and the occurrence of collapse in

pressurized circular cylindrical membranes under applied moment loading .

the collapse of pure membrane cylinders

is then compared with instability

of pressurized cylindrical shells .

this approach leads to a better

understanding of the behavior of pressurized

cylinders under bending loads .

the results suggest possibilities for

further research utilizing the

membrane approach .

.I 956

.T

elastic stability of simply supported corrugated core

sand- wich cylinders .

.A

harris,l.a. and baker,e.g.

.B

nasa tn.d1510, 1962, 331.

.W

elastic stability of simply supported corrugated core

sand- wich cylinders .

theoretical buckling coefficients

are obtained for the general

instability of simply supported, corrugated

core sandwich circular

cylinders under combined loads with the core

oriented parallel to the

longitudinal axis of the cylinder . buckling
curves are presented for axial
compression, external lateral pressure,
torsion, and some typical
interactions . the differential equations of
equilibrium used to obtain the
buckling equations were derived from the
small deflection equations of
stein and mayer which include the effect
of deformation due to
transverse shear . these equations are solved
by galerkin's equation .
remarks are made concerning the probable
validity of the results of the
small deflection theory for sandwich shells .
.I 957
.T
axisymmetric snap buckling of conical shells .
.A
newman,m. and reiss,e.l.
.B
nasa tn.d1510, 1962, 451.
.W
axisymmetric snap buckling of conical shells .
  the authors give a brief account
of some of their recent analytical and
numerical studies of cone buckling,

limiting the discussion to axisymmetric

deformations .

pertinent numerical results for the

relaxation buckling of full cones

subjected to uniform external pressure and

belleville springs deformed by axial

edge loads are presented . in addition,

bifurcation buckling problems are

discussed . for a specific case, the existence

of friedrichs' intermediate buckling

load, as applied to cones, is established .

upper and lower bounds for

its value are given .

.I 958

.T

air scooping vehicle .

.A

berner,f. and camac,m.

.B

avco-everett res. lab. r.76, 1959.

.W

air scooping vehicle .

a satellite vehicle is described

which collects gases from the

upper atmosphere and stores them in

liquid form . such a vehicle

could serve as a filling station in space, furnishing liquid oxygen or

air to other spacecraft . the vehicle

represents an alternative to

launching these liquids into orbit from

the surface of the earth . the

two methods are compared on an

economic basis, and it is shown that

the proposed vehicle permits substantial

savings when operated beyond

about one year . the feasibility of

developing such a system for

long-time operation is investigated . several

practical designs are discussed .

.I 959

.T

heat transfer in separated flows .

.A

larson,h.k.

.B

j. ae. scs. 1959, 731.

.W

heat transfer in separated flows .

results of an experimental heat-transfer investigation in

regions of separated flow are presented and compared with the

theoretical analysis of naca tn 3792 . the average heat

transfer for both laminar and turbulent separated boundary

layers was found to be from 35 to 50 per cent less than that for equivalent attached boundary layers . the overall scope of the

measurements included mach numbers from 0.3 to 4.0 and

reynolds numbers from 10 to 4 x 10 . the results for laminar

boundary layers agree well with the analysis of tn 3792 . the

results for turbulent boundary layers, however, disagree

considerably . results of velocity and temperature surveys in the

separated turbulent boundary layer are presented and partially

explain the discrepancy between the experiments and analysis .

the maximum local heat-transfer rates were found to occur in

the reattachment region of the separated boundary layers

investigated . the effect of transition on heat transfer in the separated

laminar boundary layers is described and data showing effects

of mach number and wall temperature on the transition

reynolds number of separated laminar flows are also included .

.I 960

.T

investigation of free turbulent mixing .

.A

liepman,h.w. and laufer,j.

.B

naca tn.1257, 1947.

.W

investigation of free turbulent mixing .

a discussion of the integral

relations for flow of the

boundary-layer type is presented . it is

shown that the characteristic laws of

spread of jets, wakes, and so forth,

can be obtained directly for the

laminar case and, with the help of

dimensional reasoning, for the

turbulent case as well .

measurements of the mean velocity,

the intensity and scale of the

turbulent fluctuations, and of the turbulent

shear in a two-dimensional

mixing zone are presented . the results of

these measurements are

compared with the mixing-length theories . it

is shown that both mixing

length and exchange coefficient vary across

the mixing zone . the

theories based on the assumption of constant

mixing length or exchange

coefficient are thus in error .

a discussion of the energy balance of

the fluctuating motion is

given and the triple correlation is estimated .

.I 961

.T

compressible two dimensional jet mixing at constant

pressure .

.A

korst,h.h. and page,r.h.

.B

univ. of illinois dept. mech. eng. me-tn-392-1, 1954.

.W

compressible two dimensional jet mixing at constant

pressure .

an analysis is made of turbulent constant pressure mixing for a

compressible jet boundary, taking into consideration effects of the

initial boundary layer . velocity profiles in the mixing region are

represented in a transformed plans by one-parameter families of curves,

with no specification for the mixing mechanism beyond that of an

exchange coefficient concept being made . the exchange coefficient is

represented by the bornel function of an integral transform for the x

coordinate of an intrinsic system of coordinates . this intrinsic

system and the physical coordinate system are related by means of a

momentum integral .

satisfactory correlation of theory and experimental low-speed data

is obtained with a simple form of kernal function .

an asymptotic solution, corresponding to a fully developed velocity

profile in the jet boundary, allows the calculation of the mechanical

energy level along the separating streamline in the jet boundary without

the use of empirical information .

.I 962

.T

contributions to the theory of heat transfer through

a laminar boundary layer .

.A

m. j. lighthill

.B

department of mathematics, university of manchester

communicated by s. goldstein, f.r.s.

.W

contributions to the theory of heat transfer through

a laminar boundary layer .

an approximation to the heat transfer rate across a laminar

incompressible boundary layer, for arbitrary distribution of

main stream velocity and of wall temperature, is obtained by

using the energy equation in von mises's form, and approximating the

coefficients in a manner which is most closely correct near the

surface . the heat transfer rate to a portion of surface of length l

breadth is given as

where k is the thermal conductivity of the fluid, o its prandtl number,

p its density, u its viscosity, r(x) is the skin friction,

and t(x) the excess of wall temperature over main stream temperature .

a critical appraisement of the formula indicates that it should be very

accurate for large, but that for of order 0.7 (for most gases) the

constant should be replaced by 0.73, when the error should not exceed

this yields a formula for nusselt number in terms of the reynolds number

r and the mean square root of the skin friction coefficient c, in the

case of uniform wall temperature .

however, for the boundary layer with uniform main stream, the

original formula is accurate to within 3 percent even for . by

known transformations an expression is deducted for heat

transfer to a surface, with arbitrary temperature distribution

along it, and with a uniform stream outside it at arbitrary mach number (equation (42)) . from this the temperature distribution along such a surface is deduced in the case (of importance at high mach numbers) when heat transfer to it is balanced entirely by radiation from it . this calculation, which includes the solution of a non-linear integral equation, gives higher temperatures near the nose, and lower ones farther back (figure 2), than are found from a theory which assumes the wall temperature uniform and averages the heat transfer balance . this effect will be considerably mitigated for bodies of high thermal conductivity., the author is not in a position to say whether or not it will be appreciable for metal projectiles . but for stony meteorites at a certain stage of their flight through the atmosphere it indicates that melting at the nose and re-solidification farther back may occur, for which the shape and constitution of a few of them affords evidence . an appendix shows how the method for approximating and solving von mises's equation could be used to determine the skin friction as well as heat transfer rate, but this line seems to have no advantage over established approximate methods .

.I 963

.T

a variational principle for convection of heat .

.A

agrawal, h.c.

.B

j. math. and mech. v.9, no. 6, 1960 .

.W

a variational principle for convection of heat .

authors extend variational principle of biot to various cases of heat

transfer due to forced convection . numerical results are given for

one-dimensional problems of fluid flowing between parallel walls with

uniform or parabolic velocity profiles . agreement with exact solutions

is excellent .

.I 964

.T

on the theory of discharge coefficients for round entrance flowmeters

and venturis .

.A

rivas, m.a. and shapiro, a.h.

.B

trans. a.s.m.e., v. 78, april 1956, pp 489-497 .

.W

on the theory of discharge coefficients for round entrance flowmeters

and venturis .

a theory of rounded-entrance flowmeters, based on a consideration of the

potential and boundary-layer flows in a converging nozzle, is

constructed . curves are presented showing the discharge coefficient as a

function of diameter reynolds number, with the /total equivalent length

tional length-diameter ratio of the contraction section of the asme

long-radius nozzle is presented . the theoretical curves of discharge

coefficient versus diameter reynolds number are in good agreement with

experiment over a range of reynolds number from 1 to 10 . the theory

provides a rational framework for correlating and extrapolating

experimental results,. it shows the effects of contraction shape and

location of pressure taps,. it furnishes values of discharge coefficient for untested designs,. and it suggests precautions to be taken in

design, installation, and operation .

.I 965

.T

analytic determination of discharge coefficients of flow nozzles .

.A

simmons, f.s.

.B

naca tn 3449, 1955 .

.W

analytic determination of discharge coefficients of flow nozzles .

integration of the velocity profile at the throat of a flow nozzle

yields the discharge coefficient as a function of the ratio of boundary

solution of the approximate momentum equation for the boundary layer .

the resulting expression for the discharge coefficient is then a

function of the reynolds number based on nozzle diameter and of the geometry

of the nozzle . good agreement is shown between this expression and

published experimental data on flow nozzles for reynolds numbers between

.I 966

.T

on fully developed channel flows,. some solutions and limitations, and

effects of compressibility, variable properties, and body forces .

.A

maslen, s.h.

.B

nasa tr r-34, 1959

.W

on fully developed channel flows,. some solutions and limitations, and effects of compressibility, variable properties, and body forces .

an examination of the effects of compressibility, variable properties, and body forces on fully developed laminar flows has indicated several limitations on such streams .

in the absence of a pressure gradient, but presence of a body force liquid this follows also for the case of a constant streamwise pressure gradient . these motions are exact in the sense of a couette flow . in the liquid case two solutions /not a new result/ can occur for the same boundary conditions . an approximate analytic solution was found which agrees closely with machine calculations .

in the case of approximately exact flows, it turns out that for large temperature variations across the channel the effects of convection /due to, say, a wall temperature gradient/ and frictional heating must be negligible . in such a case the energy and momentum equations are separated, and the solutions are readily obtained . if the temperature variations are small, then both convection effects and frictional heating can consistently be considered . this case becomes the constant-property incompressible case /or quasi-incompressible case for free-convection flows/ considered by many authors .

finally, there is a brief discussion of cases wherein streamwise variations of all quantities are allowed but only in such form that the independent variables are separable . for the case where the streamwise velocity varies inversely as the square root of distance along the channel, a solution is given .

.I 967

.T

a study of laminar compressible viscous pipe flow accelerated by an

axial body force, with application to magnetogasdynamics .

.A

martin, d.e.

.B

nasa tn d-855, april 1961 .

.W

a study of laminar compressible viscous pipe flow accelerated by an

axial body force, with application to magnetogasdynamics .

a study is made of the steady laminar flow of a compressible viscous

fluid in a circular pipe when the fluid is accelerated by an axial

body force . the application of the theory to the magnetofluidmechanics

of an electrically conducting gas accelerated by electric and magnetic

fields is discussed . constant viscosity, thermal conductivity, and

electrical conductivity are assumed . fully developed flow velocity and

temperature profiles are shown, and detailed results of the accelerating

flow development, including velocity and pressure as functions of

distance, are given for the case where the axial body force is constant and

for the case where it is a linear function of velocity . from these

results are determined the pipe entry length and the pressure difference

required .

.I 968

.T

rocket propulsion systems for interplanetary flight .

.A

sutton, g.p.

.B

j. aero. sc. v. 26, october 1959 .

.W

rocket propulsion systems for interplanetary flight .

a comparison is made of several different propulsion systems for

interplanetary flight . liquid and solid propellant rockets, propulsion

systems which use nuclear energy sources, are heating rockets,

magneto-plasma devices, ion rocket propulsion, solar heating rockets, and solar

sails are briefly described and their current status reviewed . engine

performance requirements for different interplanetary missions are

established . these several propulsion systems are then compared on the

basis of several performance criteria, environmental characteristics,

vehicle requirements, reliability, current status, growth potential, and

efficiency . predictions on various propulsion system capabilities and

an analysis of multiple rocket engine reliability is included . it is

concluded that electrical rockets are superior for long-time

inter-planetary flight applications, and that chemical rockets are

satisfactory for most of the immediate applications in /near/ space . none of

the several propulsion schemes discussed can be rejected until further

technical work has been accomplished .

.I 969

.T

on the use of side-jets as control devices .

.A

liepman,h.p.

.B

ars jnl. 29, 1959, 453.

.W

on the use of side-jets as control devices .

wind tunnel experiments with side-jets, issuing laterally

near the base of slender bodies in a supersonic stream, have

suggested the existence of a sizable and usable interaction .

with this interaction force, the use of jet reaction controls

may be as attractive for flight within the atmosphere as it

obviously is for flight outside the atmosphere . this note

indicates the altitude regime of interest and the order of

magnitude of the interaction bonus for a lateral control

jet located near the base of a body of revolution .

.I 970

.T

loads induced on a flat plate wing by an air jet exhausting

perpendicularly through the wing and normal to a free-stream flow of mach

number 2.0 .

.A

janos, j.j.

.B

nasa tn d-649, march 1961 .

.W

loads induced on a flat plate wing by an air jet exhausting

perpendicularly through the wing and normal to a free-stream flow of mach

number 2.0 .

measurements were made of loads induced on a flat-plate wing by an air

jet exhausting perpendicularly through the wing and normal to the

free-stream flow . the investigation was conducted at a free-stream mach

number of 2.0 and a reynolds number per foot of 14.4 x 10 . an axially symmetric sonic nozzle and two supersonic nozzles were employed for the

jets . the supersonic nozzles consisted of an axially symmetric nozzle

with exit mach number of 3.44 and a two-dimensional nozzle with exit

mach number of 1.76 . the ratio of nozzle total pressure to free-stream

static pressure was varied from 20 to 110 .

negative loads were induced on the flat-plate wing by all the jets . as

the nozzle pressure ratio was increased the magnitude of interference

loads due to jet thrust decreased . the chordwise center-of-pressure

location generally moved toward the nozzle center line as the pressure

ratio was increased .

.I 971

.T

surface pressure distributions with a sonic jet normal to adjacent flat

surfaces at mach 2.92 to 6.4 .

.A

cubbison, r.w., anderson, b.h. and ward, j.j.

.B

nasa tn d-580, february 1961 .

.W

surface pressure distributions with a sonic jet normal to adjacent flat

surfaces at mach 2.92 to 6.4 .

an investigation was made to determine the interference effects on

surface pressure distributions caused by a sonic jet exiting normal to the

surface . two configurations, a flat plate and an arrow-wing

reentry-type vehicle, with sonic nozzles near the leading edge were tested over

a range of pressure ratios and reynolds numbers for mach numbers from

the data indicate that jet pressure ratio had considerable effect on the pressure levels and distributions on both configurations . also, for a constant jet pressure ratio, the free-stream mach number effect on the distributions and levels was quite large . over the limited range investigated, the effect of reynolds number at constant mach number and pressure ratio was small compared to the mach number and pressure ratio effect .

.I 972

.T

aerodynamic interaction effects ahead of a sonic jet exhausting perpendicularly form a flat plate into a mach number 6 free stream .

.A

romeo, d.j. and sterrett, j.r.

.B

nasa tn d-743, april 1961 .

.W

aerodynamic interaction effects ahead of a sonic jet exhausting perpendicularly form a flat plate into a mach number 6 free stream .

an investigation of the effects of the interaction ahead of a two-dimensional sonic jet exhausting perpendicularly into a mach number were made at an angle of attack of 0degree at a reynolds number per foot of approximately 6 x 10 and with conditions of both transitional and turbulent separation on the flat plate . the ratio of jet stagnation pressure to free-stream static pressure was varied from 8 to 460 and the jet slot width was varied from 0.001 to 0.05 inch . the force ratio due to reaction of jet/, calculated ahead of the jet, was sizable and varied from 0.5 to 9 . in general, the ratio increased with increasing pressure ratio and decreasing slot width . for the turbulent boundary-layer separation tests it was found that the first peak pressure and the

chordwise pressure distribution of the separated boundary layer ahead

of the jet were similar to those for a separation caused by a

forward-facing step at the same test conditions .

.I 973

.T

interaction effects produced by jet exhausting laterally near base of

ogive-cylinder model in supersonic main stream .

.A

vinson, p.w., amick, j.l. and liepman, h.p.

.B

nasa memo 12-5-58w, february, 1959 .

.W

interaction effects produced by jet exhausting laterally near base of

ogive-cylinder model in supersonic main stream .

the experimentally determined interaction effects of a side jet

exhausting near the base of an ogive-cylinder model are presented and

discussed . the interaction force appears to be independent of

main-stream mach number, boundary-layer condition /laminar or turbulent/,

angle of attack, and forebody length . the ratio of interaction force to

jet force is found to be inversely proportional to the square root of

the product of jet stagnation-to-free-stream pressure ratio and jet-to-

body diameter ratio .

.I 974

.T

approximate analysis of thrust vector control by fluid injection .

.A

wu, j.m., chapkis, r.l. and mager, a.

.B

ars journal, 1677-1685, 1961 .

.W

approximate analysis of thrust vector control by fluid injection .

a study has been made of the side force generated by injection of secondary material into the main stream of a rocket nozzle . two cases have been analyzed .. gas injection and liquid injection . for the gas injection case, it is assumed that the turbulent boundary layer ahead of the injection point separates from the wall . the pressure in the separated region and the extent of the separated region are determined by a consideration of turbulent boundary layer-shock wave interaction and the accommodation height of the injected gas stream . equations are derived for calculating the side force, and the side forces predicted by the theory are compared with experimental data . the agreement between theory and experiment is fair . for the case of liquid injection, it is assumed that the liquid flows along the nozzle wall and evaporates into the main stream . the resulting side force on the nozzle wall is determined on the basis of linearized theory, thus restricting the analysis to small rates of liquid injection . the effects of small rates of heat addition are also included in the analysis . a very simple equation for calculating the side force is obtained .

.I 975

.T

one dimensional flows of an imperfect diatomic gas .

.A

eggers,a.j.

.B

naca r.959, 1950.

.W

one dimensional flows of an imperfect diatomic gas .

with the assumptions that berthelot's equation of state accounts for molecular size and intermolecular force effects, and that changes in the vibrational heat capacities are given by a planck term, expressions are developed for analyzing one-dimensional flows of a diatomic gas .

the special cases of flow through normal and oblique shocks in free air at sea level are investigated . it is found that up to a mach number of 10 the pressure ratio across a normal shock differs by less than 6 percent from its ideal gas value,. whereas at mach numbers above 4 the temperature rise is considerably below and hence the density rise is well above that predicted assuming ideal gas behavior . it is further shown that only the caloric imperfection in air has an appreciable effect on the pressures developed in the shock process considered . the effects of gaseous imperfections on oblique shock flows are studied from the standpoint of their influence on the lift and pressure drag of a flat plate operating at mach numbers of 10 and 20 . the influence is found to be small .

.I 976

.T

turbulent diffusion in the wake of a blunt nosed body at hypersonic speeds .

.A

lees,l. and hromas,l.

.B

j. ae. scs. 29, 1962, 976.

.W

turbulent diffusion in the wake of a blunt nosed body

at hypersonic speeds .

at reynolds numbers greater than about 5 x 10

corresponding to altitudes below about 180,000 ft, the hot outer inviscid

wake behind the bow shock wave produced by a blunt-nosed

body at hypersonic speeds is cooled mainly by turbulent diffusion

and conduction . turbulence originates in the inner wake formed

by the coalescence of the free shear layers (or annulus) shed from

the body surface when the boundary layer separates from the

surface . as this turbulence spreads outward, it swallows

enthalpy or momentum defect originally contained in the outer

inviscid wake . if the turbulence is locally similar--i.e., if it

behaves at each station like a slice of a low-speed /self-similar/

wake--then the turbulent diffusivity grows from a low initial

value near the body to a value corresponding to the total drag of

the body at about 300 body diameters downstream . at flight

velocities of the order of 9,000-10,000 ft per sec. the growth of the

turbulent inner wake predicted on the basis of locally similar

turbulence is in good agreement with shadowgraph measurements

of wake widths behind spheres obtained in ballistic ranges in the

region from 200 to 4,000 body diameters downstream of the body .

tentatively, one concludes that the turbulence mechanism in the

wake with respect to a fixed observer is similar to the low-speed case, in spite of the large mean temperature gradients .

in order to illustrate the behavior of an observable such as electron density in a turbulent wake behind a blunt body, the two limiting cases of thermodynamic equilibrium and pure diffusion (zero electron-ion recombination rate) are calculated for m = 22 at altitudes of 100,000 and 200,000 ft . even for the case of thermodynamic equilibrium, the predicted turbulent radar trail length is about 200 body diameters at l-band (1,300 mc) at 100,000-ft altitude and about 150 body diameters for uhf (400 mc) at 200,000 ft . one interesting result is that the width of the plasma cylinder corresponding to the plasma requency at l-band remains virtually constant at about 3.5 body diameters in the range 30 150 at 100,000-ft altitude .
these results are sufficiently encouraging that one can consider including the effects of finite chemical and electron-ion recombination rates in the analysis in order to give a more complete picture of the wake at hypersonic speeds .

.I 977

.T

concerning some solutions of the boundary layer equations in hydrodynamics .

.A

goldstein,s.

.B

proc. cam. phil. soc. 26, 1930, 18.

.W

concerning some solutions of the boundary layer equations in hydrodynamics .

the boundary-layer equations for

a steady two-dimensional motion

are solved for any given initial

velocity distribution (distribution along a

normal to the boundary wall,

downstream of which the motion is to be

calculated) . this initial velocity

distribution is assumed expressible

as a polynomial in the distance from

the wall . three cases are considered ..

first, when in the initial distribution

the velocity vanishes at the wall,

but its gradient along the normal

does not,. second, when the velocity in

the initial distribution does not

vanish at the wall,. and third, when both

the velocity and its normal gradient

vanish at the wall (as at a point

where the forward flow separates

from the boundary) . the solution is

found as a power series in some

fractional power of the distance along

the wall, whose coefficients are

functions of the distance from the wall to

be found from ordinary differential

equations . some progress is made

in the numerical calculation of

these coefficients, especially in the first

case . the main object was to

find means for a step-by-step calculation

of the velocity field in a boundary

layer, and it is thought that such a

procedure may possibly be successful

even if laborious . the same

mathematical method is used to calculate

the flow behind a flat plate along a

stream . the results are shown in

curves in the original .

.I 978

.T

temperature profiles inafinite solid with moving boundary .

.A

dewey, c.f., schlesinger, s.i. and sashkin, l.

.B

j. aero. sc. v. 27, 1960 .

.W

temperature profiles inafinite solid with moving boundary .

a numerical solution is presented to the transient heat conduction

equation for a cylinder of finite thickness with one moving boundary .

the implicit method of solution is developed with conductivity as an

arbitrary function of temperature . application is made to a sample case

of re-entry heating encountered by aerodynamic bodies, with erosion by

sublimation and combustion occurring at the body surface .

.I 979

.T

correlation of base pressure and wake structure of

sharp and blunt-nose cones with reynolds number based

on boundary layer momentum thickness .

.A

lehnert,r. and schermerhorn,v.l.

.B

j. ae. scs. 26, 1959.

.W

correlation of base pressure and wake structure of

sharp and blunt-nose cones with reynolds number based

on boundary layer momentum thickness .

it has been established in the past that there is a certain

relationship between base pressure and boundary-layer

behavior . the base-pressure and wake-flow conditions were

found to be dependent upon the local flow characteristics at the

surface of supersonic vehicles directly upstream of the base or of

the region of wake-flow separation .

in order to use existing data on cones and other shapes to

predict wake angle and base pressure on blunt bodies, an attempt was

made recently at the naval ordnance laboratory to establish

a unique relationship between given local flow conditions at the

downstream end of sharp and blunt cones at supersonic speeds

and the corresponding wake-flow conditions with zero heat

transfer .

.I 980

.T

a method of computing the transient temperature of thick walls from arbitrary variation of adiabatic-wall temperature and heat-transfer coefficient .

.A

hills, p.r.

.B

naca report 1372, 1958 .

.W

a method of computing the transient temperature of thick walls from arbitrary variation of adiabatic-wall temperature and heat-transfer coefficient .

a method of calculating the temperature of thick walls has been developed in which are used relatively new concepts, such as the time series and the response to a unit triangle variation of surface temperature, together with essentially standard formulas for transient temperature and heat flow into thick walls . the method can be used without knowledge of the mathematical tools of its development . the method is particularly suitable for determining the wall temperature in one-dimensional thermal problems in aeronautics where there is a continuous variation of the heat-transfer coefficient and adiabatic-wall temperature . the method also offers a convenient means for solving the inverse problem of determining the heat-flow history when temperature history is known .

a series of diversified problems were solved by exact analysis as well as by the new method . a comparison of the results shows the new method to be accurate . the labor involved is very modest in consideration of the nature of the thick-wall temperature problem .

limiting solutions for the /infinitely thick/ wall and for walls so

thin that thermal lag can be neglected were also obtained .

.I 981

.T

solutions to the heat-conduction equation with time dependent boundary

conditions .

.A

bergles, a.e. and kaye, j.

.B

j. aero. sc. v. 28, march 1961. pp 251-252 .

.W

solutions to the heat-conduction equation with time dependent boundary

conditions .

design charts based on the analytical solution to the problem of

one-dimensional heat flow in a solid body of constant material properties

with time-dependent boundary conditions were presented by kaye and

yeh . this solution dealt with aerodynamic heating at hypersonic speeds

where the surface coefficient of heat transfer and the temperature

potential were taken to be linear functions of time of flight . in order

to make these charts of more general application, general solutions are

presented which, together with the charts, enable rapid and reasonable

estimates to be made of the transient temperature distributions in many

practical cases .

.I 982

.T

the temperature history in a thick skin subjected to laminar heating during entry into the atmosphere .

.A

sutton, g.w.

.B

jet propulsion, vol. 28, january 1959, p 40-5

.W

the temperature history in a thick skin subjected to laminar heating

during entry into the atmosphere .

during high speed entry into the earth's atmosphere, a vehicle can be afforded thermal protection for the short period of entry heating by a thick outer skin, sometimes called a /heat sink/ . the temperature distribution in such a heat sink has been found by integrating the product of the laminar aerodynamic heating rate and the appropriate green's function for a finite-thickness wall over the generalized trajectory for a vehicle entering the earth's atmosphere at high speeds dimensional heat conduction problem for laminar heating . the maximum surface temperature that occurs during the generalized entry trajectory for any combination of wall thickness and thermal properties is obtained from which the performance of any material can be found, provided that the average thermal properties may be used . as an example of the use of the solution, the performance of copper, graphite, molybdenum and tungsten are compared .

.I 983

.T

addendum to 'heat transfer to satellite vehicles re-entering the

atmosphere .

.A

detra, r.w., kemp, n.h. and riddell, f.r.

.B

jet propulsion, vol. 27, december 1957 . pp 1256-57 .

.W

addendum to 'heat transfer to satellite vehicles re-entering the

atmosphere .

the original paper gave a correlation formula for stagnation point heat

transfer rate to a blunt body of revolution in hypersonic flow . this

note gives a somewhat refined version, based on further calculation and

shock tube data . its effect on the conclusions of the original paper is

negligible except at surface temperature parameters of over 5000

r-ft1/8 . in other problems, where heat transfer rate itself is

important, it can make a significant defference .

.I 984

.T

method of analysis for compressible flow through mixed-flow centrifugal

impellers of arbitrary design .

.A

hamrick, j.t. et al.

.B

naca report 1082, 1952 .

.W

method of analysis for compressible flow through mixed-flow centrifugal

impellers of arbitrary design .

a method is presented for analysis of the compressible flow between the

hub and the shroud of mixed-flow impellers of arbitrary design . axial

symmetry was assumed, but the forces in the meridional /hub to shroud/

plane, which are derived from tangential pressure gradients, were taken

into account .

the method was applied to an experimental mixed-flow impeller . the

analysis of the flow in the meridional plane of the impeller showed that

the rotational forces, the blade curvature, and the hub-shroud profile

can introduce severe velocity gradients along the hub and the shroud

surfaces . choked flow at the impeller inlet as determined by the

analysis was verified by experimental results .

.I 985

.T

a rapid approximate method for the design of hub shroud profiles of

centrifugal impellers of given blade shape .

.A

smith, k.j. and hamrick, j.t.

.B

naca tn 3399. 1955 .

.W

a rapid approximate method for the design of hub shroud profiles of

centrifugal impellers of given blade shape .

a rapid approximate method for the design of centrifugal compressors of

given blade shape with compressible nonviscous flow characteristics has

been developed using techniques based upon stream-filament theory .

axial symmetry is assumed, but meridional-plane forces derived from

tangential pressure gradients are included .

the method was applied to the design of an impeller in order to

determine the approximate maximum meridional streamline spacing that

could be used . three numerical solutions for different streamline spacings were made using the same hub profile, blade shape, and

prescribed velocity distribution along the hub . the shroud profiles

obtained from the three solutions, which utilized 3, 5, and 9

stream-tubes, were negligibly different . the approximate computing time

required was 15 hours per streamtube .

.I 986

.T

design and test of mixed-flow impellers, viii - comparison of

experimental results for three impellers with shroud redesigned by rapid

approximate method .

.A

osborn, w.m. et al.

.B

naca rm e56l07, 1957 .

.W

design and test of mixed-flow impellers, viii - comparison of

experimental results for three impellers with shroud redesigned by rapid

approximate method .

three centrifugal impellers with parabolic, circular, and

skewed-parabolic blading were modified by a recently developed design procedure

to reduce the velocity gradients along the hub from inlet to outlet .

all original dimensions except the shroud contours were retained .

experimental investigation showed that the modified impellers had better

performance characteristics than the original impellers at all speeds

investigated, the greatest gains occurring at speeds of 1300 feet per

second and higher . these large gains probably resulted primarily from

more favorable velocity gradients and from designing these impellers

further away from the condition necessary for eddy formation . the modified impellers were thus able to operate over a wider range of weight flows at high speeds .
the modified impellers were investigated over a range of equivalent speeds of 900 to 1500 feet per second and flow rates from maximum to the point of incipient surge . at 1300 feet per second, the peak pressure ratio and maximum adiabatic temperature-rise efficiency for the parabolic-bladed impeller were 3.07 and 0.825, respectively . for the same conditions, the circular-bladed impeller and the skewed-parabolic-bladed impeller had pressure ratios of 3.13 and 3.15 and efficiencies of 0.737 and 0.805, respectively . of the three, the parabolic-bladed impeller had the highest maximum efficiencies /0.854 to 0.800/ and the best weight-flow range over the speed range tested . on the basis of the parameters investigated, it appears that parabolic blading is superior to circular blading . the experimental results indicate that the design method of naca tn 3399 is a reliable method for use in designing centrifugal impellers .

.I 987

.T

a general theory of three dimensional flow in subsonic and supersonic turbo-machines of axial-radial-and mixed-flow types .

.A

wu, c.

.B

naca tn 2604, 1952 .

.W

a general theory of three dimensional flow in subsonic and supersonic turbo-machines of axial-radial-and mixed-flow types .

a general theory of steady three-dimensional flow of a nonviscous fluid in subsonic and supersonic turbomachines having arbitrary hub and casing shapes and a finite number of blades is presented . the solution of the three-dimensional direct and inverse problem is obtained by investigating an appropriate combination of flows on relative stream surfaces whose intersections with a z-plane either upstream of or somewhere inside the blade row form a circular arc or a radial line . the equations obtained to describe the fluid flow on these stream surfaces show clearly the several approximations involved in ordinary two-dimensional treatments . they also lead to a solution of the three-dimensional problem in a mathematically two-dimensional manner through iteration . the equation of continuity is combined with the equation of motion in either the tangential or the radial direction through the use of a stream function defined on the surface, and the resulting equation is chosen as the principal equation for such flows . the character of this equation depends on the relative magnitude of the local velocity of sound and a certain combination of velocity components of the fluid . a general method to solve this equation by both hand and high-speed digital machine computations when the equation is elliptic or hyperbolic is described . the theory is applicable to both irrotational and rotational absolute flow at the inlet of the blade row and at both design and off-design operations .

.I 988

.T

nonviscous flow through a pump impeller on a blade-to-blade surface of revolution .

.A

kramer, j.j. et al.

.B

nasa tn d-1108, 1962,

.W

nonviscous flow through a pump impeller on a blade-to-blade surface of

revolution .

the nonviscous incompressible flow through a typical pump impeller is

analyzed on a blade-to-blade surface of revolution . solutions are

obtained for a variety of inlet conditions including several with prewhirl

of the assumed location of the rear stagnation point . comparison of

results from two approximate methods of analysis showed good agreement

for the zero-angle-of-attack case and reliable indication of the

existence of an eddy on the driving face at a large positive angle of

attack .

.I 989

.T

incompressible nonviscous blade-to-blade flow through a pump rotor with

splitter vanes .

.A

kramer, j.j. et al.

.B

nasa tn d-1186, 1962 .

.W

incompressible nonviscous blade-to-blade flow through a pump rotor with

splitter vanes .

the nonviscous flow through a mixed-flow pump impeller having one splitter vane between adjacent main blades has been analyzed on a blade-to-blade surface of revolution using a previously reported analysis method . solutions were obtained for a variety of flow conditions including several cases in which whirl is imparted to the flow upstream of the impeller .

the velocity distributions on the main-blade surfaces and on the splitter-vane surfaces in the region of the splitter vane were strongly dependent on the assumed location of the rear stagnation points . solutions were obtained by assuming values of slip factor and of division of flow around the splitter in addition to assuming the location of the rear stagnation points . these solutions indicated that the velocity distributions in the splitter-vane region are largely determined by the division of flow around the splitter vane and that only the region in the immediate vicinity of the trailing edge is affected by the slip factor .

blade surface velocities were obtained from two approximate methods by specifying flow division and slip factor, and these results are compared with the more exact solutions of the analysis .

.I 990

.T

a rapid approximate method for determining velocity distribution on impeller blades of centrifugal compressors .

.A

stanitz, j.d. and prian, v.d.

.B

naca tn 2421, 1951 .

.W

a rapid approximate method for determining velocity distribution on
impeller blades of centrifugal compressors .
a rapid approximate method of analysis was developed for both
compressible and incompressible, nonviscous flow through radial- or
mixed-flow centrifugal compressors with arbitrary hub and shroud contours and
with arbitrary blade shape . the method of analysis is used to determine
approximately the velocities everywhere along the blade surfaces, but
no information concerning the variation in velocity across the passage
between blades is given .
in eight numerical examples for two-dimensional flow, covering a fairly
wide range of flow rate, impeller-tip speed, number of blades, and blade
curvature, the velocity distribution along the blade surfaces was
obtained by the approximate method of analysis and compared with the
velocities obtained by relaxation methods . in all cases the agreement
between the approximate solutions and the relaxation solutions was
satisfactory except at the impeller tip where the velocities obtained by
the approximate method did not, in general, become equal on both
surfaces of the blade as required by the joukowski condition .
.I 991
.T
wing-flow study of pressure drag reduction at transonic speed by
projecting a jet of air from the nose of a prolate spheroid of fineness
ratio 6 .
.A
lopatoff, m.
.B
naca rm l51e09, 1951 .

.W

wing-flow study of pressure drag reduction at transonic speed by

projecting a jet of air from the nose of a prolate spheroid of fineness

ratio 6 .

a study was made at transonic speeds by the naca wing-flow method of the

pressure-drag reduction obtained by projecting a high-energy jet of air

from the nose of a prolate spheroid . supplementary information was

obtained by taking shadowgraphs of the model mounted in a small

supersonic tunnel at a constant mach number of 1.5 .

the high-velocity jet was observed to alter the pressure distribution

over the body in such a way that the pressure drag of the body was

reduced,. thus, in a restricted sense, the nose jet produced a thrust on

the body . under the conditions investigated, the thrust produced by

the nose jet was never so large as that which would be expected from a

conventional rearward jet . for example, under the best conditions

tested /mach number of 1.07/ the reduction in body pressure drag caused

by the nose jet more than compensated for the negative thrust of the jet

itself . however, the magnitude of the net reduction in drag /change in

body pressure drag with jet on and jet off minus the adverse thrust of

the jet/ was only about one-half of the thrust which would be produced

by the same jet exhausting rearward . the appearance of such an

unexpectedly large effect in the first trial indicated the phenomenon to be

worth further study .

.I 992

.T

the effects of a small jet of air exhausting from the nose of a body

of revolution in supersonic flow .

.A

love, e.s.

.B

naca rm l52119a, 1952 .

.W

the effects of a small jet of air exhausting from the nose of a body

of revolution in supersonic flow .

an investigation has been made at a mach number of 1.62 to determine

the effects of a small jet of air exhausting from the nose of an

elliptical body of revolution upon boundary-layer transition and the

viscous, pressure, and total drag of the forebody at three body stations

body nose were also obtained . the tests were conducted at reynolds

numbers of 2.13 x 10 and 7.66 x 10, based on body length . the maximum

range of thrust coefficients for the small jet was from 0 to about

at the lower test reynolds number, for which the boundary layer was

laminar over the entire body in the jet-off condition, a very small flow

from the jet moved the point of transition forward to the vicinity of

the 20-percent-body station . as the jet flow was increased, the

transition point moved abruptly to the nose at a thrust coefficient of about

gardless of the type of boundary layer . at the higher test reynolds

number for which the boundary layer was largely turbulent in the jet-off

condition the total drag, including skin friction, was reduced somewhat

by the action of the jet .

although the forward-exhausting small jet was found to have the above

favorable effects upon the drag, these findings are not believed too

important since the question arises as to the benefits of the same small

jet exhausting from the rear of the body in the conventional manner .

no attempt was made to establish geometric optimums in the present investigation, yet, from a general consideration of the benefits indicated by the present results and the phenomena known to occur in the vicinity of rearward-exhausting jets, the benefits of a small jet exhausting rearward would appear to exceed those of the same small jet exhausting forward, particularly so when the flow over the body is laminar in the jet-off condition .

.I 993

.T

the extent of the jet interference flow fields .

jet effects on cylindrical afterbodies housing sonic and supersonic nozzles which exhaust against a supersonic stream at angles of attack from 90degree to 180degree .

.A

hayman, l.o. and mcdearmon, r.w.

.B

.W

the extent of the jet interference flow fields .

jet effects on cylindrical afterbodies housing sonic and supersonic nozzles which exhaust against a supersonic stream at angles of attack from 90degree to 180degree .

an investigation has been made to determine jet effects on cylindrical afterbodies housing sonic and supersonic nozzles which exhaust against a supersonic stream at angles of attack from 90 to 180 . the tests were conducted at a free-stream mach number of 2.91 and at free-stream reynolds numbers, based on body diameter, of 0.15x106 and stream static pressure investigated was from jet off to about 400 .

the data presented herein showed that, in general, variation of the ratio of jet total pressure to free-stream static pressure, jet-exit mach number, and ratio of jet-exit diameter to body diameter had large influences on the body pressures on the windward halves of the after-bodies and negligible influences on the leeward pressures . there was a negligible effect of reynolds number on the body pressures . the ratio of jet total pressure to free-stream static pressure also had a large influence on the base pressures at all angles of attack . schlieren studies showed details of the shock-wave structure caused by the jet and the extent of the jet interference flow fields .

.I 994

.T

investigation of a retrocket exhausting from the nose of a blunt body into a supersonic free stream .

.A

charczenko, n. and hennessy, k.w.

.B

nasa tn d-1016, 1962 .

.W

investigation of a retrocket exhausting from the nose of a blunt body into a supersonic free stream .

the pressure distribution and pressure drag of a blunt body with a supersonic jet issuing upstream from its center were determined at free-stream mach numbers of 1.60, 2.00, and 2.85 . the thrust of the jet issuing from the model nose was varied to study its effects on flow around the model and to determine variation of pressure distribution and pressure drag of the model with the thrust .

at all mach numbers investigated, the pressure drag decreased with

increasing retrorocket thrust until a minimum value was reached .

further increases in retrorocket thrust resulted in increases in the

pressure drag . the resultant drag /pressure drag plus retrorocket

thrust but excluding base and skin-friction drag/ of the model was

reduced by retrorocket operation below the drag for a jet-off condition,

except at very low retrorocket thrust coefficients . the flow about the

nose of the blunt body was very unstable throughout the range of mach

numbers and retrorocket thrust coefficients investigated .

.I 995

.T

.A

.B

.W

.I 996

.T

extension of boundary layer separation criteria to

a m=6 .5 utilizing flat plates with forward-facing

steps .

.A

sterrett,j.r. and emery,j.c.

.B

nasa tn.d618, 1960.

.W

extension of boundary layer separation criteria to

a m=6 .5 utilizing flat plates with forward-facing

steps .

an experimental investigation has been made of the separation phenomena on a flat plate to which forward-facing steps were attached to force separation . both laminar and turbulent flows were investigated over a mach number range of approximately distributions, shadowgraph and chemical film techniques, the pressure rise at separation, the laminar plateau pressure, and the turbulent peak pressure were determined . boundary-layer surveys were made on a smooth flat plate and on a flat plate with roughness to force transition . examinations of the separated flow showed that the predominant variable in the determination of the pressure distribution was the location of transition relative to the separation point and reattachment . pure laminar, transitional, and turbulent types of separation were found in this mach number range . the peak static-pressure-rise ratios for identical forward-facing steps at a mach number of 6.25 were approximately 1.5 and 5.0, respectively, for pure

laminar and turbulent

separation . the effect of reynolds number on the

peak pressure rise for

turbulent separation for the lower mach number

range was found to be very

minor provided the step height was of the

order of the boundary-layer

thickness . as the mach number is increased,

the peak pressure

coefficient for turbulent separation decreased

from approximately 0.18

at a mach number of 4 to about 0.13 at a mach

number of 6.25 . the

pressure coefficient at the separation point for

laminar separation decreases

from approximately 0.014 at a mach number of

value at a mach number of 6.5 . the results

obtained with forward-facing

steps agree with the trends predicted, based

upon lower mach number

studies .

.I 997

.T

experimental and theoretical studies of axisymmetric free jets .

.A

love, e.s. et al.

.B

nasa tr r-6, 1959 .

.W

experimental and theoretical studies of axisymmetric free jets .

some experimental and theoretical studies have been made of axisymmetric free jets exhausting from sonic and supersonic nozzles into still air and into supersonic streams with a view toward problems associated with propulsive jets and the investigation of these problems .

for jets exhausting into still air, consideration is given to the effects of jet mach number, nozzle divergence angle, and jet static-pressure ratio upon jet structure, jet wavelength, and the shape and curvature of the jet boundary . studies of the effects of the ratio of specific heats of the jets are included as are observations pertaining to jet noise and jet simulation .

for jets exhausting into supersonic streams, an attempt has been made to present primarily theoretical curves of the type that may be useful in evaluating certain jet interference effects and in formulating experimental studies . the primary variables considered are jet mach number, free-stream mach number, jet static-pressure ratio, ratio of specific heats of the jet, nozzle exit angle, and boattail angle . the simulation problem and the case of a hypothetical hypersonic vehicle are examined

.I 998

.T

equations, tables and charts for compressible flow .

.A

ames research staff.

.B

naca r.1135, 1953.

.W

equations, tables and charts for compressible flow .

this report, which is a revision and extension of naca tn

useful in the analysis of high-speed flow of a compressible fluid .

the equations provide relations for continuous one-dimensional

flow, normal and oblique shock waves, and prandtl-meyer

expansions for both perfect and imperfect gases . the tables

present useful dimensionless ratios for continuous one-

dimensional flow and for normal shock waves as functions of mach

number for air considered as a perfect gas . one series of charts

presents the characteristics of the flow of air (considered a perfect

gas) for oblique shock waves and for cones in a supersonic air

stream . a second series shows the effects of caloric

imperfections on continuous one-dimensional flow and on the flow

through normal and oblique shock waves .

.I 999

.T

static aerodynamic characteristics of short blunt cones

with various nose and base cone angles at mach numbers

of 0. 6 to 5. 5 and angles of attack to 180 .

.A

treon,s.l.

.B

nasa tn.d1327, 1962.

.W

static aerodynamic characteristics of short blunt cones

with various nose and base cone angles at mach numbers
of 0. 6 to 5. 5 and angles of attack to 180 .
wind-tunnel tests have been performed
at mach numbers from 0.6 to 5.5
to determine coefficients of normal force,
axial force, and pitching
moment for short blunt cones, as affected
by changes in nose and base
cone angles . models with nose half-angles
of 10 and 20 were investigated .
the 10 nose half-angle models were tested
with a flat base and with base
cones of 50 and 70 half-angle . the 20
nose half-angle model had a 50
half-angle base cone . reynolds numbers
for the test ranged from about
maximum diameter .
variations in the base cone angle
resulted in significant changes in
the aerodynamic characteristics, with
lesser effects resulting from changes
in nose cone angle . in particular,
the model with the 50 half-angle
conical base had only one trim angle
flat base and 70 half-angle conical
base had two trim angles (a = 0 and
a = 180) . estimated variations of

the aerodynamic characteristics with

angle of attack by means of a modified

newtonian theory were in good

agreement with the experimental results .

the theory, however, failed to

predict the trim point at a = 180 for

the flat-based model .

.I 1000

.T

free-flight measurements of the static and dynamic

stability and drag of a 10 blunted cone at mach numbers

3 .5 and 8 .5 .

.A

intrieri, p. f.

.B

nasa tn. d1299, 1962.

.W

free-flight measurements of the static and dynamic

stability and drag of a 10 blunted cone at mach numbers

3. 5 and 8 .5 .

tests were made of a short blunt-nosed

without a 50 half-angle conical afterbody

in a pressurized ballistic

range at nominal mach numbers of 3.5 and

of 90,000 and 220,000, respectively . it

was found that the models were

statically stable about the center-

of-gravity location tested but
exhibited neutral dynamic stability for flight
at constant altitude . the
static stability was not affected by the
but was nonlinear with angle of attack
and varied with mach number . the
nonlinear variation of the pitching moment
with angle of attack was
accurately approximated by a cubic polynomial .
the static stability was only
qualitatively predicted by modified newtonian
theory . the drag
characteristics were in good agreement with values
calculated by use of modified
newtonian theory .

calculations of the oscillatory
behavior of the configurations flying
an example entry trajectory through
the martian atmosphere indicated the
configurations to be dynamically
satisfactory . pitching motions should
converge to a small fraction of the
amplitude at entry, provided the
initial angle of attack and pitch
rate are not large enough to cause
tumbling .

.I 1001

.T

wind tunnel investigation of the static and dynamic stability

characteristics of a 10degree semivertex angle blunted cone .

.A

wehrend, w.r.

.B

nasa tn d-1202, 1962 .

.W

wind tunnel investigation of the static and dynamic stability

characteristics of a 10degree semivertex angle blunted cone .

the static and dynamic stability characteristics of a blunted 10degree

semivertex angle cone were studied . the cone which had a modified

spherical segment nose was tested with a flat base and with a truncated

conical base .

all tests were performed in air at mach numbers from 0.65 to 2.20 with

the angle-of-attack range from -4degree to +18degree . presented are

measurements of the normal force, axial force, base pressure, and

pitching moment from the static tests, and the damping-in-pitch moment

from the dynamic tests .

both models had satisfactory stability characteristics throughout the

test mach number range but the addition of the conical afterbody had a

large destabilizing effect .

.I 1002

.T

preliminary investigations of spiked bodies at hypersonic speeds .

.A

bogdonoff, s.m. and vas, i.e.

.B

jnl. aero. sci. february 1959, p. 65-74 .

.W

preliminary investigations of spiked bodies at hypersonic speeds .

generally accepted solutions for the problems of hypersonic flight

appear, at the moment, to be centered around the use of blunt bodies to

minimize the heat-transfer rates . there are, however, several other

solutions to the problem, and, as part of an exploratory study of these

solutions, a detailed examination has been made of the flow over blunt

bodies equipped with a spike . these tests, carried out at a mach number

of about 14 in the princeton helium hypersonic tunnel, have

investigated the effect of varying spike lengths for flat-faced and

hemispherically-nosed axially symmetric bodies . detailed pressure distributions

have been obtained as well as heat-transfer rates .

these exploratory studies have shown that the use of a spike protruding

from a hemispherical-nosed cylinder at m 14 decreased the pressure

level by an order of magnitude and the heat transfer to a fraction of

that measured on a hemisphere without a spike . the general technique

appears to hold considerable promise for hypersonic flight .

.I 1003

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

the real-gas hypersonic flow parameters for helium have been calculated for stagnation temperatures from 0 f to 600 f and stagnation pressures up to 6,000 pounds per square inch absolute . the results of these calculations are presented in the form of simple correction factors which must be applied to the tabulated ideal-gas parameters . it has been shown that the deviations from the ideal-gas law which exist at high pressures may cause a corresponding significant error in the hypersonic flow parameters when calculated as an ideal gas . for example, the ratio of the free-stream static to stagnation pressure as calculated from the thermodynamic properties of helium for a stagnation temperature of 80 f and pressure of 4,000 pounds per square inch absolute was found to be approximately 13 percent greater than that determined from the ideal-gas tabulation with a specific heat ratio of

.I 1004

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

the effects of contamination of helium by air upon static-pressure, total-pressure, heat-transfer, and temperature measurements have been investigated in the 2-inch helium tunnel at the langley research center . within the scope of the tests, even a small amount of air is shown to affect these measurements . the heat-transfer and temperature measurements were made on a 26.6 half-angle cone and demonstrated the effects of contamination qualitatively . the wall static and center-line pitot pressures show that if the contaminating air is held to less than about 0.2 percent by volume, the error in indicated mach number is

less than 1 percent as calculated from the rayleigh pitot equation .

the corresponding errors in wall static and center-line pitot pressures

are about 1.7 and 0.4 percent, respectively .

.I 1005

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

equations based on newtonian impact theory have been derived and a

computational procedure developed with the aid of several design-type

charts which enable the determination of the aerodynamic forces and

moments acting on arbitrary bodies of revolution undergoing either

separate or combined angle-of-attack and pitching motions . bodies with

axially increasing and decreasing cross-sectional area distributions are

considered,. nose shapes may be sharp, blunt, or flat faced . the

analysis considers variations in angle of attack from -90 to 90 and

allows for both positive and negative pitching rates of arbitrary

magnitude . the results are also directly applicable to bodies in

either separate or combined sideslip and yawing maneuvers .

.I 1006

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

the inviscid flow of a perfect gas over blunt-nosed axisymmetric and two-dimensional bodies at zero angle of attack has been calculated numerically on an ibm 7090 computer . the computation consisted of the fuller blunt-body solution for the subsonic and transonic regions and the method of characteristics for the supersonic region . the flow fields about a number of blunt bodies were studied, and the calculated results showed good agreement with experimental shock-wave shapes, surface-pressure distributions, and flow-field surveys .

.I 1007

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

this report presents equations, tables, and figures for use in the analysis of helium flow at supersonic and hypersonic speeds . the contents of the report and presentation of the data parallel that of a similar reference work (naca rep. 1135) prepared for air flow .

the perfect-gas relations for continuous one-dimensional flow, normal- and oblique-shock waves, and prandtl-meyer expansions are the same as for air but are presented here for completeness . the tables present the values of useful dimensionless ratios for continuous one-dimensional flow and for normal-shock waves as functions of mach number . the helium viscosity relation as a function of temperature,

mass-flow rates as a function of mach number and temperature, and the
reynolds number as a function of mach number and stagnation temperature
are plotted . the oblique-shock characteristics of wedges and cones in
helium at mach numbers of 12, 16, 20, and 24 are presented in a series
of plots . throughout all the computations, helium is considered to be
a perfect gas .

.I 1008

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

representative experimental results are presented to show the
current status of the panel flutter problem . results are presented for
unstiffened rectangular panels and for rectangular panels stiffened by
corrugated backing . flutter boundaries are established for all types
of panels when considered on the basis of equivalent isotropic plates .
the effects of mach number, differential pressure, and aerodynamic
heating on panel flutter are discussed . a flutter analysis of
orthotropic panels is presented in the appendix .

.I 1009

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

charts of thermodynamic properties for equilibrium air are presented with sufficient accuracy to permit the calculation of flow parameters in hypersonic nozzles operating at stagnation temperatures up to 4,950 r and pressures up to 1,000 atm . flow parameters calculated from these charts are presented for a series of stagnation temperatures between use of these parameters, it is possible to calibrate a nozzle in the conventional way . a method is also presented from which the flow parameters for conditions other than those chosen herein may be calculated . real-gas effects on the calculation of a hypersonic nozzle contour are shown by an example calculation in which the nozzle contour for mach number 12 was determined by including real-gas effects, and this contour was compared with one calculated by ideal-gas considerations . also presented are the approximate limiting mach numbers at which equilibrium air will just condense for various combinations of stagnation temperatures and pressures .

.I 1010

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

air-flow properties in nozzles were calculated and charted for equilibrium flow and two types of frozen flows . in one type of frozen flow, air was assumed to be in equilibrium from the nozzle reservoir to arbitrary points where chemical reactions and molecular vibrations became frozen . in the other type, it was assumed that molecular

vibrations were in equilibrium throughout the nozzle and that chemical

reactions became frozen at arbitrary points . the calculations were

made for a range of stagnation pressures up to 10,000 poinds per square

inch absolute and stagnation enthalpies up to 24,500 btu per pound .

the flow properties charted were temperature, pressure, density,

velocity, dynamic pressure, mach number, reynolds number, molecular

weight fraction, and mass flow . equilibrium flow properties through

normal shock waves were also included .

.I 1011

.T

free-flight measurements of the static and dynamic .

.A

.B

.W

free-flight measurements of the static and dynamic .

charts have been prepared relating the thermodynamic properties of

air in chemical equilibrium for temperatures to 15,000 k and for

pressures from 10 to 10 atmospheres . also included are charts showing

the composition of air, the isentropic exponent, and the speed of

sound . these charts are based on thermodynamic data calculated by the

national bureau of standards .

.I 1012

.T

principles of creep buckling weight-strength analysis

of aircraft structures .

.A

shanley,f.r.

.B

part iii. p 343, mcgraw-hill, new york, 1952.

.W

principles of creep buckling weight-strength analysis

of aircraft structures .

the possibility of a gradual instability

failure of a column under

compressive load has been recognized

for some time . marin presented an

analysis of creep buckling based on

a theory of creep bending, but did not

take into account the average stress

due to axial loading . the theory also

neglected the transient (nonlinear)

portion of the creep curve .

in efficient column design, the

average stress should be relatively high in

comparison with the bending stresses,.

that is, the column should be as straight

as possible and the slenderness ratio

should not be too great . under these

conditions marin's theory is not

directly applicable, although it gives good

agreement with tests of columns

having large slenderness ratios or large

eccentricities .

.I 1013

.T

principles of creep buckling weight-strength analysis .

.A

.B

.W

principles of creep buckling weight-strength analysis .

published work on creep buckling has implied that failure of columns after a critical time is caused by initial imperfections . such analyses are relatively complex and ultimately leave the choice of selecting the proper value of the initial imperfection to the designer . furthermore, recent test results on creep buckling of columns have indicated that there is a random and relatively unimportant effect of small initial imperfections on the critical time .

to avoid the difficulties associated with initial imperfections, a formulation of the creep buckling phenomenon in terms of classical stability theory is presented . the theory permits the extension of known solutions for plastic buckling of certain thin plates and shells to creep buckling problems .

.I 1014

.T

principles of creep buckling weight-strength analysis .

.A

.B

.W

principles of creep buckling weight-strength analysis .

a problem of creep stability of columns and plates
is considered . in an analysis use is made
of two forms of the creep theory based on the strain
hardening hypothesis . for a uniformly
compressed palte a comparison is made between the
results according to the flow theory and strain
theory .

.I 1015

.T

principles of creep buckling weight-strength analysis .

.A

.B

.W

principles of creep buckling weight-strength analysis .

the determination of column deflections and
column buckling loads has been considered for
many years . the available theories, however,
do not provide for materials which creep with
time under constant loads . for the design of
structural members made of these materials, a
consideration of creep may be of practical
importance . plastics, concrete, and some metals
creep at normal temperatures while other
metals creep only at high temperatures and at stress
values beyond the yield point . a consideration
of creep in the design of some structures appears
appropriate in view of the modern developments

in plastics and the presence of high stress values

which are sometimes beyond the yield stress .

this paper gives a rational theory for predicting

creep deflections in columns . a special case

using this theory is applied to the interpretation of

some preliminary tests of an aluminum alloy .

.I 1016

.T

principles of creep buckling weight-strength analysis .

.A

.B

.W

principles of creep buckling weight-strength analysis .

  the relation of the time-dependent tangent-modulus

load--as conceived by shanley--to actual column capacity

is clarified . it may be interpreted as a limiting case of the

conservative estimate . the time-dependent

tangent-modulus load is, therefore, an approximation to a

conservative estimate . the approximation, however, may

be either conservative or nonconservative when applied

to imperfect or real columns . typical cases are discussed

and experimental results for two alloys are cited .

.I 1017

.T

note on creep buckling of columns .

.A

gerard,g.

.B

j. ae. scs. 19, 1962, 714.

.W

note on creep buckling of columns .

it appears from librove's interesting analysis that, for

the case of creep buckling of columns, the initial

imperfections contained in ordinary columns provide the mechanism by

which failure due to creep occurs after a period of time . in fact,

it can be concluded from this analysis that a theoretically perfect

column that is initially loaded below the time-independent critical

load will not buckle at all . this is an interesting contrast to the

case of static buckling where small initial imperfections play an

insignificant role, since the failing load of an initially imperfect

column is substantially the same as that of a theoretically perfect

column . it is of interest, therefore, to conjecture whether there is

any possible mechanism by which a column containing no initial

imperfections can fail as a result of creep when the initial load is

less than the theoretical buckling load .

.I 1018

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the results of short-time elevated-temperature creep tests of

objective of obtaining procedures for predicting column lifetime .

semiempirical lifetime curves are obtained with the aid of a previously

published column creep theory and are used for deriving column curves .

the semiempirical lifetime curves are also used to study the effect of

varying applied stress and out-of-straightness . in the range

considered, small variations in out-of-straightness are found to be of

little practical significance,. whereas, small stress variations change

the column lifetime considerably . for the range of out-of-straightness

encountered in the tests, the data can be presented in plots that do not

explicitly include out-of-straightness, and plots of this type should be

satisfactory for predicting column lifetime for design purposes .

.I 1019

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a method for estimating allowable

load capacities of columns subject

to creep is presented . the method,

which utilizes approximate stress

distributions derived from

isochronous-stress-strain curves to estimate

column load capacities, is shown to

be conservative for the time for which

the estimate is made .

an application of the method is

made to test data on as-received and on stabilized 24s-t4 aluminum alloy . a comparison of the computed column capacities with experimental capacities indicates that the method is satisfactory for estimating the decrease in capacity with increasing time .

easily obtained, time-dependent tangent-modulus loads are discussed . they are interpreted as being approximations to allowable load-capacity estimates . a limited application is made to test data, and the results appear promising . it is concluded that if certain limitations are recognized, the method may prove to be useful because of its simplicity .

a presentation of the results of an experimental investigation of the effects of column imperfection and column-material variation is made . it is found that column-capacity variations of the order of 10 per cent can result from column-imperfection differences and column-material variation .

the results of an experimental study of the variation of column capacity with temperature of exposure are presented .

they indicate that column

efficiency, as measured by decrease in capacity,

can be acceptable for very

short times at the higher temperatures . the

efficiency at these higher

temperatures falls rapidly, however, with increasing time .

.I 1020

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the results of short-time

creep-buckling and creep-bending tests of

slenderness ratio 111 are presented .

the tests were performed at 600 f,

and strain measurements were taken

with high-temperature electric-resistance

strain gages . a description

of the development of the gages is given

in an appendix . the column

tests show that the critical time decreases

much more rapidly with

increasing load than with increasing initial

deviation from straightness .

the bending tests indicate that the steady

creep rate of the curvature

is a simple power function of applied

moment . these latter results,

together with a previously derived

creep-buckling theory, are used to

develop a semiempirical formula suitable

as a guide for the determination

of the critical time for columns .

.I 1021

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

this paper describes theory and tests of the creep collapse

of long thin aluminum-alloy cylinders under external radial

pressure . steady-state creep is assumed in the theoretical

derivation . the test temperatures were between 300 and 500 f .

the collapse time for each cylinder was calculated theoretically .

agreement between theoretical and test results was fair .

.I 1022

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

forty-three cylinders of 40-inch

length and 16-inch diameter, made

of 5052-0 aluminum-alloy sheets of

thickness, were subjected to bending

moments constant along the cylinder

and in time in an oven which maintained

a constant temperature of 500 f

during the test . all the cylinders

failed by buckling . the time that

elapsed between load application and

collapse was measured .

.I 1023

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a theory of creep buckling is presented in which the instantaneous

elastic and plastic deformations following the application of a load, as

well as the steady creep deformations, are considered in an approximate

manner . equations are given from which the critical time, that is, the

time elapsing between load application and the collapse of the column,

can be computed .

.I 1024

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the general dynamic equation of creep bending of a beam loaded laterally and axially was derived for a linearly viscoelastic material whose mechanical properties can be characterized by four parameters . the material can exhibit instantaneous and retarded elasticity as well as pure flow .

the equation derived was used to obtain the creep bending deflection of a beam in pure bending and of a column with initial sinusoidal deviation from straightness . as expected, the ratio of the creep deflections of the beam in pure bending and the deflections of a corresponding purely elastic structure is identical to the ratio of the creep strain and the corresponding elastic strain of a bar under simple tension or compression .

the results of the analysis of the

creep deflection of the column

showed that the deflections increase

continuously with time and become

infinitely large only when the loading

time is correspondingly large .

however, large deflections are obtained

in reasonably short periods of

time if the applied load is near to the

euler load of the column . the

deflection-time curves obtained from a

numerical example are of the same

type as those determined by experiment

with aluminum columns .

.I 1025

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the creep of a slightly crooked section column carrying a

constant load is studied theoretically . the material of the

column is characterized by a strain-time relationship, under

constant uniaxial stress, of the form, where is

the total strain, is the constant stress, is the time, and e,a,b,

and k are material constants . this form was selected because it

applies to at least two alloys--75s-t6 aluminum alloy at 600 f .
and a low-alloy steel at 800 and 1,100 f . however, the
analysis is intended for any material having creep properties of the
same form and for which the material constants are known . a
strain-time relationship under variable uniaxial stress, necessary
for the column analysis, is formulated from the constant-stress
properties with the aid of shanley's engineering hypotheses of
creep .

the analysis leads to the conclusion that the lateral deflection
approaches infinity--that is, the column collapses--in finite time .
results are given showing the maximum length of time the
column can support a given load before it collapses and the growth
of stresses, strains, and deflections prior to collapse .

.I 1026

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

this paper is concerned with the solution of the creep buckling of
columns . instantaneous elastic and plastic deformations, as well as
the transient and secondary creep, are considered . formulae for the
critical time at which a column fails are presented for integral values
of the exponents appearing in the creep law .

.I 1027

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a phenomenological relation between

stress, strain rate, and

temperature is suggested to account for the

behavior of polycrystalline metals

above the equicohesive temperature .

the properties of the metal

included in the relation are elasticity,

linear thermal expansion, and

viscosity . the relation may be integrated

under various conditions to

provide information on creep rates, creep

rupture, stress-strain curves,

and rapid-heating curves . it is shown

that for one material - 7075-t6

aluminum-alloy sheet - the information

yielded by the relation for these

four applications agrees reasonably

well with test data .

.I 1028

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the phenomenological theory previously

proposed in naca technical

note 4000 for the behavior of metals at

elevated temperatures has been

modified to yield transient creep curves

by assuming that the metal

consists of two phases, each with its own

elasticity and viscosity . the

extended theory satisfies the basic

requirements for a theory of

transient creep at elevated temperatures ..

that the transient creep be closely

connected with the subsequent steady

creep, and that the apparent

exponent of the time in the transient region

be permitted wide variations

between 0 and 1 . from this theory it is

possible to construct

nondimensional creep curves which extend continuously

from the transient region

into the steady-state region . the

corresponding family of creep curves

for any metal may be obtained from

the nondimensional family by use of

.T

appropriate constants . the constants

required are those obtained from

steady creep measurements, together

with two additional constants which

represent the difference between the

phases . the transient creep curves

resulting from this theory are compared

with the experimental curves

for pure aluminum, gamma iron, lead, and

agreement is found .

.I 1029

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a variational theorem is presented for a body undergoing

creep . solutions to problems of the creep behavior of plates,

columns, beams, and shells can be obtained by means of the

direct methods of the calculus of variations in conjunction with

the stated theorem . the application of the theorem is

illustrated for plates and columns by the solution of two sample

problems .

.I 1030

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

some general topics in elastic stability are discussed . in particular, attention is given to the relationship between adjacent-equilibrium-position and energy techniques, to the effects of nonlinearity, and to the sensitivity of certain stability problems to the character of the loading .

.I 1031

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a general variational theory of elastic stability that was originated by e. trefftz (1) is applied to the problem of buckling of rings of rectangular cross section subjected to uniform external pressure . the theory is believed to be more rigorous than previous treatments of the problem, since it avoids conventional assumptions of curved-beam theory, such as the assumptions that plane sections remain plane and that radial stresses vanish . the classical result of levy (2) is confirmed for a ring of infinitesimal thickness . new results are obtained which show the effect of the finite thickness of a ring on the coefficients in the buckling formula .

.I 1032

.T

on the conservativeness of various distributed force systems .

.A

sol r. bodner

.B

assistant professor of engineering, brown university, providence,

r.i.

.W

on the conservativeness of various distributed force systems .

the necessity of determining the conservativeness of force systems in

instability problems is discussed in reference 1 . it is shown that,

whereas kinetic methods are generally applicable for the determination

of instability loads, the statical methods usually employed are valid

only for conservative and nongyroscopic systems . small changes in

the character of the loading could make an otherwise conservative

system nonconservative and cause a large change in the magnitude of the

buckling load . the buckling load of the cantilever column example

in reference 1 is, for tangential end loading, eight times that for

conservative, constant directional loading .

.I 1033

.T

the design of tubes under uniform external pressure on the

basis of assumed inaccuracies .

.A

.B

.W

the design of tubes under uniform external pressure on the

basis of assumed inaccuracies .

since the failure of tubes under uniform external pressure depends

very much upon the various kinds of imperfections in them,

it seems logical to derive a design formula for such tubes in

which the quantities depending on imperfections will appear explicitly .

the most common imperfection in tubes is an initial ellipticity,

the limiting value of which in each type of tube is usually well-known

from numerous inspection measurements . the deviation of the shape

of the tube from a perfect circular form can be defined by the

initial radial deflections w' .

.I 1034

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

a long, thin-walled cylindrical shell is loaded by a uniform

external pressure . equations are developed for the time behavior

of the shape of the cross section under the following conditions ..

formations expressible by a power creep law,. (b) the initial and

subsequent mode shape of the deviations from circularity of any

cross section is two-lobed,. and (c) the shell construction is of the

sandwich type, with concentric cylindrical membranes taking

normal stresses and an annular core supporting shear without

deformation . explicit solutions are obtained for the particular
case of the cubic creep law . it is shown that the nondimensional
amplitude of the cross-sectional mode shape (briefly, shape
factor) will become infinite in a finite time . curves of shape factor
versus time and of collapse time versus initial value of the shape
factor are presented . also given are an explicit expression for
and a curve of the expected variation in collapse time owing to
uncontrollable deviations from a nominal initial value of the
shape factor . it is shown that the expected variation is small
if the nominal initial shape factor value is sufficiently large .

.I 1035

.T

note on creep buckling of columns .

.A

.B

.W

note on creep buckling of columns .

the stability of a compressed elastic
ring has been studied by a method which
can be extended to solve the problem
of the stability of a flexible heavy structure
spread by a system of hoops as in a crinoline
skirt . the original work by levy, which
was developed by timoshenko and love,
cannot be generalized to problems in
which the compressing forces are affected by
the deformation of the ring .

it is shown that the load at which a ring will buckle depends not only upon the magnitude of the load but also upon its first derivative relative to the radial distance . a positive derivative causes the ring to buckle at a higher load . when this result is applied to a cone of heavy and loosely draped fabric spread by a rigid hoop of radius and a larger and flexible hoop of radius below it, both hoops being in horizontal planes, then various modes of buckling other than oval are possible according to the relative magnitudes of and . it is found that oval buckling changes to three-wave buckling when three-wave changes to four-wave when, and as and approach nearer to equality the buckled form progressively changes to more waves . when applied to a structure spread by many horizontal hoops of which the top one is rigid and oval, it is found that all other hoops, if each is designed to the criterion, will have the same absolute deviation from circularity as the rigid hoop . if any one hoop is designed so that, then the oval shape of the rigid hoop is magnified on all flexible

hoops .

.I 1036

.T

on transverse vibrations of thin, shallow elastic shells .

.A

reissner,e.

.B

q. app. math., 13, 1955, 169.

.W

on transverse vibrations of thin, shallow elastic shells .

according to marguerre (proc. 5th internat. congress

appl. mech., cambridge, mass., 1938, wiley, new york,

shells are governed by three simultaneous differential

equations in the three displacements . the author has

considerably simplified this theory for the case of transverse

vibrations by ignoring the longitudinal inertia terms, thus

reducing the problem to that of solving two simultaneous

differential equations in a stress function and one

displacement component . this simplification is justified by an

order-of-magnitude analysis, and illustrated by considering the

vibrations of a paraboloidal shell with a rectangular

boundary .

.I 1037

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

it is shown that a perfectly straight bar, subjected

to a state of uniform shear stress, will

buckle, in a manner similar to a column under

compression, if the shear stress exceeds

a certain critical value . the buckling equations

are obtained by the newtonian

approach and also by the application of the principle

of minimum potential energy . in

order to provide additional insight into this buckling

mechanism, a simple model is

introduced and analyzed .

.I 1038

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

the deformation and complete stress

distribution are determined for

each of the following edge-loaded thin

shells .. (1) a right circular

cylinder, (2) a frustum of a right

circular cone, and (3) a portion of a

sphere . the locations of maximum

circumferential and meridional stresses

are also found . equations are

developed for discontinuity shear and

moment at the following junctions ..

circular cylinder, (2) axial change

of thickness in a cone, (3) change

of thickness in a portion of a sphere,

cylinder and a portion of a sphere,

.I 1039

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

an experimental investigation was

made (1) to evaluate previously

published theoretical procedures for

the prediction of stress

distribution for cases of radially symmetric

abrupt change in wall thickness of

thin-walled cylinders subject to

internal pressure and (2) to

investigate the significance of stresses

attributable to the presence of

thickness changes typical of design practice .

one theory was adequate in itself for solution of the case of continuous middle surface,. use of the second theoretical procedure was required to determine the additional stresses arising from discontinuous middle surfaces at the change in thickness .

comparisons were made between theoretical and experimental stress distributions for cases with continuous middle and continuous inner surfaces for radially symmetric changes in wall thickness of a cylinder subject to internal pressure for diameter to larger wall thickness ratios of 117 and 28 and for the case of a continuous outer surface for a ratio of 28 . in all tests the ratio of wall thicknesses at the change in wall thickness was 0.4 .

there was reasonably good correlation between theoretical and experimental curves of stress distribution . on the basis of this correlation, it was concluded that the applicable theories were valid . it

was shown that inclusion of the stresses
arising from the condition of
discontinuous middle surfaces at a change
in thickness has an important
effect on stress distribution .
in the case of a cylinder with a
continuous outer surface, the
maximum mean effective stress was of sufficient
magnitude to indicate that
this geometry should be avoided in design
if possible . the maximum
mean effective stress was not increased to
a significant degree by the
presence of a change in wall thickness in
the other cases .

.I 1040

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

experimental measurements of pressures and heat-transfer
rates over three blunt afterbodies of small fineness ratio in fully
separated wakes are presented . the afterbodies are generally
similar in shape but have different stepdown heights from the end
of the forebody .

tests were made by means of a new shroud technique over a range of reynolds numbers closely corresponding to typical flight conditions at mach numbers on the order of 20, considering models on the order of 5 ft. in diameter at about 120,000 ft. altitude . stagnation temperatures on the order of 1,300 r. to

strictly speaking, the test flows correspond to prototype flows which would be created by a forebody consisting of a sharp or slightly blunted 54 half angle cone which turns cylindrical for a short distance and then connects with the afterbody . judiciously interpreted, the results may be considered to have a somewhat wider applicability for approximation purposes . the results are presented and compared with each other in terms of nondimensional variables based on flow conditions at the end of the forebody .

the pressure distribution along an afterbody is seen to be roughly uniform in each run . for a given point on an afterbody, the ratio of pressure to the stagnation pressure at the forebody end (or exit) decreases with increasing stagnation pressure or reynolds number . the present pressures and pressure-reynolds number variations (fig. 8) are compared with values obtained from chapman's mach 2 or 3 base-pressure data,. qualitative and some quantitative agreement is noted . in the reynolds number range comparable to those of the present tests, chapman's exit boundary layers were considered to be laminar . an approximate check of the heat-transfer rate at the forebody end in the present tests also indicates a laminar rate . no information was obtained concerning the possible transition of the free-mixing separated boundary layer covering the wake . an adverse pressure gradient on the cylindrical end of the forebody figs. 7(a) and 7(c) was observed .

heat-transfer rates are seen to be roughly uniform over an afterbody in each run, although some increase in the streamwise direction is noted . the afterbody nusselt number (n) varies with the reynolds number (r evaluated at the forebody end) roughly in the manner n r where generally (fig. 13) .

heat rates on the rear faces of the afterbodies are almost twice the values on the sides . the heat rates on the large-step body are higher than those on the body of zero stepdown height .

in an addendum, it is shown that the prandtl-meyer expansion angle of the flow leaving the afterbody increases with increasing test reynolds number, and that the corresponding local mach number square increases linearly with reynolds number . the effect is to keep the local wake reynolds numbers virtually constant with increasing test reynolds number while the afterbody heat rates increase sharply . the expansion angle on the afterbody of zero stepdown height is significantly smaller than on the stepped down bodies,. this may affect the decreased heat rates on this body .

.I 1041

.T

analysis of stresses in the elements of shell structure .

.A

c. e. taylor and

e. wenk, jr.

.B

structural research engineer, david taylor model basin head,

structures division, david taylor model basin

.W

analysis of stresses in the elements of shell structure .

the love-meissner analysis for thin shells has previously been applied

to cones of uniform wall thickness, and solutions for the stress

resultants were given in terms of kelvin's functions . since

tabulation of these functions for large arguments is not practical,

considerable computation was still required . in the present paper, the authors

define special functions which eliminate the necessity of evaluating

kelvin/s functions and which may be used with simple algebraic and

trigonometric functions to compute the boundary forces and displacements

for cones for various loading conditions . these special functions also

make clear the magnitude of errors which result from geckeler/s and

other approximate solutions .

.I 1042

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

the report presents information

on the stress problems in the

analysis of pressurized cabins of

high-altitude aircraft not met with

in other fields of stress analysis

relating to aircraft . the material

may be roughly divided into shell

problems and plate problems, the

former being concerned with the

curved walls of the cabin or pressure

vessel and the latter being concerned

with small rectangular panels of

its walls, framed by stiffeners, but

not necessarily plane .

.I 1043

.T

on transverse vibrations of thin, shallow elastic shells .

.A

.B

.W

on transverse vibrations of thin, shallow elastic shells .

a numerical analysis is given for the solution of the

general equations of thin shells of revolution subjected to

rotationally symmetric pressure and temperature

distributions .

the basic differential equations are in a very general

form, which permits the geometry of the shells considered,

to be specified by discrete data points .

the analysis determines elastic stresses, strains and

displacements for multi-layer and multi-sectional shells of

revolution . surface loads, temperatures, thicknesses and material properties may vary arbitrarily in the meridional direction . temperatures and material properties can also vary through the thickness .

the solution is obtained by direct computation using a numerical method that employs two by two coefficient matrices,. and hence avoids the problems of slow convergence . the solution has been programmed in a semi-algebraic language which can be used on most high speed computers . comparisons of numerical solutions to known exact and approximate solutions of the thin shell equations are made to demonstrate the accuracy of this method .

.I 1044

.T

on the theory of thin elastic shells .

.A

reissner,e.

.B

reissner anniv. 1949, 231.

.W

on the theory of thin elastic shells .

general equations for the symmetrical finite deflection of a rotationally symmetric thin shell are first obtained . for small deflections these equations are reduced to a pair of equations for the change of slope of the shell surface and the product of the undeformed radius of the shell to the radial stress . this choice of dependent variable is shown to be advantageous . two cases of shallow shells give

particularly simple solutions .. parabolic shells of nth degree and
uniform thickness, and parabolic shells of mth degree with
thickness varying as mth power of the distance from the
apex . for the first case, the solutions can be expressed in
terms of cylinder functions,. for the second, in powers of
the paper concludes with a discussion of the asymptotic
solutions for small thickness .

.I 1045

.T

the bending strength of pressurized cylinders .

.A

zender,g.w.

.B

j. ae. scs. 29, 1962, 362.

.W

the bending strength of pressurized cylinders .

discussion of previously presented experimental data for the
loading of pressurized cylinders, in terms of membrane theory .

.I 1046

.T

the bending strength of pressurized cylinders .

.A

.B

.W

the bending strength of pressurized cylinders .

a theoretical solution is given
for the critical stress of

thin-walled cylinders loaded in torsion .

the results are presented

in terms of a few simple formulas and

curves which are applicable

to a wide range of cylinder dimensions

from very short cylinders of

large radius to long cylinders of small

radius . theoretical

results are found to be in somewhat better

agreement with

experimental results than previous theoretical

work for the same range

of cylinder dimensions .

.I 1047

.T

the bending strength of pressurized cylinders .

.A

.B

.W

the bending strength of pressurized cylinders .

be described here is attributed to the russian

investigator v. g. galerkin, whose original papers are

inaccessible to the present writer . his knowledge

of the method is derived from a description given in

a paper by e. p. grossman . grossman states

that the method was given by galerkin in his treatise

p. 897), and that applications to oscillation problems

were first made by v. p. lyskov . it is pointed
out by grossman that galerkin's process in applications
to mechanics leads to the same results as
lagrange's principle of virtual work, but employs a
special co-ordinate system .

the method of galerkin belongs to the same general
class as those of rayleigh and ritz, for it seeks
to obtain an approximate solution of a differential
equation with given boundary conditions by taking
a function which satisfies these conditions exactly,
and proceeds to specialise the function in such a
manner as to secure approximate satisfaction of the
differential equation . the selected function is
a linear combination of n independent functions, and
the coefficients are determined by a process of
integration .

the galerkin process can be considered from two
points of view, (a) simply as a means for the
approximate solution of differential equations, and
treatment of problems concerning the statics and
dynamics of elastic and other deformable bodies .
these two aspects are treated separately in parts 1
and 2 of the paper respectively, and will now
be briefly discussed .
which satisfies the boundary conditions, in
the differential equation be . since the result should be
zero, is the error in the differential equation .

then the galerkin process consists in choosing the n

coefficients in the function in such a manner that

n distinct weighted means of the error, taken throughout

a certain range of representation, shall all

be zero .

as a generalised force, and the multipliers used

to weight the errors are the virtual displacements

corresponding to increments of each of the generalised

co-ordinates in turn . thus the vanishing of the

weighted mean is here interpreted as the vanishing

of the virtual work in the appropriate displacement .

the degree of accuracy attaindd can be increased

indefinitely by increasing the number of

independent functions employed, but this entails a great

increase of labour . however, when the functions

are well chosen, an excellent approximation can be

obtained by the use of a very small number, as is

sufficiently shown by the examples included in this paper .

.I 1048

.T

a small deflection theory for curved sandwich plates .

.A

stein,m. and mayers,j.

.B

naca r.1008, 1951.

.W

a small deflection theory for curved sandwich plates .

a small-deflection theory that takes into account deformations

due to transverse shear is presented for the elastic-behavior

analysis of orthotropic plates of constant cylindrical curvature

with considerations of buckling included . the theory is

applicable primarily to sandwich construction .

.I 1049

.T

elastic constants for corrugated core sandwich plates .

.A

libove,c. and hubka,r.e.

.B

naca tn.2289, 1951.

.W

elastic constants for corrugated core sandwich plates .

the sandwich plate consisting of

corrugated sheet fastened between

two face sheets is considered . application

of existing theories to the

analysis of such a sandwich plate requires

the knowledge of certain

elastic constants . formulas and charts are

presented for the evaluation

of these constants . the formulas for three

of these constants were

checked experimentally and found to give

values in close agreement with

the experimental values .

.I 1050

.T

compressive buckling of simply supported curved plates

and cylinders of sandwich construction .

.A

stein,m. and mayers,j.

.B

naca tn.2601, 1952.

.W

compressive buckling of simply supported curved plates

and cylinders of sandwich construction .

theoretical solutions are presented

for the buckling in uniform

axial compression of two types of simply

supported curved sandwich

plates .. the corrugated-core type and the

isotropic-core type . the

solutions are obtained from a theory for

orthotropic curved plates in

which deflections due to shear are taken

into account . results are

given in the form of equations and curves .

.I 1051

.T

the stability of thin-walled unstiffened circular cylinders

under axial compression including the effects of internal

pressure .

.A

harris,l.a.

.B

j. ae. scs. 24, 1957, 587.

.W

the stability of thin-walled unstiffened circular cylinders

under axial compression including the effects of internal

pressure .

in the design of high-speed aircraft the importance of

unpressurized and pressurized monocoque cylinders necessitates a

reliable analysis procedure for the compressive buckling of

cylindrical shells . analysis by the classical small-deflection theory has

proved inadequate . recent large-deflection theoretical

treatments of the problem have shown reasonable correlation with

experiments but require a prior knowledge of the initial

imperfections of the cylinder . developed in this paper is a

semiempirical procedure which permits a compressive buckling analysis of

cylindrical shells with a knowledge of the cylinder geometry only .

this analysis is achieved by correlating experimental data

statistically with theoretical parameters .

in order to provide data not previously available, an extensive

series of axial compression tests of pressurized cylinders has been

performed . these data, together with all other known test data,

are analyzed semiempirically . in the analysis best-fit curves

are presented using theoretical parameters and shapes of curves

where applicable . unpressurized and pressurized cylinder

compressive buckling curves are then developed as 90 per cent probability curves from the test data . in general, these

statistically defined design curves are significantly lower than

previously available design curves .

.I 1052

.T

recent advances in the buckling of thin shells .

.A

william a. nash

.B

professor of engineering mechanics, university of florida

.W

recent advances in the buckling of thin shells .

the importance of the field of shell analysis is evidenced by the

fact that in august, 1959, the international union of theoretical

and applied mechanics conducted a symposium on the theory of thin

elastic shells in delft, holland . this special meeting was attened by

approximately 65 scientists in this field from 14 countries . this

symposium indicated that considerable interest currently exists in

such relatively new topics as the buckling of bimetallic shells,

pressurized shells, creep buckling, and dynamic buckling, as well as

in the more traditional problems involving isotropic shells of various

geometries .

.I 1053

.T

spherical cap snapping .

.A

keller,h.b. and reiss,e.l.

.B

j. ae. scs. 26, 1959, 643.

.W

spherical cap snapping .

a nonlinear boundary value problem for the determination of the rotationally symmetric deformations of a clamped spherical cap under external pressure is solved by finite differences . the numerical solutions are obtained by employing a previously developed iteration procedure . a special case of the difference equations is solved explicitly and yields a justification of the iteration method as well as insight into the properties of the more accurate numerical solutions .

buckled and unbuckled equilibrium states are obtained and the shape of the pressure-deflection curve which is usually assumed for these states is verified for a large class of caps . close estimates are given for the upper and lower buckling loads and an intermediate buckling load--i.e., the /dead-weight/ load . the stresses and deflections in the buckled and unbuckled states are examined and compared with an asymptotic solution valid in the interior of very thin shells . boundary layers are found to develop in the buckled states both as the loading increases and as the thickness of the shell decreases .

.I 1054

.T

iterative solutions for the non-linear bending of circular plates .

.A

keller,h.b. and reiss,e.l.

.B

comm. on pure and app. math. 11, 1958, 273.

.W

iterative solutions for the non-linear bending of circular

plates .

the authors study non-linear von karman equations

for bending of a thin circular plate under uniform normal

pressure . discussion is mainly concerned with plates

clamped at the edges and with zero radial displacement,

but analysis is valid for other edge conditions . solution is

by an iterative procedure whose convergence properties

are studied by means of integral equations . method is

then applied to finite difference formulation of the

differential equations in order to obtain numerical solutions .

numerical results are compared with previous work by

other authors and the advantages of the present method

are indicated .

.I 1055

.T

non-linear bending and buckling of circular plates .

.A

keller,h.b. and reiss,e.l.

.B

proc. 3rd u.s. nat. cong. app. mech. 1958, 375.

.W

non-linear bending and buckling of circular plates .

iterative solutions of finite difference

approximations of the non-linear von karman

plate equations are presented . results

are obtained for circular plates under a

variety of boundary conditions subjected to

either uniform lateral pressure or uniform

edge thrust .

the solution, carried out numerically on

the aec univac at new york university,

yields a complete description of stresses

and deflections for an apparently unlimited

range of load parameters . in addition, boundary

layer phenomena are discussed . for

computing purposes, this iterative method proves

to be superior to the previously used

power series method and may be applicable to

other non-linear problems .

.I 1056

.T

axisymmetric large deflections of circular plates subjected

to thermal and mechanical load .

.A

newman,m. and forray,m.

.B

proc. aerospace forum i session, s.m.f. paper ff-30, 1962, 56.

.W

axisymmetric large deflections of circular plates subjected to thermal and mechanical load .

this paper is concerned with the nonlinear axisymmetric analysis of circular plates with in-plane edge restraint . both temperature and mechanical loads are accommodated as an extension of investigations performed for the isothermal mechanical loading problem . an exact mathematical formulation within the framework of the v. karman large strain-displacement relations is developed . the equilibrium equations and boundary conditions are then derived by utilizing the calculus of variations for arbitrary axisymmetrical temperatures and normal distributed loading . the satisfaction of equilibrium and compatibility equations requires the solution of two simultaneous nonlinear ordinary differential equations subject to the prescribed boundary conditions . analytical solutions of such equations are apparently not possible and therefore numerical procedures must be employed .

a finite difference procedure utilizing /relaxed iterations,/ developed by h. keller and e. reiss, and employed by them for the solution of isothermal problems with apparently unlimited load parameter ranges, is used here for combined thermo-mechanical problems . numerical results are presented for the special case of a simply supported circular plate with radially immovable boundaries, subject to a uniform pressure and an arbitrary temperature variation through the thickness

tained for a large range of temperature and load

parameters . however, because of space limitations, only

a limited amount of data are presented in this paper .

.I 1057

.T

the uniform section disk spring .

.A

almen,j.o. and laszlo,a.

.B

asme trans. 58, 1936, 305.

.W

the uniform section disk spring .

the authors point out in this paper that initially coned

annular-disk springs of uniform cross section may be

proportioned to give a wide variety of load-deflection

curves not readily obtainable with the more conventional

forms of springs, and that, although the versatility of this

type spring has long been indicated, the formulas

available have not been presented in a manner to disclose

readily the effect of spring proportions on characteristics .

therefore the authors have derived the formulas presented

in this paper with the intention that the formulas will aid

the designer in arriving at suitable characteristics by

choice of spring geometry . these new formulas have

been in use for several years at the general motors

corporation research laboratories section, and their reliability has been checked by tests of springs used in a

variety of special test equipment .

.I 1058

.T

the conical disk spring .

.A

wempner,g.a.

.B

proc. 3rd u.s. nat. cong. app. mech. 1958, 473.

.W

the conical disk spring .

this paper presents approximate formulas

to describe the behavior of the conical

disk spring . it is assumed that the shallow

conical shell remains conical when

subjected to the axially symmetric edge loads .

the principle of stationary potential

energy is used to derive the relations between

load and deflection . formulas relating

the applied loads, stresses, and deflections are

given for several types of edge

constraint . the analysis is essentially a refinement

and extension of the previous work

of almen and laszlo .

.I 1059

.T

the nonlinear conical spring .

.A

schmidt,r. and wempner,g.a.

.B

asme trans. 81e, 1959, 681.

.W

the nonlinear conical spring .

the large symmetric deformations of shallow conical shells are of interest in the design of nonlinear conical disk springs . in most applications a uniformly distributed axial load acts at the inner and outer edges,. these edges are otherwise free . several approximations have been proposed to describe the behavior of these springs . a first approximation (1) is based on the assumption that meridional strains are negligible . this requires that the shell remain conical after deformation and also that the extensional strain of meridional lines on the middle surface vanish . another approximation (2) retains only the assumption that the shell remains conical . the first assumption satisfies neither of the two boundary conditions at the free edges,. the latter violates the condition of vanishing moment at the free edges . recently the authors presented a series solution (3) for a special case, namely, the case of an annular plate under similar loading . numerical solutions for the shallow conical shell under these conditions of load have also been obtained (4) . an examination of these results indicates that the meridional bending stresses are of much smaller magnitude than the circumferential bending stresses . hence the present analysis is based on the neglect of the meridional bending moment .

.I 1060

.T

buckled states of circular plates .

.A

keller,h.b., keller,j.b. and reiss,e.l.

.B

q. app. math. 20, 1962, 55.

.W

buckled states of circular plates .

authors discuss the thin elastic circular disk of constant

thickness subjected to a constant compressive thrust applied at its

edge . the analysis presented is based upon the nonlinear von

karman equations of plate theory and is applied to disks with

completely clamped and completely, simply supported edges .

.I 1061

.T

turbulent mixing of a rocket exhaust jet with a supersonic stream

including chemical reactions .

.A

vasilu, j.

.B

j. aero. sc. v. 29, january 1962, pp 19-28 .

.W

turbulent mixing of a rocket exhaust jet with a supersonic stream

including chemical reactions .

the equations for the turbulent mixing of a two-dimensional supersonic

jet issuing into an ambient supersonic stream are formulated . both streams consist of a mixture of chemically active and possibly reacting gases, therefore any heat release by chemical reaction is included,. the net mass rate of production of species is obtained on the assumption that the reaction rate constant is given by an expression reducible to the classical arrhenius law . the equations first given in terms of the x, and y coordinates, are expressed in dimensionless form and in terms of the x and coordinates, where is the stream function . the resulting expressions are all of the /heat conduction/ type,. they are put in a finite difference form by using the crank-nicolson method of substituting finite difference approximations for both the /time/ and /space/ derivatives . the mixture is assumed to consist of six species, namely h2o, h2, o2, co2, co, and n2, and the oxidation of h2 and co is assumed to take place according to a single-step chemical reaction . the solution of the problem is based on the simultaneous solution of 8n linear algebraic equations in 8n unknowns, n being the number of internal grid points at every step in the x-direction, and 8 the total number of unknowns at each grid point, namely velocity, temperature, and concentration for each of the six species . a method of obtaining initial and boundary conditions from available inviscid jet flow solutions is discussed . the equations are programed for calculation on an ibm-704 computer . finally, one typical case is considered, and plots of velocity, temperature, and concentration profiles are given for the initial stages of development of the mixing layer .

.I 1062

.T

an experimental and theoretical investigation of second-order wing-body interference at high mach number .

.A

wilby,p.g.

.B

ffa r.91, stockholm, 1962.

.W

an experimental and theoretical investigation of second-order

wing-body interference at high mach number .

the second-order wing-body interference theory

of landahl and beane is used in the

theoretical calculation of the pressure distributions

over the wing of a wing-body combination .

results are compared with experimental values

obtained from wind-tunnel tests, at a mach

number of 7.35, on a cone-cylinder non-lifting body

with a triangular wing of wedge section

set at incidences of 0, 3, 6 and 10 . it is shown

that interference effects can be very large

and can be calculated theoretically with good accuracy .

.I 1063

.T

on obtaining solutions to the navier-stokes equations

with high speed digital computers .

.A

russell,d.b.

.B

arc 23,797, 1962.

.W

on obtaining solutions to the navier-stokes equations

with high speed digital computers .

the purpose of this paper is

to show how to obtain steady

state solutions to the navier-stokes

equations on a high-speed digital

computer . first the relative merits

of various finite difference

formulae are discussed . thereafter

the main part of the paper is

concerned with the methods used to

solve the finite difference

equations and an investigation is made

of all the simpler iterative

methods .

.I 1064

.T

propeller slipstream effects as determined from wing

pressure distribution on a large-scale six-propeller

vtol model at static thrust .

.A

winston,m.m.

.B

nasa tn.d1509, 1962.

.W

propeller slipstream effects as determined from wing

pressure distribution on a large-scale six-propeller vtol model at static thrust .

during static-thrust tests of a

large-scale general research model having a

tilting wing and double-slotted flaps,

static-pressure measurements were made on

a wing segment behind one propeller to

survey the effects of the slipstream . for

the conditions of highest slipstream energy,

the hovering end point of aerodynamic

parameters for aircraft having vertical and

short take-off and landing capability

the tilt-wing configuration (zero

flap deflection) was a 6 spanwise variation

in effective angle of attack in a

span of slightly less than 1 propeller diameter .

effective changes in camber

on the tilt-wing configuration as a result of

slipstream rotation, the radial

velocity gradient, and the resultant spanwise

flow were negative and had a maximum

magnitude of less than 2-percent chord . for

the deflected-slipstream

configuration (double-slotted flaps deflected), effects

important to the hovering

performance were found, including a 40-percent spanwise

variation in effective thrust

recovery and a 20 spanwise variation in effective thrust turning .

.I 1065

.T

a free-flight investigation of ablation of a blunt

body to a mach number of 13 .1.

.A

winters,c.w., witte,w.g., rashis,b. and hopko,h.n.

.B

nasa tn.d1500, 1962.

.W

a free-flight investigation of ablation of a blunt

body to a mach number of 13 .1.

a five-stage rocket-propelled

research-vehicle system was flown to a maximum

mach number of 13.1 at an altitude of

approximately 78,000 feet to determine

ablation characteristics of teflon in free

flight . continuous in-flight

measurements were made using sensors developed by

the national aeronautics and space

administration . the sensors were located

on the blunted face of a nose cone

constructed from teflon, with one at the stagnation

point and two others at a surface

distance of 0.62 radius on opposite sides of the

stagnation point . the

ablated-length measurements were in close agreement with analytical predictions . the

analytical predictions, upon inclusion of the

pertinent material property values,

should be applicable to other materials as well

as teflon .

.I 1066

.T

wind tunnel measurements of aerodynamic damping derivatives

of a launch vehicle vibrating in free-free bending

modes at mach numbers from 0. 70 to 2. 87 and comparisons

with theory .

.A

hanson,p.w. and doggett,r.v.

.B

nasa tn.d1391, 1962.

.W

wind tunnel measurements of aerodynamic damping derivatives

of a launch vehicle vibrating in free-free bending

modes at mach numbers from 0. 70 to 2. 87 and comparisons

with theory .

the aerodynamic damping of a

flexibly mounted aeroelastic model

with a blunted conical nose and a

cylindrical afterbody was measured at

mach numbers from 0.70 to 1.20 at

several levels of dynamic pressure

and two weight conditions and at

mach numbers from 1.76 to 2.87 at one weight condition . the first two free-free flexible modes of vibration were investigated . also investigated at mach numbers from 0.9 to 1.2 was the aerodynamic damping in the first free-free modes of a model which had a /hammerhead/ nose (the base diameter of the blunted cone was greater than the diameter of the afterbody which necessitated a reflex angle downstream from the cone base) .

two basically different methods, the /electrical power-input/ and the /decaying oscillations/ methods were used to determine the damping and frequencies . the experimentally determined values are compared with some applicable theories . the results of the investigation indicate that the aerodynamic damping in the elastic modes of vibration was small for all configurations tested . the maximum aerodynamic damping measured in the first mode was on the order of damping . the aerodynamic damping was found to be even less for vibration

modes higher than the first .

reduced-frequency effects were found to be

negligible for the range investigated .

agreement of calculated

aerodynamic damping derivatives with the

experimental results was not good .

generally, the experimentally determined

derivatives were larger than

those predicted by the various theories

used . the bond-packard theory

appeared to give the best agreement for

the first free-free vibration

mode but gave the worst agreement for the

second mode . measurements

made on the configuration that had a

hammerhead nose indicated small

negative aerodynamic damping in the mach

number range from 0.95 to 1.00 .

aerodynamic stiffness effects were found

to be small and within the

experimental scatter . (wind-on frequency

determination was accurate

only to approximately 1 percent .)

.I 1067

.T

plastic stability theory of geometrically orthotropic plates and cylindrical shells .

.A

gerard,g.

.B

j. ae. scs. 29, 1962, 956.

.W

plastic stability theory of geometrically orthotropic

plates and cylindrical shells .

a linear eighth-order equilibrium differential equation for

plastic buckling of geometrically orthotropic thin cylindrical

shells is derived . this equation is used to obtain explicit

solutions for long flat plates and wide columns under axial

compression and moderate-length cylinders under external pressure,

torsion, and axial compression .

.I 1068

.T

instability analysis of cylindrical shells under hydrostatic

pressure .

.A

herrmann,g.

.B

nasa tn.d1510, 1962, 239.

.W

instability analysis of cylindrical shells under hydrostatic

pressure .

to determine the elastic buckling

pressure of simply supported

cylindrical shells subjected to lateral
and axial hydrostatic forces,
various versions of linear bending
theories have been employed in the
past . for certain shell dimensions,
however, the expressions commonly
used may yield substantially differing
results . in what follows,
recent work on this problem by
a. e. armenakas and the writer is
briefly reviewed . this work consisted
primarily in employing a general
bending theory of circular cylindrical
shells under the influence of
initial stress, developed earlier by
the same authors, to re-examine
the problem mentioned, and compare the
results with those of previous
investigations . the outcome was the
establishment of a simple but
accurate expression for the buckling
pressure applicable to a wide
range of shell dimensions .

.I 1069

.T

design and testing of honeycomb sandwich cylinders
under axial compression .

.A

cunningham,j.h. and jacobson,m.j.

.B

nasa tn.d1510, 1962.

.W

design and testing of honeycomb sandwich cylinders

under axial compression .

experimental results for 36 diameter honeycomb cylinders

fabricated with thin (0.010) aluminum faces and cores prove that it is

quite feasible to stabilize thin faces so they can be loaded beyond

the yield point . the effect of initial imperfections and the various

modes of failure are discussed .

.I 1070

.T

buckling of orthotropic and stiffened conical shells .

.A

singer,j.

.B

nasa tn.d1510, 1962,463.

.W

buckling of orthotropic and stiffened conical shells .

donnell type stability equations for thin circular orthotropic

conical shells are presented and solved for external pressure, axial

compression and combined loading . the solution is likewise applied

to stiffened conical shells . correlation with equivalent cylindrical

shells yields a simple approximate stability analysis for orthotropic

or ring-stiffened conical shells under hydrostatic pressure . the general instability of stiffened conical shells under hydrostatic

pressure is also analysed by a more accurate approach . preliminary

experimental results for buckling of ring-stiffened conical shells

under hydrostatic pressure are presented and discussed .

.I 1071

.T

stability of thin torispherical shells under uniform

internal pressure .

.A

mescall,j.

.B

nasa tn.d1510, 1962, 671.

.W

stability of thin torispherical shells under uniform

internal pressure .

the stability of the toroidal portion of a torispherical shell

under internal pressure is considered from the point of view of the

linear buckling theory . a detailed stress analysis of the prebuckled

shell is made employing asymptotic integration . the change in

potential energy of the shell is then minimized using a rayleigh-ritz

procedure for actual computation of the critical pressure . numerical

results reveal that elastic buckling may occur for very thin shells

whose material has a relatively high value of the ratio of yield stress

to elastic modulus .

.I 1072

.T

ignition and combustion in a laminar mixing zone .

.A

marble,f. and adamson,t.c.

.B

jet prop. 24, 1954, 85.

.W

ignition and combustion in a laminar mixing zone .

the analytic investigation of laminar combustion processes which are essentially two- or three-dimensional present some mathematical difficulties . there are, however, several examples of two-dimensional flame propagation which involve transverse velocities that are small in comparison with that in the principal direction of flow . such examples occur in the problem of flame quenching by a cool surface, flame stabilization on a heated flat plate, combustion in laminar mixing zones, etc . in these cases the problem may be simplified by employing what is known in fluid mechanics as the boundary-layer approximation, since it was applied first by prandtl in his treatment of the viscous flow over a flat plate . physically it consists in recognizing that if the transverse velocity is small, the variations of flow properties along the direction of main flow are small in comparison with those in a direction normal to the main flow . the analytic description of the problem simplifies accordingly . the present analysis considers the ignition and combustion in the laminar mixing zone between two parallel moving gas streams . one stream consists of a cool combustible mixture, the second is hot combustion products . the two streams come into contact at a given point and a laminar mixing process follows in which the velocity distribution is modified by viscosity, and the temperature and composition distributions by conduction, diffusion, and chemical reaction . the decomposition of the combustible stream is assumed to follow first-order reaction kinetics with temperature dependence according to the arrhenius law . for a given initial velocity, composition, and temperature distribution, the questions to be answered are .. (1) does the combustible material ignite,. and (2) how far downstream of the initial contact point does the flame appear and what is the detailed process of development . since the hot stream is of infinite extent, it is found that ignition always takes place at some point of the stream . however, when the temperature of the hot stream drops below a certain value, the distance required for ignition increases so enormously that it essentially does not occur in a physical apparatus of finite dimension . the complete development of the laminar flame front is computed using an approximation similar to the integral technique introduced by von karman into boundary layer theory .

.I 1073

.T

a practical method for numerical evaluation of solutions of partial differential equations of the heat-conduction type .

.A

crank,j. and nicolson,p.

.B

proc. cam. phil. soc., 43, 1947, 50.

.W

a practical method for numerical evaluation of solutions

of partial differential equations of the heat-conduction

type .

three approximate methods for the solution of the

nonlinear equation of heat flow in a medium where heat is

being generated by a chemical reaction are compared . the

equations are where

subscripts indicate partial differentiations and q, k, a are

.I 1074

.T

theoretical and experimental investigation of second-order supersonic

wing-body interference .

.A

landahl, m., drougge, g. and beane, b.j.

.B

j. aero. sc. v. 27, september 1960. pp 694-702 .

.W

theoretical and experimental investigation of second-order supersonic

wing-body interference .

approximate second-order solutions for the supersonic flow around

wing-body combinations are calculated, using two different theoretical models

small and the wing sweep small in comparison with that of the mach cone are considered . the analysis is restricted to such high mach numbers

that m-2 1, and an approximate formula common to the two models is then

found for the second-order interference term . this formula can also be

used to correct experimental pressure distributions for the effect of

nonuniformities in the wind-tunnel flow .

in order to test the theory, wind-tunnel experiments on non-lifting

cone-cylinder bodies in combination with wings of simple shapes were

performed . pressure distributions were measured at m 3 and m 4, both

around the bodies and on the wings separately, as well as in

combination, and it was found that the second-order interference was

predicted reasonably well by the simplified theory .

.I 1075

.T

an experimental and theoretical investigation of second-order supersonic

wing-body interference, for a non-lifting body with wings at incidence .

.A

wilby, p.g.

.B

aero. res. inst. of sweden, ffa report 87, 1960 .

.W

an experimental and theoretical investigation of second-order supersonic

wing-body interference, for a non-lifting body with wings at incidence .

pressure distributions on the wing of two wing-body combinations are

measured experimentally at mach numbers 3 and 4 with the wing at various

incidences in the range 0degree to 10degree . the results are compared

with theoretical results which include interference effects calculated

according to the second-order supersonic wing-body interference theory

due to landahl and beane /1/ . this theory, having been tested
previously for non-lifting wing-body combinations, is thus tested also for
wings at incidence . the agreement between theory and experiment is
found to vary with mach number and wing sweepback . for the higher mach
number and moderate sweepback the theory gives a good prediction of
pressure distribution, but for the most adverse condition of low mach
number and large sweepback the theory is found to overestimate the
interference effects . this is expected as the theory assumes the
sweepback of the wings is small compared with that of the mach line . an
empirical guide to the limit of application of the interference theory
is given . within this limit the agreement between theory and experiment
is found to deteriorate only a little with increase of incidence, over
the range tested .

.I 1076

.T

an approximate method for determining the displacement effects and
viscous drag of laminar boundary layers in two-dimensional hypersonic
flow .

.A

bertram, m.h.

.B

naca tn 2773, 1952 .

.W

an approximate method for determining the displacement effects and
viscous drag of laminar boundary layers in two-dimensional hypersonic
flow .

a simplified approximate theory is presented by means of which the laminar boundary layer over an insulated two-dimensional surface may be calculated, a linear velocity profile being assumed, and an estimate made of its effect in changing the pressure distribution over the profile upon which the boundary layer is formed . skin friction is also determined . comparisons of results from this theory are made with experimental results at a mach number of 6.86 and a reynolds number of

.I 1077

.T

a method of solution with tabulated results for the attached oblique shock wave system for surfaces at various angles of attack, sweep, and dihedral in an equilibrium real gas including the atmosphere .

.A

trimpi, r.l. and jones, r.a.

.B

nasa tr r-63, 1960 .

.W

a method of solution with tabulated results for the attached oblique shock wave system for surfaces at various angles of attack, sweep, and dihedral in an equilibrium real gas including the atmosphere .

a new method is derived for solving the attached oblique shock-wave system for surfaces at various angles of attack, sweep, and dihedral in any real gas in equilibrium . results are tabulated for the following ranges .. angle of attack, 0degree to 65degree,. angle of sweep, 0degree to 75degree,. angle of dihedral, 0degree to 30degree,. mach number, 3 to 30,. and /effective specific-heat ratio/ parameter, 1.10 to 1.67 . both the method and tabulated solutions are easily adaptable to flight in any gas or in the atmosphere of any planet . an illustrative example is presented based on the ardc 1956 model atmosphere .

.I 1078

.T

the steady flow of a viscous fluid past a circular cylinder at reynolds

numbers 40 and 44 .

.A

apelt, c.j

.B

a.r.c. r + m 3175 .

.W

the steady flow of a viscous fluid past a circular cylinder at reynolds

numbers 40 and 44 .

this paper describes the numerical solution of the complete

navier-stokes equations for the steady flow of an incompressible viscous fluid

of unlimited extent past a circular cylinder at reynolds number 40 . a

new device developed for the numerical solution is described . the

results of the investigation are ..

good agreement with experimental results .

higher reynolds numbers even though they may not exist in nature . a

solution has been obtained at reynolds number 44 but it has not been

carried to the same accuracy as the solution at reynolds number 40 .

portion of the cylinder continues to increase with reynolds number in

such steady-state solutions up to a reynolds number 44 and no indication

has been found that this process will not continue as the reynolds

number is increased beyond 44 .

.I 1079

.T

finite difference formulae for the square lattices .

.A

nicley,w.g.

.B

q.j.mech. app. math, 1, p35.

.W

finite difference formulae for the square lattices .

the paper gives approximate formulae

for derivatives (including combinations

like and), and integrals, of a function

of two independent variables, in terms

of its values at nodes of a square lattice,

primarily for use in the numerical solution

of partial differential equations . consideration

is given to the form, as well as to

the magnitude, of the leading terms in the

error, and what is believed to be for

most purposes optimum combinations are

thus selected for the simpler compact

sets of nodes .

.I 1080

.T

viscous flow round a sphere at low reynolds numbers . /l40/ .

.A

jenson, v.g.

.B

proc. roy. soc. series, a. v. 249, p. 346, 1959 .

.W

viscous flow round a sphere at low reynolds numbers . /l40/ .

relaxation methods are outlined, and the present problem formulated in

modified spherical polar co-ordinates . the results of calculations made

for r 5, 10, 20, 40 are presented in the form of stream function and

vorticity distributions,. and further results of pressure distributions,

velocity distributions, and drag coefficients, calculated from them .

these results are shown to compare favourably with experimental work,

showing a steady trend from symmetrical stokes's flow, towards boundary

layer flow . the phenomenon of separation of the forward flow and

development of a circulating wake, is explained and illustrated, the first

formation of a wake being at r 17 .

.I 1081

.T

numerical solution of the navier-stokes equations for the flow around a

circular cylinder at reynolds number 40 .

.A

kawaguti, m.

.B

j. phys. soc. japan, v. 8, no. 6 p. 747, 1953 .

.W

numerical solution of the navier-stokes equations for the flow around a

circular cylinder at reynolds number 40 .

the steady two-dimensional flow around a circular cylinder submerged in

a viscous fluid for the case r 40 is investigated, integrating

numerically the exact navier-stokes equations . the main results are as

follows . /i/ the steady flow solution exists even for the reynolds number as high as 40 . moreover, it seems that the solution goes over

smoothly to the solution of the kirchhoff discontinuous flow theory

which seems to be the limiting flow for the case r . /ii/ the flow

pattern and the coefficients of pressure and drag are in good agreement

with the experimental data .

.I 1082

.T

the flow past pitot tube at low reynolds numbers, part 1-dash the

numerical solution of the navier-stokes equations for steady viscous

axisymmetric flow, part 2-dash the effects of viscosity and orifice

size on a pitot tube at low reynolds numbers .

.A

lester, w.g.s.

.B

o.u.e.l. no. 136, a.r.c. 22, 070, f.m. 2983 .

.W

the flow past pitot tube at low reynolds numbers, part 1-dash the

numerical solution of the navier-stokes equations for steady viscous

axisymmetric flow, part 2-dash the effects of viscosity and orifice

size on a pitot tube at low reynolds numbers .

in this report numerical methods used to solve the navier-stokes

equations for steady viscous two-dimensional flow are extended to include

the case of axial symmetry . the equations and their finite difference

approximations are derived working in cylindrical polar co-ordinates

with the stokes' stream function and the vorticity as variables . a new

method of dealing with the boundary conditions is given .

the effects of viscosity and orlfice size on a blunt-nosed pitot tube have been theoretically investigated up to a reynolds number of ten,

where the reynolds number has been based on the radius of the tube .

results are expressed in terms of a pressure coefficient where p is the

pressure measured in the tube, p the density of the fluid, and p and u

the static pressure and velocity in an undisturbed flow at the position

of the tube .

the values of c for a blunt-nosed tube are found to be less than those

for tubes with hemispheroidal heads, but always greater than unity in

the range considered . the effect of the orifice size is to decrease c

as the orifice size increases, this decrease is very small but increases

with the reynolds number . at a reynolds number of ten the decrease is

at most five per cent of the value of c when there is no orifice .

it is suggested that the decrease of c below unity found in some

experimental investigations at a higher reynolds number could be due to the

effects of orifice size .

.I 1083

.T

an investigation of fluid flow in two dimensions .

.A

thom, a.

.B

a.r.c., r + m, 1194. 1928 .

.W

an investigation of fluid flow in two dimensions .

flow of an inviscid fluid . -dash there are in existence several methods

of obtaining numerical solutions to the two-dimensional flow of a

perfect fluid for given boundary conditions . part 2 of the present paper gives a method of obtaining a numerical solution for viscous steady flow

solution of the simpler problem, illustrating it with examples bearing

on the experimental work described in part iv .

.I 1084

.T

the flow past circular cylinders at low speeds .

.A

thom, a.

.B

proc. roy. soc. series a. vol. 141, p. 651, 1933 .

.W

the flow past circular cylinders at low speeds .

this paper deals chiefly with calculations and experiments on the flow

past circular cylinders, but the arithmetical methods of solution of the

equations of steady viscous flow proposed and used in section i, are

applicable to other equations and may be of interest .

.I 1085

.T

note on the convergence of numerical solutions of the navier-stokes

equations .

.A

thom, a. and apelt, c.j.

.B

a.r.c. r + m 3061. 1956 .

.W

note on the convergence of numerical solutions of the navier-stokes

equations .

a criterion is given for the convergence of numerical solutions of the
navier-stokes equations in two dimensions under steady conditions . the
criterion applies to all cases of steady viscous flow in two dimensions
and shows that if the local 'mesh reynolds number', based on the size of
the mesh used in the solution, exceeds a certain fixed value, the
numerical solution will not converge .

.I 1086

.T

a note on the numerical solution of fourth order differential
equations .

.A

woods,l.c.

.B

aero. quart. 5, 1954, 176.

.W

a note on the numerical solution of fourth order differential
equations .

an old numerical method
of solving fourth order differential equations
is put in relaxation form . the higher
order correction terms are included and the
technique is illustrated by an example .
the method has the advantage of being more
rapidly convergent than the usual relaxation
procedure for fourth order equations .
some comments are made on the numerical
solution of the viscous flow equation .

.I 1087

.T

convergence rates of iterative treatments of partial differential

equations .

.A

frankel, s.p.

.B

math. tables aids comp. v. 4. p. 65, 1950 .

.W

convergence rates of iterative treatments of partial differential

equations .

the development of high-speed digital computers has made feasible the

numerical solution by iterative methods of some partial differential

equations . the convergence rates of several such iterative methods are

estimated here . it is found that with the familiar elementary iterative

methods some quite simple problems require prohibitive computational

labor .

the iterative methods here considered are related to the various forms

of the southwell /relaxation method/ in that they involve successively

applied local corrections to improve an approximate solution . however,

these iterative methods are routinized in conformity with the

requirements of automatic computers while the relaxation method is flexible and

depends in an essential way on the skill of its practitioners .

.I 1088

.T

iterative methods for solving partial difference equations of elliptic

type .

.A

young, d.

.B

trans. amer. math. soc. vol. 76, p. 92, 1954 .

.W

iterative methods for solving partial difference equations of elliptic

type .

this paper considers linear systems /1/ where a includes matrices of a

sort frequently occurring in the solution of elliptic partial

differential equations by difference methods /in particular, a o/ . rewriting

superscript is number of iteration cycle/ are used to compute u

when u are used . also, one may /over-relax/ ..

ser. a. 210, 307-357 /1910/ who suggested changing from time to time

to speed up convergence . in the present paper over-relaxation /with

fixed w/ is combined with immediate introduction of newly-computed u's,

a la gauss-seidel . various theorems on convergence are proved$. in

particular, it is shown that there exists an ordering of the equations

and an optimum value wb such that in general /3/ converges much more

rapidly than the gauss-seidel method /w 1/ . means are suggested for

estimating wb,. the sensitivity of the rate of convergence to the choice

of w is studied . the paper concludes with a theoretical comparison of

gauss-seidel and the method proposed, /successive over-relaxation/, for

solving dirichlet's difference problem over a square using a high-speed

computing machine .

.I 1089

.T

aerodynamic characteristics of propeller-driven vtol aircraft .

.A

kirby,r.h.

.B

nasa tn.d730, 1961.

.W

aerodynamic characteristics of propeller-driven vtol

aircraft .

this paper discusses the two major configurations that are usually considered for achieving vtol while keeping the fuselage essentially horizontal - that is, the tilt-wing and the deflected-slipstream configurations .

because of the high turning losses incurred by deflected-slipstream configurations in hovering and because of the wing-stalling problem of the pure tilt-wing configurations during the transition, it appears that a combination of the two principles should be used . this tilt-wing and flap configuration should make use of a programed extensible-chord slotted flap together with a leading-edge high-lift device in order to avoid the performance and handling qualities problems associated with wing stalling during the transition while keeping the wing area as low as possible for efficiency in cruising flight .

.I 1090

.T

pressure distribution and force measurements on a vtol

tilting wing-propeller model . pt .ii, analysis of

results .

.A

currie,m.m. and dunsby,j.a.

.B

rep. lr-284, nat. res. council of canada, 1960.

.W

pressure distribution and force measurements on a vtol

tilting wing-propeller model . pt .ii, analysis of

results .

this report presents in graphical

form the results presented in

reference 1 from pressure distribution and force

measurements on a half-wing model

of a twin-engined tilt-wing vtol configuration .

the profound influence of the

slipstream on these results is discussed in some

detail .

.I 1091

.T

data from a static thrust investigation of a large

scale general research vtol-stol model in ground effect .

.A

huston,r.j. and winston,m.m.

.B

nasa tn.d397, 1960.

.W

data from a static thrust investigation of a large

scale general research vtol-stol model in ground effect .

the model was tested at two different elevations with the wing pivot at 1.008 and 2.425 propeller diameters above the ground . the slipstream of the propellers was deflected by tilting the wing and propellers, by deflections of large-chord trailing-edge flaps, and by combinations of flap deflection and wing tilt . tests were conducted over a range of propeller disk loadings from 7.41 to 29.70 pounds per square foot . force data for the complete model and pressure distributions for the wing and flaps behind one propeller were recorded and are presented in tabular form without analysis .

.I 1092

.T

wing-nacelle-propeller interference for wings of various spans . force and pressure distribution tests .

.A

robinson,r.g. and herrnstein,w.h.

.B

naca r.569, 1936.

.W

wing-nacelle-propeller interference for wings of various spans . force and pressure distribution tests .

an experimental investigation was made in the n. a. c. a. full-scale wind tunnel to determine the effect of wing span on nacelle-propeller characteristics and, reciprocally, the lateral extent of nacelle and propeller influence on a monoplane wing . the results provide a check on the validity of the previous research on nacelles and propellers with 15-foot-span wings tested in the

the scale propeller and the n. a. c. a. cowling

used in the former researches were tested in three typical
tractor locations with respect to a thick wing of 5-foot
chord and 30-foot span . the span was progressively
reduced to 25,20, and 15 feet and the same characteristics
were measured in each case .
  the efficiency factors--propulsive efficiency, nacelle
drag efficiency, and net efficiency--were obtained for each
wing length by means of force tests and the values are
compared to determine the effect of span .
pressure-distribution measurements show the lateral extent of the
nacelle interference and the propeller-slipstream effect
on the span loading for the various conditions . complete
polar curves and curves showing the variation of nacelle
drag with lift coefficient are also included .
  force and pressure-distribution tests concur in
indicating that, for engineering purposes, the influence of a
nacelle and of a propeller, in a usual combination, may
be considered to extend laterally on a wing the same
maximum distance, or about five nacelle diameters or two
propeller diameters outboard of their common axes . all
important effects of scale nacelle-propeller
combinations may be measured within practical limits of accuracy
by tests of a 15-foot-span wing .
.I 1093
.T
induced interference effects on jet and buried-fan
vtol configurations in transition .

.A

spreemann,k.+.

.B

nasa tn.d731, 1961.

.W

induced interference effects on jet and buried-fan

vtol configurations in transition .

recent investigations of some jet and buried-fan configurations

have indicated that in the transition speed range, configurations with

considerable area surrounding the jet or buried fan can encounter large

losses in lift and nose-up pitching moments due to the pressures induced

on the lower surfaces by the interaction of the jet and free-stream

flow . the obvious way of minimizing these effects is to reduce the

surface area surrounding the jets or buried fans, that is, to consider

these effects in the preliminary stages of the airplane design .

.I 1094

.T

investigation of the effects of ground proximity and

propeller position on the effectiveness of a wing with

large chord slotted flaps in redirecting propeller

slipstream downward for vertical take-off .

.A

kuhn,r.e.

.B

naca tn.3629, 1956.

.W

investigation of the effects of ground proximity and propeller position on the effectiveness of a wing with large chord slotted flaps in redirecting propeller slipstream downward for vertical take-off .

an investigation of the effects of ground proximity and propeller position on the effectiveness of a wing equipped with large-chord slotted flaps in redirecting the slipstreams from large-diameter propellers downward for vertical take-off has been conducted in a static-thrust facility at the langley aeronautical laboratory .

the results indicate that, with the propeller thrust axis on the wing chord plane, both the angle through which the slipstream is deflected and the ratio of resultant force to thrust are reduced as the ground is approached . at positions nearest the ground some of the loss in resultant force is regained . lowering the thrust axis below the wing chord plane reduces the adverse effects of the ground and also reduces the large diving moments associated with the slotted-flap arrangement .

the static-thrust efficiency of the

propellers is slightly reduced by

the ground effect .

.I 1095

.T

investigation of effectiveness of large-chord slotted

flaps in deflecting propeller slipstreams downward

for vertical take-off and low-speed flight .

.A

kuhn,r.e. and draper,j.w.

.B

naca tn.3364, 1955.

.W

investigation of effectiveness of large-chord slotted

flaps in deflecting propeller slipstreams downward

for vertical take-off and low-speed flight .

an investigation of the effectiveness

of a wing equipped with

large-chord slotted flaps in rotating the

thrust vector of propellers through

the angles required for vertical

take-off and for flight at very low speeds

has been conducted in the facilities

of the langley 300 mph 7- by 10-foot

tunnel .

under conditions of static thrust

and with zero incidence between the

thrust axis and the wing chord plane,

the slotted flaps were effective in

rotating the thrust vector upward about

than 10 percent of the thrust . when an

auxiliary vane was added above the

wing, the thrust vector was rotated upward

configuration, vertical take-off

could be achieved with an initial attitude

of 16 and at airplane weights

up to 90 percent of the total propeller

thrust . the addition of 10

incidence between the thrust axis and the wing

increased the upward rotation

of the thrust vector about 10 . for the

same turning angle, the diving

moments associated with the slotted-flap

configurations were approximately

twice as large as the diving moments of

the configurations with plain

flaps and two auxiliary vanes .

.I 1096

.T

qualitative measurements of the effective heats of

ablation of several materials in supersonic air jets

at stagnation temperature up to 11,000 f.

.A

rashis,b., witte,w.g. and hopko,r.n.

.B

naca rm. l58e22, 1958.

.W

qualitative measurements of the effective heats of

ablation of several materials in supersonic air jets

at stagnation temperature up to 11,000 f.

the effective heats of ablation of a number of materials were

derived from tests in supersonic air jets at stagnation temperatures

ranging from 2,000 f to 11,000 f . the materials included the plastics

teflon, nylon, lucite, and polystyrene,. the inorganic salts ammonium

chloride and sodium carbonate, several phenolic resins of varied resin

content and type of reinforcement,. and a melamine-fiber glass

laminate .

.I 1097

.T

experimental ablation cooling .

.A

bond,a.c., rashis,b. and levin,l.

.B

naca rm.l58e15a, 1958.

.W

experimental ablation cooling .

this paper presents the results

of an experimental investigation on

the ablation of a number of promising

materials for heating conditions

comparable to those which may be

encountered by unmanned reentry
satellite vehicles, as well as for higher
heating conditions comparable to
those associated with reentry ballistic
missiles . materials tested
included the plastics teflon, nylon,
and lucite,. the inorganic salts
ammonium chloride and sodium carbonate,.
graphite,. a phenolic resin and
fiber glass composition,. and the
commercial material haveg rocketon .
results of these tests indicated
heat-absorption capabilities which are
several times greater than those of
current metallic heat-sink materials .
the results with teflon showed that
for hemispherical noses there was no
apparent effect of size or
stagnation-point pressure on ablation rate
for the range of variables covered in
the tests . for flat-faced
configurations, however, there was a definite
increase in the ablation rate with
increased stagnation-point pressure .
the results for the several
materials tested at heating rates
associated with reentry ballistic

missiles showed considerable increase

in the effective heats of ablation

over the results obtained at lower

heating rates . this trend of

increased effectiveness with increased

heating potential is in

agreement with the predictions of ablation

theories . comparisons of the

results for several materials tested

at the higher heating rates showed

graphite to have the lowest ablation

rate of all materials tested .

.I 1098

.T

an experimental investigation of ablating material

at low and high enthalpy potentials .

.A

rashis,b. and walton,t.e.

.B

nasa tm.x-263, 1960.

.W

an experimental investigation of ablating material

at low and high enthalpy potentials .

the ablation performance characteristics

of a number of materials

were derived from tests conducted in

a mach number 2.0 ethylene-heated

high-temperature air jet having a

maximum stagnation enthalpy potential

of approximately 1,200 btu lb . the

tests were conducted with 6-

inch-diameter blunt nose shapes . the

surface of most of the materials after

testing was generally smooth and the

unablated portions of the specimens

were in appearance the same as before

testing . in all cases, the back

or inside surface of the specimens

exhibited no evidence of heating .

an evaluation of the enthalpy

potential effect was obtained by

comparison of the present data with

previous tests conducted, on the

in a subsonic arc-heated air

jet . the stagnation enthalpy potential

of this facility was

approximately 7,000 btu lb . for teflon, the

effective heat of ablation increased

from approximately 1,250 btu lb to

enthalpy potential was increased from

.I 1099

.T

a theoretical study of stagnation point ablation .

.A

roberts, l.

.B

nasa tr r-9, 1959 .

.W

a theoretical study of stagnation point ablation .

a simplified analysis is made of the shielding mechanism which reduces

the stagnation-point heat transfer when ablation takes place at the

surface . the most significant result of the analysis is that the

effective heat capacity of the ablation material increases linearly with

stream enthalpy . the automatic shielding mechanism is discussed and

the significant thermal properties of a good ablation material are given

parameters .

.I 1100

.T

an analytical investigation of ablation .

.A

rashis, b. and hopko, r.n.

.B

nasa tm x-300, 1960 .

.W

an analytical investigation of ablation .

an analytical procedure is described which enables the derivation of

effective heat of ablation relationships for any type of boundary layer

from transpiration cooling results . the procedure enables the inclusion

of such effects as the ratio of wall temperature to local stream

temperature, surface radiation, and surface combustion .

the predicted effective heats of ablation for a three-dimensional

laminar stagnation boundary layer for teflon material were in agreement with those derived from tests conducted at boundary-layer enthalpy potentials of 800 and approximately 7,000 btu/lb .
the predicted equilibrium surface temperatures on nonablating surfaces behind an ablating material were in agreement with the values derived from tests conducted with inconel cylinders having teflon hemispherical nose pieces .

.I 1101

.T

a sensor for obtaining ablation rates .

.A

winters, c.w. and bracalente, e.m.

.B

nasa tn d-800, 1961 .

.W

a sensor for obtaining ablation rates .
a variable-capacitance ablation-rate sensor which allows continuous measurements of ablation rates for teflon and similar polymers has been developed and tested in an ethylene-heated high-temperature jet at stagnation temperatures ranging from 2,400degree to 3,800degree f .
the data /length changes/ were measured by using the same telemeter equipment as that used in rocket-propelled flight vehicles . test
results indicate measurement error to be a maximum of 4 percent between the telemetered length changes and the length changes that were obtained from photographic records of the test .

.I 1102

.T

a five-stage solid fuel sounding rocket system .

.A

swanson, a.g.

.B

nasa memo 3-6-59 l, 1959 .

.W

a five-stage solid fuel sounding rocket system .

a five-stage solid-fuel sounding-rocket system which can boost a payload of 25 pounds to an altitude of 525 nautical miles and that of 100 pounds to 300 nautical miles is described . data obtained from a typical flight test of the system are discussed .

.I 1103

.T

pressures, densities and temperatures in the upper atmosphere .

.A

the rocket panel .

.B

phys. rev. vol. 88, december 1, 1952. pp 1027-42 .

.W

pressures, densities and temperatures in the upper atmosphere .

averaged and internally consistent values of atmospheric pressure, density, and temperature from the ground to an altitude of 219 km have been determined and compiled by the united states groups active in upper-atmospheric research by rockets . additional relevant data by similar groups engaged in research on meteors and on the anomalous propagation of sound are also included, particularly in a brief discussion of variations with time and with place of these three atmospheric parameters .

.I 1104

.T

aerodynamic heating of blunt nose shapes at mach numbers up to 14 .

.A

stoney, w.e.

.B

naca rm l58 e05a, 1958 .

.W

aerodynamic heating of blunt nose shapes at mach numbers up to 14 .

results are presented from recent investigations of the aerodynamic heating rates of blunt nose shapes at mach numbers up to 14 . data obtained in flight and wind-tunnel tests have shown that the flat-faced cylinder has about 50 percent the stagnation-point heating rates of the hemisphere over nearly the entire mach number range . tests made at a mach number of 2 on a series of bodies made up of hemispherical segments of varying radius of curvature showed that slight amounts of curvature can decrease the local rates at the edge of the flat-faced cylinders with only a slight increase in the stagnation rate . the total heat transfer to such slightly curved bodies is also somewhat smaller than the total heat transfer to flat-faced cylinders .

comparison of several tests with theoretical heating-rate distributions showed that both laminar and turbulent local rates can be predicted by available theories /given the pressure distribution about the body/ reasonably well, although the scatter of the available data still leaves open the choice between the theories at the edge of the bodies, where they usually differ .

tests on a flat-faced cylinder at a mach number of 2.49 and at angles
of attack up to 15degree showed the movement of the apparent stagnation
point from the center of the body to the 50 percent windward station at
creased about 30 percent while that near the leeward edge decreased
about 20 percent at 15degree angle of attack .
preliminary results on a concave nose have indicated the possibility
that this type of design may be developed to give heating rates
significantly lower than even the flat-faced cylinder rates . the test results
have also shown, however, the existence of an unsteady flow phenomenon
which can increase the heating rates to extremely high values .

.I 1105

.T

numerical solutions for supersonic flow of an ideal
gas around blunt two-dimensional bodies .

.A

fuller,f.b.

.B

nasa tn.d791, 1961.

.W

numerical solutions for supersonic flow of an ideal
gas around blunt two-dimensional bodies .
the method described is an inverse one,. the shock shape is chosen
and the solution proceeds downstream to a body . bodies blunter than
circular cylinders are readily accessible, and any adiabatic index can
be chosen . the lower limit to the free-stream mach number available in
any case is determined by the extent of the subsonic field, which in
turn depends upon the body shape . some discussion of the stability of

the numerical processes is given . a set of solutions for flows about

circular cylinders at several mach numbers and several values of the

adiabatic index is included .

.I 1106

.T

free-flight observation of a separated turbulent flow

including heat transfer up to mach 8. 5.

.A

mcconnell,d.g.

.B

nasa tn.d278, 1961.

.W

free-flight observation of a separated turbulent flow

including heat transfer up to mach 8. 5.

a turbulent boundary layer separated

by a forward-facing step was

observed on the cylindrical portion

of a hemisphere-cone-cylinder test

vehicle . tip blunting, producing a

shear flow, was found to induce

higher pressures on the cylindrical

portion than were predicted from

ballistic tunnel data of unblunted projectiles .

an approximate method for

predicting this blunt-body pressure

distribution was hypothesized . these

findings, along with the hypothesis,

were substantiated by a wind tunnel

test of a similar body . the peak pressure

ratios of the separation were

smaller in magnitude than flat plate

theory predicted because of the

effect of the shear flow . the decrement

in heating of the separated

flow, relative to the corresponding

attached flow, was found to compare

well with the expected results .

.I 1107

.T

the flow field over blunted flat plates and its effect

on turbulent boundary growth and heat transfer at a

mach number of 4. 7.

.A

tendeland,t.

.B

nasa tn.d689, 1961.

.W

the flow field over blunted flat plates and its effect

on turbulent boundary growth and heat transfer at a

mach number of 4. 7.

surface pressures, impact and

static pressure distributions in the

flow field over the plate, and local

heating rates were measured on a

flat plate with various leading-edge

diameters . the tests were conducted

at a mach number of 4.7 and a

free-stream reynolds number of 3.8x10 per

foot .

it was found that the shape of

the shock wave indicated the existence

of an outward deflection of the flow

over the plate . the flow deflection

caused an outward deflection of the

shock-wave asymptote of approximately

the shock-wave angle calculated

including boundary-layer growth . the

mach number distributions in the

shear layer evaluated from pitot and

static pressure surveys agreed with

predictions based on shock-wave shape .

the predicted turbulent

heat-transfer coefficients for the blunted

flat plates agreed with the

measured heat-transfer coefficients . a

comparison between the measured

heat-transfer coefficients for the blunted

flat plates and the calculated

coefficients for a sharp leading-edged

plate indicated that the

coefficients were highest near the leading edge of the most blunted plate . the

measured heat-transfer coefficients

dropped to approximately 80 percent

of the sharp-plate values at a

considerable distance from the leading

edge for all of the blunted flat plates .

.I 1108

.T

a study of second-order supersonic flow theory .

.A

vandyke,m.d.

.B

naca r.1071, 1952.

.W

a study of second-order supersonic flow theory .

an attempt is made to develop a second approximation to the

solution of problems of supersonic flow which can be solved by

existing first-order theory . the method of attack adopted is an

iteration process using the linearized solution as the first step .

for plane flow it is found that a particular integral of the

iteration equation can be written down at once in terms of the

first-order solution . the second-order problem is thereby

reduced to an equivalent first-order problem and can be readily

solved . at the surface of an isolated body, the solution reduces

to the well-known result of busemann . the plane case is

considered in some detail insofar as it gives insight into the

nature of the iteration process .

again, for axially symmetric flow the problem is reduced to a first-order problem by the discovery of a particular integral . for smooth bodies, the second-order solution can then be calculated by the method of von karman and moore . bodies with corners are also treated by a slight modification of the method . the second-order solution for cones represents a considerable improvement over the linearized result . second-order theory also agrees well with several solutions for other bodies of revolution calculated by the numerical method of characteristics .

for full three-dimensional flow, only a partial particular integral has been found . as an example of a more general problem, the solution is derived for an inclined cone . the possibility of treating other inclined bodies of revolution and three-dimensional wings is discussed briefly .

.I 1109

.T

unsteady laminar compressible boundary layers on an infinite plate with suction or injection .

.A

yang,k-t.

.B

j. ae. scs. 1959, 653.

.W

unsteady laminar compressible boundary layers on an infinite plate with suction or injection .

this study deals with unsteady compressible laminar

boundary layers on an infinitely extended porous plate . an integral
solution based on two types of assumed velocity and
temperature profiles is presented for the general case where the unsteady
free-stream velocity and rate of surface suction or injection are
both arbitrary . also indicated is an exact solution, applicable,
however, only to certain specific unsteady free-stream and
surface suction or injection variations . the reliability and range
of validity of the integral solutions is then established on the
basis of numerical results from the exact solution . finally,
several general qualitative conclusions of the unsteady effects
of free-stream velocity and surface suction or injection on laminar
boundary-layer behavior are made .

.I 1110

.T

on supersonic flow past a slightly yawing cone .

.A

stone,ah.

.B

j. math. phys. 27, 1948, 67.

.W

on supersonic flow past a slightly yawing cone .

this paper is concerned
with the motion of a circular cone, of
not too blunt an angle, through air at high
speed . if the direction of motion of
the cone coincides with its axis of symmetry,
the resulting air flow is well known .

here we consider the perturbation produced

by a small /yaw/--i.e., the case in

which the cone is moving not quite in the

direction of its axis . the results are

confirmed experimentally, and have applications

to ballistics, though we are not

concerned with the latter here,. they may

also be useful as providing a check on

various approximate methods of wider

applicability . the square of the yaw is

neglected--an approximation of which the

validity is discussed . (similar

methods can be applied to the second-order effects

of the yaw, which are also of ballistic

significance,. but the computations have not

yet been completed .) it should be

observed that, because of the lack of symmetry,

the flow will be neither

irrotational nor isentropic .

.I 1111

.T

some research on high speed flutter .

.A

garrick, i.e.

.B

3rd a.a.aero. conf. september 1951. r.a.s. 1952, pp 419-446j .

.W

some research on high speed flutter .

paper presents brief discussions of many topics currently of interest in

the flutter field . these include /a/ the sonic speed case, /b/

oscillating pressure field of propellers, /c/ wing flutter with various

configurations including effects of body modes, and /d/ propeller stall

flutter .

.I 1112

.T

unsteady aerodynamic forces on a slender body of revolution

in supersonic flow .

.A

bond,r. and packard,b.b.

.B

nasa tn.d859, 1961.

.W

unsteady aerodynamic forces on a slender body of revolution

in supersonic flow .

linearized slender-body theory

is applied to the computation of

aerodynamic forces on an oscillating,

or deforming, body in supersonic

flow . the undeformed body is a body

of revolution and the deformed body

is represented by movement of a line

through the centers of the cross

sections which are assumed to remain

circular . the time dependence is

based on sinusoidal motion .

for a body of vanishing thickness

the slender-body theory yields

the apparent mass approximation as it

is obtained for incompressible

crossflow around a cylinder .

both linearized slender-body theory

and the apparent mass

approximation are used to calculate the

pitching-moment coefficients on a rigid

slender body with a parabolic arc nose

cone, and these coefficients are

compared with some experimental results .

.I 1113

.T

an electronic apparatus for automatic recording of

the logarithmic decrement and frequency for oscillations

in the audio and subaudio frequency range .

.A

olsson,c.o. and orlik-ruckeman,k.

.B

aero. res. inst. sweden, r.52, 1954.

.W

an electronic apparatus for automatic recording of

the logarithmic decrement and frequency for oscillations

in the audio and subaudio frequency range .

an electronic apparatus for automatic

evaluation of the damping of a harmonic
oscillation has been designed and constructed .
the apparatus is based on the idea
of representing the harmonic damped
oscillation by a rotating vector on the screen
of a cathode-ray tube in such a way, that
the rate of decrease of the length of the
vector is a measure of the damping . the
results are obtained simultaneously with
the oscillation test as two numbers in
decimal digits, which are inversely
proportional to the logarithmic decrement and
the frequency, respectively . the
apparatus, which is named the /dampometer/,
has been used for some time for free
oscillation measurements of the dynamic
stability derivatives of aeroplane models
in windtunnels, and has proved to be very
satisfactory . it gives results of usually
higher accuracy than evaluation methods in
common use, and permits a most considerable
saving of time .

.I 1114

.T

steady and fluctuating pressures at transonic speeds
on two space-vehicle payload shapes .

.A

coe,c.f.

.B

nasa tm.x503, 1961.

.W

steady and fluctuating pressures at transonic speeds

on two space-vehicle payload shapes .

steady and fluctuating pressures

have been measured at mach numbers

which were varied from 0.6 to 1.2 on

two bodies of revolution typical of

two space-vehicle payload shapes, the

centaur and the able v .

the results of the investigation

showed that significant fluctuations

of pressure occurred on both bodies

between mach numbers of 0.75 and 1.00 .

the maximum fluctuations measured at

any mach number and angle of attack

occurred in the region of the normal

shock wave as a result of

shock-wave motion . large regions of unsteady

pressure also occurred as a result

of separation on the converging afterbody

of the able-v model . the maximum

pressure fluctuations occurring on the

bodies increased with increasing

angle of attack . for angles other than

are indicated since pressure fluctuations

were larger on the upper half

of the bodies than on the lower half .

no definite conclusions could be

drawn regarding the form of the

spectral densities of pressure

fluctuations in the region of the shock

wave . the spectral densities in regions

of separation following the

shock wave appeared flat except for some

increase in energy level below

due to slight model motions .

.I 1115

.T

measurements of aerodynamic forces for various mean

angles of attack on an airfoil oscillating in pitch

and on two finite-span wings oscillating in bending

with emphasis on damping in the stall .

.A

rainey,a.g.

.B

naca r.1305, 1957.

.W

measurements of aerodynamic forces for various mean

angles of attack on an airfoil oscillating in pitch

and on two finite-span wings oscillating in bending

with emphasis on damping in the stall .

the oscillating air forces on a two-dimensional wing

oscillating in pitch about the midchord have been measured at various

mean angles of attack and at mach numbers of 0.35 and 0.7 .

the magnitudes of normal-force and pitching-moment coefficients

were much higher at high angles of attack than at low angles of

attack for some conditions . large regions of negative damping

in pitch were found, and it was shown that the effect of increasing

the mach number from 0.35 to 0.7 was to decrease the initial

angle of attack at which negative damping occurred .

measurements of the aerodynamic damping of a 10-

percent-thick and of a 3-percent-thick finite-span wing oscillating in

the first bending mode indicate no regions of negative damping

for this type of motion over the range of variables covered . the

damping measured at high angles of attack was generally larger

than that at low angles of attack .

.I 1116

.T

general instability of stiffened cylinders .

.A

becker,h.

.B

naca tn.4237, 1958.

.W

general instability of stiffened cylinders .

theoretical buckling stresses

are determined in explicit form for

circular cylinders with

circumferential and axial stiffening . the

loadings are axial compression, radial

pressure, hydrostatic pressure,

and torsion . analyses were confined

to moderate-length and long

cylinders . the investigation was based

upon the use of a form of donnell's

equation derived by taylor which is

applicable to orthotropic cylinders .

the derivation of this equation is

presented in this report .

.I 1117

.T

stability of orthotropic cylindrical shells under combined

loading .

.A

hess,t.e.

.B

ars j. 31, 1961.

.W

stability of orthotropic cylindrical shells under combined

loading .

the increasing use of fiber and whisker

reinforced materials makes necessary the availability of

methods of analyzing cylinders and cones

composed of an orthotropic material . this paper treats

the buckling of such shells under a combination

of axial compression and uniform external pressure .
the differential equation approach of flugge is
used, and the resulting buckling equation is
presented in terms of axial and circumferential
bending rigidities, shear rigidity, poisson's ratio,
geometry parameters and mode shapes . design
curves are presented which allow quick
determination of critical loads on cylinders, and, by using
the equivalent cylinder concept, on conical shells
of small included angle . the curves also lend
themselves to /tailoring/ of materials to fit the load
carrying requirements of the structure .

.I 1118

.T

elastic stability of orthotropic shells .

.A

becker,h. and gerard,g.

.B

j. ae. scs. 29, 1962.

.W

elastic stability of orthotropic shells .

a small-deflection theory for general instability of orthotropic
circular cylindrical shells has been derived for external pressure,
torsion, and axial compression . for the first two types of
loading, comparison of the theory with experimental data for
orthotropic cylinders reveals agreement comparable with that
obtained for isotropic shells . for axial compression, experimental

data have been found to agree reasonably well with theory for

orthotropic cylinders, in contrast to the agreement usually

obtained for isotropic cylinders .

.I 1119

.T

plastic stability theory of thin shells .

.A

gerard,g.

.B

j. ae. scs. 24, 1957.

.W

plastic stability theory of thin shells .

considerable interest is currently centered on the

role of deformation and flow types of plasticity

theories in the solution of stability problems . for

thin flat plates, deformation theory combined with

classical stability theory appears to yield results which

are in substantially good agreement with test data .

on the other hand, flow or incremental theories appear

to require the introduction of initial imperfections in

order to obtain a satisfactory degree of correlation with

tests .

thus, in view of the current state of development of

plastic stability theory, it appears fruitful to exploit

the mathematical simplicity inherent in deformation

theory in the investigation of the plastic stability of

thin shells . although there may be theoretical objections to deformation theories as a class, test data on flat plates do suggest the predictive value of the results obtained from this theory .

in this paper, a set of equilibrium differential equations for the plastic buckling of thin shells of constant unequal radii is derived . this set of three equations applies to flat plates, cylinders, and spheres under any loading system leading to buckling . for particular problems such as buckling of cylinders under axial compression, torsion or lateral pressure, and spheres under external pressure, the set of equations can be reduced to a single eighth-order partial differential equation of the donnell type in terms of the radial displacement only . these donnell-type equations are used to obtain solutions for plastic buckling of spheres under external pressure and long and moderate length cylinders under lateral pressure or torsion loads . the limiting cases of a simply supported flat plate under compression or shear, represent the solutions for short cylinders under lateral pressure or torsion, respectively .

.I 1120

.T

a unified theory of plastic buckling of columns and plates .

.A

stowell,e.z.

.B

naca r.898, 1948.

.W

a unified theory of plastic buckling of columns and

plates .

on the basis of modern plasticity considerations, a unified

theory of plastic buckling applicable to both columns and plates

has been developed . for uniform compression, the theory

shows that long columns which bend without appreciable

twisting require the tangent modulus and that long flanges

which twist without appreciable bending require the secant

modulus . structures that both bend and twist when they

buckle require a modulus which is a combination of the secant

modulus and the tangent modulus .

.I 1121

.T

compressive and torsional buckling of thin-wall cylinders

in the yield region .

.A

gerard,g.

.B

naca tn.3726, 1956.

.W

compressive and torsional buckling of thin-wall cylinders

in the yield region .

based on assumptions which have

led to the best agreement between

theory and test data on inelastic

buckling of flat plates, a general set

of equilibrium differential equations

for the plastic buckling of

cylinders has been derived . these equations

have been used to obtain

solutions for the compressive and torsional

buckling of long cylinders in

the yield region .

test data are presented which indicate

satisfactory agreement with

the theoretical plasticity-reduction

factors in most cases . where a

difference in results exists, test data

are in substantially better

agreement with the results obtained by use of

the maximum-shear law rather

than the octahedral-shear law to transform

axial stress-strain data to

shear stress-strain data .

.I 1122

.T

on the role of initial imperfections in plastic buckling

of cylinders under axial compression .

.A

gerard,g.

.B

j. ae. scs. 29, 1962.

.W

on the role of initial imperfections in plastic buckling

of cylinders under axial compression .

in a recent paper lee treated the complex problem of the

plastic buckling and postbuckling behavior of an axially

compressed cylindrical shell containing initial imperfections,

representing an important step forward in our understanding of this

common, yet perplexing, structural element . lee drew two

major conclusions .. (a) even with initial imperfections the

incremental theory of plasticity considerably overestimates the

buckling strength as compared with the deformation theory,

which is in substantially good agreement with experiments, and

strength of cylindrical shells subject to axial compression are

significant .

it is the purpose of this note to discuss the second conclusion

in terms of lee's experimental and theoretical results, other

experimental data on inelastic buckling of 7075-t6

aluminum-alloy cylinders, and recent theoretical results on the inelastic

buckling of cylinders in the axisymmetric and circumferential

modes . in particular, this writer does not believe that lee has

proved that initial imperfections are important for the group

of cylinders that he has tested . on the contrary, it is believed

that initial imperfections are completely insignificant for this

group of cylinders although of probable significance in other

cases .

.I 1123

.T

an extension of donnell's equation for a circular cylindrical shell .

.A

tsu-tao loo

.B

associate professor of mechanics, rensselaer polytechnic institute,

troy, n.y.

.W

an extension of donnell's equation for a circular cylindrical shell .

in 1933, donnell succeeded in simplifying the equations of equilibrium

for a circular cylindrical shell

he obtained simple relations between the critical buckling shearing

stress and the physical properties of a thin circular cylinder under

pure torsion . his approach reduces the tedious computations involved

in the classical solutions and is still in good agreement with them .

furthermore, it is easy to show that the well-known classical solution

for critical compressive stress of the cylinder under axial compression

can readily be obtained from donnell's equation .

.I 1124

.T

design of missile bodies for minimum drag at very high

speeds - thickness ratio, lift, and center of pressure

given .

.A

strand,t.

.B

j. ae. scs. 1959, 568.

.W

design of missile bodies for minimum drag at very high speeds - thickness ratio, lift, and center of pressure given .

newtonian flow theory has been used to develop a procedure for the design of minimum drag bodies of revolution having a given thickness ratio and center of pressure .

it is shown that the optimum body shape is independent of lift . center of pressure location, however, exerts a powerful influence on both the shape of the body and on the drag coefficient at zero lift .

.I 1125

.T

collapse by instability of thin cylindrical shells under external pressure .

.A

windenburg,d.f. and trilling,c.

.B

asme trans. 56, 1934, 819.

.W

collapse by instability of thin cylindrical shells under external pressure .

this paper discusses the collapse by instability of thin-walled cylindrical vessels subjected to external pressure . the most important of the theoretical and empirical formulas that apply to this subject are presented in a common notation . a new and simple instability formula is developed .

three classes of tubes are considered .. tubes of infinite

length,. tubes of finite length with uniform radial

pressure only,. and tubes of finite length with both uniform

radial and axial pressure . collapsing pressures

calculated by the various formulas are presented in tabular

form as a means of comparing the formulas .

the formulas are discussed briefly and checked against

the results of tests conducted at the u. s. experimental

model basin for the bureau of construction and repair,

navy department .

this paper is a sequel to one previously published as a

part of the work of the a.s.m.e. special research

committee on the strength of vessels under external pressure .

.I 1126

.T

an engineer's conceptual approach to the buckling of

cylindrical shell (axial loading) .

.A

cunningham,j.h.

.B

ias paper 62-106, 1962.

.W

an engineer's conceptual approach to the buckling of

cylindrical shell (axial loading) .

by using the well known analogy between

the bending of a beam on an elastic

foundation and the axial symmetric

displacement of a cylinder, a physical
insight is obtained for the buckling of
cylindrical shells under axial
compression . the technique is equivalent
to classical small deflection theory
and provides good agreement with the more
elaborate solutions for the buckling
strength of various sandwich, multi-layered,
and orthotropic cylinders,
including the effects of internal pressure
or an elastic core .

.I 1127

.T

the buckling of sandwich type panels .

.A

hoff,n.h. and mautner,s.f.

.B

j. ae. scs. 1945, 285.

.W

the buckling of sandwich type panels .

fifty-one flat rectangular sandwich-type panels were tested
in edgewise compression with the unloaded edges of the panels
restrained by v-grooves . the sandwich consisted of papreg
faces and a cellular cellulose acetate core . the thickness of the
faces varied from 0.00675 to 0.02025 in.,. the core, from 0.066
to 0.741 in.,. the width of the panel, from 4 to 11 in . the length
of the panel was always 10.5 in . the buckled shape consisted of

a ripple of short wave length across the panel . it was either
symmetric, the two faces bulging out symmetrically according
to sine curves, or skew, the two faces deflecting in the same sense
according to sine curves having a phase angle of 90 .
  a strain energy theory of buckling is presented for both the
symmetric and the skew cases, and the buckling load in the
symmetric case is also calculated by integration of the differential
equation . the agreement between the theoretic and the
experimental buckling stress is reasonable, that between the predicted
and actual buckled shape good . a simple formula is developed
which permits a choice of the most suitable core material when the
mechanical properties of the face material are given .

.I 1128

.T

face wrinkling and core strength in sandwich construction .

.A

yuseff,s.

.B

j. ae. scs. 1960.

.W

face wrinkling and core strength in sandwich construction .
  the effect of initial waviness on the wrinkling of
faces in sandwich construction is studied . formulae are
derived to determine the failing stress when the faces
wrinkle due to failure of the core in tension, compression
or shear . the importance of core strength requirements
in maintaining surface smoothness is noted . a

comparison of theory with experiments is made, and the

agreement between the two is found to be reasonably good .

the strength of the core . williams has related the strength

of the core in tension and shear to an arbitrarily assumed

initial irregularity which, to ensure laminar flow in a wing

is assumed to have a maximum admissible value (initial

wave amplitude critical wavelength=0.0005 to 0.001) .

.I 1129

.T

general instability of a ring stiffened circular cylindrical

shell under hydrostatic pressure .

.A

bodner,s.r.

.B

j. app. mech. 24, 1957, 269.

.W

general instability of a ring stiffened circular cylindrical

shell under hydrostatic pressure .

the general instability load of a ring-stiffened, circular

cylindrical shell under hydrostatic pressure is determined

by analyzing an equivalent orthotropic shell . a set of

differential equations for the stability of an orthotropic

shell is derived and solved for the case of a shell with simple

end supports . the solution is presented in terms of

parameters of the ring-stiffened, isotropic shell, and a

relatively simple expression for the general instability load

is obtained . some numerical examples and graphs of results are presented . in addition, an energy-method

solution to the problem is outlined, and the energy and

displacement functions that could be used in carrying out a

rayleigh-ritz approximation are indicated .

.I 1130

.T

handbook of structural stability . pt .vi . strength

of stiffened curved plates and shells .

.A

becker,h.

.B

naca tn.3786, 1958.

.W

handbook of structural stability . pt .vi . strength

of stiffened curved plates and shells .

a comprehensive review of failure

of stiffened curved plates and

shells is presented .

panel instability in stiffened

curved plates and general instability

of stiffened cylinders are discussed .

the loadings considered for the

plates are axial, shear, and the

combination of the two . for the

cylinders, bending, external pressure,

torsion, transverse shear, and

combinations of these loads are considered .

general instability in stiffened
cylinders was investigated . for
bending and torsion loads, test data
and theory were correlated . for
external pressure several existing
theories were compared . as a result
of this investigation a unified theoretical
approach to analysis of
general instability in stiffened cylinders
was developed .

.I 1131

.T

the effect of axial constraint on the instability of
thin conical shells under external pressure .

.A

singer,j.

.B

j. app. mech. 29, 1962, 212.

.W

the effect of axial constraint on the instability of
thin conical shells under external pressure .

author considers elastic axial restraint which may be (1)
active from the beginning of loading and (2) active only at the onset
of buckling . buckling loads for the two cases are related by a
simple conversion factor . effect of the restraint on the
axisymmetric type buckling is negligible, but the amplification of the
critical load for the nonaxisymmetric type buckling may be very

large for type (1) restraint . design curves are included for a
range of l d.and restraint stiffness . results are of doubtful
value considering the known inadequacy of the linear theory of
buckling under axial compression .
.I 1132
.T
general instability of ring stiffened cylindrical shells
subject to external hydrostatic pressure - a comparison
of theory and experiment .
.A
galletly,d.d.
.B
j. app. mech. 25, 1958, 259.
.W
general instability of ring stiffened cylindrical shells
subject to external hydrostatic pressure - a comparison
of theory and experiment .
  tests are described of a number of machined-stiffened
cylinders subjected to external hydrostatic pressure, and the
observed general instability strengths compared with predictions
from theories of kendrick and nash . agreement with kendrick
was found rather good . results also are presented from
electrical strain gages which show in detail the growth of embryonic
lobes and nonlinear characteristics of deformation at the
threshold of buckling . weakening effects of imperfect circularity are
discussed .
.I 1133

.T

a simple extension of southwell's method for determining the elastic

general instability pressure of ring-stiffened cylinders subject

to external hydrostatic pressure .

.A

g. d. galletly and t. e. reynolds

.B

david taylor model basin,

washington, d. c.

.W

a simple extension of southwell's method for determining the elastic

general instability pressure of ring-stiffened cylinders subject

to external hydrostatic pressure .

a simple extension of southwell's method is presented whereby the

elastic general instability pressure of ring-stiffened cylindrical

shells subject to external hydrostatic pressure may be determined . an

actual application of the method is illustrated in the paper and the

results of several other examples are summarized .

.I 1134

.T

torispherical shells - a caution to designers .

.A

galletly,g.d.

.B

j. eng. for industry, 1959, 51.

.W

torispherical shells - a caution to designers .

it has recently become apparent, through a rigorous stress analysis of a specific case that designing torispherical shells by the current edition of the asme code on unfired pressure vessels can lead to failure during proof-testing of the vessel . the purpose of the present paper is to show in what respects the code fails to give accurate results . as an illustrative example, a hypothetical pressure vessel with a torispherical head having a diameter-thickness ratio of 440 was selected . the supports of the vessel were considered to be either on the main cylinder or around the torus . the vessel was subjected to internal pressure and the elastic stresses in it were determined rigorously and by the code . a comparison of the two revealed that the code predicted stresses in the head which were less than one half of those actually occurring . furthermore, the code gave no indication of the presence of high compressive circumferential direct stresses which exceeded 30,000 psi for practically the entire torus . if the head had been fabricated using a steel with a yield point of would have failed or undergone large deformations, whereas the code would have

predicted that it was safe . the code's rules
for torispherical heads are thus in need of
revision for certain geometries . the
implications of the foregoing results are currently
being studied by the asme,. in the interim,
however, designers should exercise care in
applying the code to torispherical shells .

it is also shown in the paper that the use
of the membrane state as a particular
solution of the differential equations is not a
good approximation for toroidal shells of the
type considered .

.I 1135

.T

limit analysis of symmetrically loaded thin shells
of revolution .

.A

drucker,d.c. and shield,r.t.

.B

j. app. mech. 26, 1959, 61.

.W

limit analysis of symmetrically loaded thin shells
of revolution .

the yield surface for a thin cylindrical shell
is shown to be a very good approximation to
the yield surface for any symmetrically
loaded thin shell of revolution . hexagonal

prism approximations to this yield surface,

appropriate for pressure vessel analysis, are

described and discussed in terms of limit

analysis . procedures suitable for finding

upper and lower bounds on the limit pressure

for the complete vessel are developed and

evaluated . they are applied for illustration

to a portion of a toroidal zone or knuckle held

rigidly at the two bounding planes . the

combined end force and moment which can be

carried by an unflanged cylinder also is discussed .

.I 1136

.T

design of thin walled torispherical and toriconical

pressure - vessel heads .

.A

shield,r.t. and drucker,d.c.

.B

j. app. mech. 28, 1961, 292.

.W

design of thin walled torispherical and toriconical

pressure - vessel heads .

the failure under hydrostatic test of a large storage vessel

designed in accordance with current practice stimulated earlier

analytical studies . this paper gives curves and a table useful for

the design and analysis of the knuckle region of a thin

torispherical or toriconical head of an unfired cylindrical vessel . a simple but surprisingly adequate approximate formula is presented for

the limit pressure, np, at which appreciable plastic deformations

occur ..

where p is the design pressure, is the yield stress of the

material, and n is the factor of safety . the thickness t of the knuckle

region is assumed uniform . upper and lower bound calculations

were made for ratios of knuckle radius r to cylinder diameter d

of 0.06, 0.08, 0.10, 0.12, 0.14, and 0.16, and ratios of spherical cap

radius l to d of 1.0, 0.9, 0.8, 0.7, and 0.6 . toriconic1a heads may

be designed or analyzed closely enough by interpreting in

table 1 as the complement of the half angle of the cone .

.I 1137

.T

on the theory of thin elastic toroidal shells .

.A

clark,r.a.

.B

j. math. phys. 29, 1950, 146.

.W

on the theory of thin elastic toroidal shells .

the author obtains asymptotic solutions to the problem

of rotationally symmetric small deflection of thin toroidal

elastic shells . he first reduces the problem to that of

integrating a single linear nonhomogeneous ordinary differential

equation involving two parameters . asymptotic formulae

for the complementary function are obtained by applying

the general method of langer (trans.amer.math.soc.33, advantage of yielding results valid near the points where

the tangent plane is perpendicular to the axis of revolution,

where the methods of asymptotic integration customary in

shell theory fail (see the preceding review) . for two

problems in which only the complementary function is required,

the author's results are compared with those obtained by

wissler (dissertation, zurich, 1916) by a method of power

series expansion,. the agreement is within 4 or better . the

author observes that the usual method of obtaining

asymptotic expressions for a particular integral, being based on

using as an approximation the complementary function

obtained from the membrane theory, will fail near points

where the tangent plane is perpendicular to the axis of

revolution . he therefore introduces a new method, which

he states was developed jointly with e. reissner . he applies

his results to the cases of an joint loaded symmetrically

and parallel to its axis, a corrugated pipe subject to axial

load, and a corrugated cylinder subject to axial pressure .

many numerical calculations are involved and there are two

tables of functions occuring in the solutions .

.I 1138

.T

asymptotic solutions of toroidal shell problems .

.A

clark,r.a.

.B

quart. app. math. 16, 1958, 47.

.W

asymptotic solutions of toroidal shell problems .

method of asymptotic integration developed by e. reissner and author is refined, and solutions previously obtained for problems of bending of curved tube and of a toroidal expansion joint subject to an axial force are generalized and extended . results are compared to those obtained by l. beskin . for large values of a certain parameter, agreement is good .

.I 1139

.T

the effect of entrance velocity on the flow of a rarefied gas, through a tube .

.A

pond,h.l.

.B

j. ae. scs. 29, 1962, 917.

.W

the effect of entrance velocity on the flow of a rarefied gas, through a tube .

the flow of a rarefied gas through a circular tube is considered . molecules entering the tube have a mass velocity directed down the tube, as well as a randomly directed thermal velocity . it is assumed that the conditions for free-molecule flow hold, and that molecules striking the tube wall are reflected diffusely . the mass velocity and tube dimensions are restricted only by the limitation to free-molecule flow . the theory is illustrated by an example of the effect of an entrance tube on the measurement of pressure and density by a gage placed on a satelite .

.I 1140

.T

shock-standoff distance for spherical bodies at high

mach numbers .

.A

chaudhury,p.r.

.B

j. ae. scs. 29, 1962, 745.

.W

shock-standoff distance for spherical bodies at high

mach numbers .

usaf-sponsored development of a simple expression for

shock-standoff distance, by consideration of an apropriate specific-heat

ratio behind the shock . the theory gives predictions which are

almost as accurate as those of the van dyke and hays methods .

.I 1141

.T

the wake behind an oscillating vehicle .

.A

zeisberg,s.l.

.B

j. ae. scs. 29, 1962, 1344.

.W

the wake behind an oscillating vehicle .

the incompressible laminar far wake behind an oscillating

vehicle is analyzed with the use of the oseen linearization, and the assumption that the wake cross section is axisymmetric .

time-wise drag variations are thus represented as changes in the

wake diameter and velocity defect . it is shown that the

unsteadiness in this flow can be predicted by a quasi-steady theory .

.I 1142

.T

effect of wall divergence on sonic flows in solid wall

tunnels .

.A

tirumalesa,d. and satyanarayana,b.

.B

j. roy. soc. 66, 1962, 125.

.W

effect of wall divergence on sonic flows in solid wall

tunnels .

the effect of wall divergence (in excess of the normal

value given for compensating the boundary layer growth

on the tunnel walls), on the simulation of sonic flows in

solid wall wind tunnels is presented in this note which is

a condensed version of ref. 1 .

.I 1143

.T

a one-foot hypervelocity shock tunnel in which high-enthalpy

real gas flows can be generated with flow times of

about 180 milliseconds .

.A

cunningham,b.e. and kraus,s.

.B

nasa tn.d1428, 1962.

.W

a one-foot hypervelocity shock tunnel in which high-enthalpy

real gas flows can be generated with flow times of

about 180 milliseconds .

a shock tunnel is described in

which high-enthalpy, real-gas air

flows can be generated with flow times

of about 180 milliseconds . this

shock tunnel is operated with a

combustion-heated driver gas and consists

of a combustion chamber, shock tube,

supersonic nozzle, test section, and

vacuum tank . an essential feature of

this shock tunnel is a means for

achieving a constant-pressure air reservoir

for the duration of the test .

air streams with velocities in excess of

achieved at a mach number of about 10 .

the corresponding stream total

enthalpy is about 4,500 btu lb and the

stagnation pressure is 3.25 psia .

.I 1144

.T

slipstream flow around several tilt-wing vtol aircraft models

operating near the ground .

.A

william a. newsom, jr., and louis p. tosti

.B

technical note d-1382

.W

slipstream flow around several tilt-wing vtol aircraft models

operating near the ground .

a collection of data from a number of brief investigations made with three different models to determine the character of the slipstream flow along the ground is presented for multiple-propeller tilt-wing vtol aircraft configurations operating near the ground . in general, the tests involved tuft surveys and slipstream dynamic-pressure measurements for several tilt-wing vtol models . a more extensive series of tests, including some measurements of the erosion of gravel by the slipstream and some measurements of the unsteady rolling, yawing, and pitching moments, was also made on one of the models operating in the hovering condition near the ground .

the results of the flow studies indicated the presence of a stronger and deeper slipstream flow along the center line of the aircraft, and to some extent along parallel planes between adjacent propellers (on one wing), than to the side of the aircraft . this effect is caused by an intensification of the individual slipstreams as they meet at the planes of flow symmetry . the intensified flow along the center line of the aircraft is amplified by the presence of the fuselage and causes the dynamic pressure to be greater in front of the aircraft than would be expected on the basis of the slipstream of the individual propellers .

in the erosion tests it was found that gravel, if sufficiently small,

was rapidly eroded by the slipstream and that this gravel could be
thrown high into the air if it struck even very small fixed obstacles on
the ground (obstacles with a height less than the diameter of the
gravel) . results of the investigation of moment fluctuations
indicated that there are large, erratic variations of rolling, yawing,
and pitching moments and that the propellers, reacting to an erratic
inflow from the recirculating slipstream, are the primary source of
these moments .

.I 1145

.T

buckling of core-stabilized cylinders under axisymmetric external
loads .

.A

don o. brush and bo o. almroth

.B

lockheed missiles and space company

.W

buckling of core-stabilized cylinders under axisymmetric external
loads .

an equation is derived for the elastic stability of a circular
cylindrical shell which is filled with a soft elastic core and is subjected to
general axially-symmetric lateral pressure combined with a central
axial force . numerical results are given for three lateral pressure
distributions of interest in rocket motor case analysis.. uniform
pressure, linearly varying pressure, and a circumferential band
of pressure located at an arbitrary distance from one end of the
cylinder . comparison is made with results of previous theoretical

and experimental investigations, where available .

.I 1146

.T

thermal buckling of cylinders .

.A

melvin s. anderson

.B

nasa langley research center

.W

thermal buckling of cylinders .

several theoretical and experimental investigations on the buckling

of cylinders due to both axial and circumferential thermal stresses are

reviewed . differences that exist among the various results are

discussed and areas of future work are indicated .

.I 1147

.T

heat transfer to bodies traveling at high speed in the upper

atmosphere .

.A

jackson r. stalder and david jukoff

.B

report 944

.W

heat transfer to bodies traveling at high speed in the upper

atmosphere .

a general method has been developed, using the methods of kinetic

theory, whereby the surface temperatures of bodies can be calculated for steady flight at any speed in a rarefied gas . the particular solution was made for a flat plate., however, the calculations can be easily extended to bodies of arbitrary shape .

it was found that the aerodynamic heating problem in the absence of solar radiation, that is, for the case of nocturnal flight, becomes of negligible importance at altitudes of 125 miles and higher and up to steady flight speeds of 36,000 feet per second . the effect of solar radiation, for the case of daytime flight, becomes increasingly important as the flight altitude is increased . at an altitude of 150 miles and higher, solar radiation is the predominating factor that determines skin temperature . owing to the strong effect of solar radiation on skin temperatures at high altitudes, the desirability of nocturnal flight is indicated in order to minimize skin temperatures .

in order to maintain low skin temperatures, it was found that the angle of inclination of the body with respect to the flight path should be kept as small as possible . this may be accomplished in practice by designing the body to be finely tapered and by flying the body at small angles of attack .

it is pointed out that skin temperatures may be reduced by insuring thermal contact between portions of the skin inclined at positive and negative angles with respect to the flight path . as much surface as possible should be inclined at negative angles . practically, this may be accomplished by boattailing the body .

in the event that an internal skin-cooling system is employed, it is shown that the rate of internal cooling must be of the same order of magnitude or greater than the rate at which heat is lost naturally by emitted radiation . if the cooling rate is below the natural

radiation rate, cooling has little effect upon skin temperatures .
it is shown that, in the case of a missile designed to fly over a wide
range of altitudes and speeds, it is desirable to make the emissivity
of the skin as high as possible . this conclusion, however, is
based upon a skin surface for which the emissivity is independent
of the wave length of the emitted and absorbed radiant energy .
a possible method of reducing surface temperatures is indicated by
the decrease in skin temperature which accompanies a decrease in
thermal accommodation coefficient . this phenomenon may be
used to advantage if it is possible to decrease the accommodation
coefficient by altering the surface characteristics of the skin .

.I 1148

.T

knudsen flow through a circular capillary .

.A

w. c. demarcus and e. h. hopper

.B

carbide and carbon chemicals company, k-25 plant, post office box p,
oak ridge, tennessee

.W

knudsen flow through a circular capillary .
the problem of knudsen flow through a circular capillary has been
often discussed, usually by the momentum transfer method . however,
p. clausing gave a rigorous formulation for the problem and obtained an
integral equation for which he gave an approximate solution . from
time to time the accuracy of clausing's solution has been questioned
and since clausing did not give a rigorous estimate of his error we

have reinvestigated the problem .

.I 1149

.T

similar temperature boundary layers .

.A

dragutin stojanovic

.B

university of beograd, jugoslavia

.W

similar temperature boundary layers .

conditions for the existence of similar solutions are known for (a) two-dimensional, incompressible, steady and nonsteady laminar boundary layers and (b) three-dimensional, incompressible, steady, laminar boundary layers for a body of revolution rotating in a fluid at rest or a body of revolution in a rotating fluid flow . corresponding conditions for the existence of similar temperature boundary layers in both cases are given for constant and variable wall temperatures . the general conclusion is that, in all these cases, with or without viscous heating, and with constant wall temperature, conditions for the existence of similar velocity boundary layers are at the same time the conditions for the existence of similar temperature boundary layers . if the wall temperature is variable, the conditions for the existence of similar velocity boundary layers are at the same time the conditions for the existence of similar temperature boundary layers if the wall temperature varies as a power of the local free-stream velocity or surface velocity . numberical solutions are given for the nondimensional temprature distributions function and the nondimensional temperature gradient at the wall for several prandtl numbers in the case

of a rotating flow over an infinite plate at rest .

.I 1150

.T

preliminary results of density measurements from an air force satellite.

.A

mcisaac, j., pond, h. and stergis, g.

.B

air force, cambridge lab., 63-24. february, 1963 .

.W

preliminary results of density measurements from an air force satellite.

atmospheric density was determined from a singly mounted ionization

gauge flown on an air force satellite . included is a brief description

of the experiment and theory as well as a discussion of some of the

problems involved in performing these measurements . density data are

given for the altitude range of 370 to 400 km during early morning hours

for the two days 17 and 18 june 1961 . results are compared with those

of the 1961 revised u.s. standard atmosphere .

.I 1151

.T

experiments on supersonic blunt-body flows .

.A

kendall,j.m.

.B

j.p.l.prog. r.20-372, 1959.

.W

experiments on supersonic blunt-body flows .

recently, progress has been made in the theoretical calculation of the inviscid flow between the detached shock wave and the surface of a blunt body travelling at supersonic speed . detailed experimental data are needed for comparison . experiments have been made in the supersonic wind tunnels of the jet propulsion laboratory on spheres, disks, and blunted cones over the mach number range 1.8 to 5.0 in air . surface pressures, shock wave shape and detachment distance, certain velocity gradients, the sonic line location, and some streamline paths were determined . the sonic line is found as the trace of the termination point of a weak shock generated by a probe ahead of the model . for a sphere, good agreement between theory (of van dyke) and experiment is found .

.I 1152

.T

on periodically oscillating wakes in the oseen approximation .

.A

lin,c.c.

.B

studies in maths . and mechs .,

presented to r.von mises, p 170, academic press, 1954.

.W

on periodically oscillating wakes in the oseen approximation .

studies in maths . and mechs .,

the oscillating vortex wake behind an obstacle at reynolds numbers of order 10 is studied by means of the oseen approximation .

.I 1153

.T

a study of the simulation of flow with free stream mach number 1 in a

choked wind tunnel .

.A

spreiter, j.r., smith, d.w. and hyett, b.j.

.B

nasa tr r-73, 1960 .

.W

a study of the simulation of flow with free stream mach number 1 in a

choked wind tunnel .

the degree to which experimental results obtained under choking

conditions in a wind tunnel with solid walls simulate those associated with

an unbounded flow with free-stream mach number 1 is investigated for the

cases of two-dimensional and axisymmetric flows . it is found that a

close resemblance does indeed exist in the vicinity of the body, and

that the results obtained in this way are generally at least as accurate

as those obtained in a transonic wind tunnel with partly open test

section . some of the results indicate, however, that substantial

interference effects, particularly those of the wave reflection type, may

be encountered under certain conditions, both in choked wind tunnels and

in transonic wind tunnels, and that the reduction of these interference

effects to acceptable limits may require the use of models of unusually

small size .

.I 1154

.T

on the influence of wall boundary layers in closed transonic test

sections .

.A

berndt s.b.

.B

ffa rep. 71, stockholm, 1957 .

.W

on the influence of wall boundary layers in closed transonic test sections .

the boundary layers at the test section walls of a transonic wind tunnel are known to reduce the wall interference . in the present paper this effect is studied by means of small perturbation theory, assuming viscosity to be negligible when perturbing a turbulent boundary layer . an approximation for thin boundary layers leads to a modified boundary condition at the wall of the test section, expressing the normal streamline slope induced by changes in mass flow density and crossflow within the boundary layer . this boundary condition is applied to the linearized equations of subsonic flow and to the non-linear transonic equations at choking, the cases of plane and circular test sections only being treated in detail .

the results of linear theory show that all corrections except the three-dimensional angle-of-attack correction are considerably reduced by the presence of the boundary layers at mach numbers greater than 0.9, the essential part of their influence being due to the change of mass flow density with pressure . in the case of choking the analysis indicates that the presence of boundary layers will increase the maximum model size for which the flow can be interpreted as corresponding to mach number one in free flight . finally, the technique of using artificial thickening of the wall boundary layers for a reduction of wall interference is considered, though without reaching a definite

conclusion as to its value as compared to other techniques .

.I 1155

.T

some experimental investigations on the influence of wall boundary

layers upon wind tunnel measurements at high subsonic speeds .

.A

petersohn, e.g.m.

.B

ffa n.r. 44, stockholm, 1952 .

.W

some experimental investigations on the influence of wall boundary

layers upon wind tunnel measurements at high subsonic speeds .

pressure distribution measurements and drag determination by means of

balance measurements have been carried out for a number of models at

high subsonic velocity in wind tunnels, where the boundary layer of the

walls has been varied . within the investigated range it appeared that a

thickening of the boundary layer reduced the disturbing influence of

the walls, which also caused an increase of the choking mach number .

the phenomenon described should be of a certain importance from the

point of view of wind tunnel technique, since it is possible to increase

the choking velocity for a given model by means of thickening the

boundary layer .

.I 1156

.T

experimental investigation of attenuation of strong

shock waves in a shock tube with hydrogen and helium as driver gases .

.A

jones,j.j.

.B

nasa tn.4072, 1957.

.W

experimental investigation of attenuation of strong

shock waves in a shock tube with hydrogen and helium

as driver gases .

an experimental investigation

has been made of the attenuation of

strong shock waves in air in a shock

tube . time-history measurements

were made of the static pressure at

several stations in the wall of the

tube . the internal diameter of the

tube is 3.75 inches . shock-

wave-velocity data were taken for a distance

along the tube of about 120 feet .

the range of the shock-wave mach number

covered was from 5 to 10 and

the initial pressure ahead of the shock

wave varied from 5 to 100

millimeters of mercury . hydrogen and helium

were used as driver gases .

a helium-driven shock wave was found

to decay only about one-half

as rapidly as a hydrogen-driven shock wave .

the pressure level had

little effect on the attenuation rate of

a shock wave of given strength

for the pressure range investigated . the

static-pressure measurements

indicated that a severe pressure gradient

existed in the latter portion

of the air flow . this gradient limits the

testing time useful for

obtaining reliable aerodynamic data .

.I 1157

.T

hypersonic shock tunnel .

.A

nagamatsu, h.t. et al.

.B

ars j.v. 29, may 1959, pp 332-340 .

.W

hypersonic shock tunnel .

a hypersonic shock tunnel has been developed for obtaining fluid

mechanic information at the high mach numbers and corresponding stagnation

temperatures encountered in flight by long range ballistic vehicles and

satellites . this report describes the hypersonic shock tunnel and

presents some of the results obtained in the driven tube and in the nozzle

helium is ignited in the driver to produce strong shock waves in air .

a shock velocity in air as high as 55,000 fps with a calculated equilibrium temperature of 16,000 k has been produced in the driven tube . the effects of high stagnation temperatures upon the detached shock wave and the pressure distribution for blunt bodies have been observed in the nozzle test section . the detachment distance devreased greatly at high temperatures . the pressure distribution for the hemisphere was found to be less than that predicted by the modified newtonian theory . shock wave boundary layer interaction at the leading edge of a flat plate was observed, and the results agreed with the analytical prediction . a detached shock wave was observed for a blunt two-dimensional body at very low densities in the test section with a flow mach number of 19.6 .

.I 1158

.T

the tailored-interface hypersonic shock tunnel .

.A

charles e. wittliff, merle r. wilson, and abraham hertzberg

.B

cornell aeronautical laboratory, inc.

.W

the tailored-interface hypersonic shock tunnel .

the /tailored-interface/ hypersonic shock tunnel provides a means for producing the high mach number, high stagnation temperature flow conditions encountered in hypersonic flight . various gasdynamic phenomena associated with shock tunnels are discussed, and experimental evidence of the successful application of this technique is presented . as an indication of its research application, the results of heat-transfer experiments on a hemisphere-cylinder model are presented and compared with theory .

.I 1159

.T

experimental investigation of the effect of yaw on

rates of heat transfer to transverse circular

cylinders in a 6500-foot-per-second

hypersonic air stream .

.A

bernard e. cunningham and samuel kraus

.B

.W

experimental investigation of the effect of yaw on

rates of heat transfer to transverse circular

cylinders in a 6500-foot-per-second

hypersonic air stream .

a technique has been developed by which air can be shock-compressed

by helium to 3660 degrees rankine to generate a 6500-foot-per-second

air stream with a flow duration of 40 milliseconds . the resulting

equipment is described . experiments were conducted to determine

rates of heat transfer to transverse circular cylinders of 0.003-,

greater than 100 . the cylinders were tested at a nominal mach number

of 11 with a stagnation reynolds number (evaluated with free-stream

mass flow and stagnation viscosity of 4.00 times 10 to the 4th power

per foot .

.I 1160

.T

recent advances in gaseous detonation .

.A

gross,r.a. and oppenheim,a.k.

.B

ars j. 29, 1959, 173

.W

recent advances in gaseous detonation .

a review of recent work in gaseous detonation is presented . early work is briefly mentioned and treatises listed . theoretical calculations of chapman-jouguet detonations are reviewed, compared and the ambiguity concerning the speed of sound in a reacting gas mixture discussed . experimental chapman-jouguet measurements are reviewed . recent studies of the interior of a detonation wave are presented . standing detonation wave research, detonation limits, two-dimensional detonations, spectra, ionization and magnetohydrodynamic treatments are brought to the reader's attention . a qualitative description of the development of a flame to a detonation is presented . experimental observations are examined and recent theoretical attempts to explain these observations are reviewed .

.I 1161

.T

stagnation point heat transfer measurements in dissociated air .

.A

rose,p.h. and stark,w.i.

.B

j. ae. scs. 26, 1958, 86.

.W

stagnation point heat transfer measurements in dissociated

air .

the results of an experimental investigation of the laminar heat

transfer at the stagnation point of a blunt body in partially

dissociated air are presented and are compared to the theoretical

treatment of fay and riddell . heat-transfer results are

presented for air temperatures as high as 8,000 k. where more than

periments were performed in a shock tube and the new

experimental techniques and principles are discussed briefly .

simulation of flight stagnation conditions at velocities up to satellite

velocity of 26,000 ft. sec. is shown to be possible in shock tubes and

data has been obtained over a large altitude range at these

velocities .

.I 1162

.T

force-test investigation of the stability and control

characteristics of a 1/8 scale model of a tilt-wing

vertical take-off and landing airplane .

.A

tosti,l.p.

.B

nasa tn.d44, 1960.

.W

force-test investigation of the stability and control characteristics of a 1/8 scale model of a tilt-wing vertical take-off and landing airplane .

a force-test investigation has been made to determine the aerodynamic characteristics of a scale model of a tilt-wing vertical-take-off-and-landing airplane in the short- and long-wing configurations . the model had two 6-blade dual-rotating propellers that were not interconnected mounted on a wing that could be tilted up to an incidence angle of about 90 for vertical take-off and landing .

the investigation included measurements of both the longitudinal and lateral stability and control characteristics in the normal- forward-flight, transition, and hovering ranges . tests in the forward-flight and transition conditions were made at various wing incidences and power conditions . tests in the hovering condition were made in the presence of the ground . the data are presented without analysis .

.I 1163

.T

force-test investigation of the stability and control

characteristics of a scale model of a tilt-wing vertical

take-off and landing aircraft .

.A

newson,w.a. and tosti,l.p.

.B

nasa memo 11-3-58l, 1959.

.W

force-test investigation of the stability and control

characteristics of a scale model of a tilt-wing vertical

take-off and landing aircraft .

a wind-tunnel investigation has

been made to determine the

aerodynamic characteristics of a scale

model of a tilt-wing vertical-

take-off-and-landing aircraft . the model

had two 3-blade single-rotation

propellers with hinged (flapping)

blades mounted on the wing, which

could be tilted from an incidence

of 4 for forward flight to 86 for

hovering flight .

the investigation included

measurements of both the longitudinal

and lateral stability and control

characteristics in both the normal

forward flight and the transition

ranges . tests in the forward-flight

condition were made for several values

of thrust coefficient, and tests

in the transition condition were made

at several values of wing incidence

with the power varied to cover a range

of flight conditions from

forward-acceleration (or climb) conditions to

deceleration (or descent) conditions

the control effectiveness of the

all-movable horizontal tail, the ailerons

and the differential propeller pitch

control was also determined . the

data are presented without analysis .

.I 1164

.T

effect of ground proximity on the aerodynamic characteristics

of a four- engined vertical take-off and landing transport

airplane model with tilting wing and propellers .

.A

newson,w.a.

.B

naca tn.4124, 1957.

.W

effect of ground proximity on the aerodynamic characteristics

of a four- engined vertical take-off and landing transport airplane model with tilting wing and propellers .

an investigation has been made

to study the effect of ground

proximity on the aerodynamic characteristics

of a four-engine vertical-

take-off-and-landing transport-airplane

model with tilting wing and

propellers . tests were made with the wing

at an angle of incidence of 90,

the position used for vertical take-off

or landing . with the model at

various heights above the ground, the

lift, drag, and pitching moment

were measured and tuft studies were

made to determine the flow field

caused by the propeller slipstream .

data were obtained for the complete

model, for the model with horizontal

tail removed, and for the

wing-propeller combination alone .

the results of the investigation

showed that, when the model was

hovering near the ground, there was a

strong upwash in the plane of

symmetry and also an increase in lift

of about 10 percent of the

propeller thrust . about one-half of this lift resulted from an increase

in propeller thrust and one-half resulted

from an up load on the

fuselage induced by the upwash . as the model

approached the ground, it also

experienced an increasing nose-down

pitching moment that evidently

resulted from the up load on the fuselage,

the rear part of which was

longer than the front part . the addition

of the horizontal tail which

was located about halfway up the vertical

tail did not increase the

nose-down pitching moment because the

fuselage decreased the energy of

the upwash before it reached the tail .

.I 1165

.T

an investigation of the effect of downwash from a vtol

aircraft and a helicopter in the ground environment .

.A

o'bryan,t.c.

.B

nasa tn.d977, 1961.

.W

an investigation of the effect of downwash from a vtol

aircraft and a helicopter in the ground environment .

dynamic-pressure measurement, in ground effect, have been obtained about a single-rotor helicopter and a dual-propeller vtol aircraft . the results indicate that the slipstream dynamic pressure along the ground, some distance from the center of rotation, is not a function of disk loading but merely a function of the gross weight or thrust of the aircraft . furthermore, for a given gross weight the thickness of this outward flowing sheet of air is less for a small-diameter propeller (higher disk loading propeller) .

the variation of the dynamic-pressure flow field for single and dual propellers or rotors is significantly different in the plane of symmetry between the two rotors than in a direction normal to this plane . the interaction of the two flows produces a region of upflow in this plane where the fuselage is located, and the decay of the maximum dynamic pressure with distance ahead of the fuselage is slower .

.I 1166

.T

an investigation to determine conditions under which

downwash from vtol aircraft will start surface erosion

from various types of terrain .

.A

kuhn,r.e.

.B

nasa tn.d56, 1959.

.W

an investigation to determine conditions under which

downwash from vtol aircraft will start surface erosion

from various types of terrain .

results of an investigation with

small-scale equipment of the

conditions under which the downwash from

a hovering vertical-take-off-

and-landing (vtol) aircraft will start

surface erosion indicate that the

onset of erosion depends only on the

dynamic pressure of the outward

flow of air near the surface . for a

rotor or propeller at a height of

about 1 slipstream diameter above the

surface, this surface dynamic

pressure was found to be equal to the

disk loading . for the vtol

aircraft supported by a ducted fan, the

surface dynamic pressure with the

ducted fan exit at a height of about

one-half the exit-area loading . the

surface dynamic pressure decreases

rapidly with increasing height of the

vtol device .

erosion of sand and loose dirt

started at surface dynamic pressures

of 1 to 3 lb sq ft, which is in general

agreement with helicopter

experience . thoroughly soaking the sand and

loose-dirt surfaces increased the

resistance to erosion to surface dynamic

pressures of 30 to 50 lb sq ft .

spray from water started at surface

dynamic pressures of 1.5 to

pressures up to about 1,000 lb sq ft .

.I 1167

.T

an experimental study of the effect of downwash from

a twin-propeller vtol aircraft on several types of

ground surfaces .

.A

o'bryan,t.c.

.B

nasa tn.d1239, 1962.

.W

an experimental study of the effect of downwash from a twin-propeller vtol aircraft on several types of ground surfaces .

a full-scale, twin-propeller vtol aircraft with a maximum gross weight of 3,400 pounds has been operated on the ground to study the effect of downwash on several types of ground surfaces .

static operation over loose snow indicated a zone of obliterated vision ahead of the pilot in an arc of approximately 10 on each side of the plane of symmetry . an arc 10 to 45 each side of the center line was found to be an area of fair visibility while the arc from 45 to 90 was an area of poor visibility .

static operation in the presence of loose surface material indicated that the downwash cleared the area near the aircraft of these particles without recirculation or damage to any components .

short-time operation at moderate forward speed over loose gravel, with the thrust axis at an angle of

in propeller-blade erosion and numerous

small dents and fabric

punctures in the sides of the fuselage .

the propeller-blade erosion was

superficial except for the leading

edges where several layers of glass

fiber were eroded .

.I 1168

.T

damage incurred on a tilt-wing multipropeller vtol/stol

aircraft operating over a level, gravel-covered surface .

.A

pegg,r.j.

.B

nasa tn.d535, 1960.

.W

damage incurred on a tilt-wing multipropeller vtol/stol

aircraft operating over a level, gravel-covered surface .

a summary is presented of the

damage experienced by a tilt-wing

vtol stol aircraft as a result of

operating from a level surface covered

with loose gravel . the damage was

inadvertently incurred as the aircraft

was performing a taxiing-turn maneuver

over an area of level macadam

surface containing loose and embedded

crushed stone . observers from a chase

aircraft commented that the wing was

tilted at approximately 76 with

respect to the ground when the damage

occurred . deposits of stone in the

open fuselage, punctures in the rotor

blade skin, and damage to the

compressor blades of the engine occurred

due to the circulation of the

crushed gravel .

.I 1169

.T

hangling qualities experience with several vtol research

aircraft .

.A

reeder,j.p.

.B

nasa tn.d735, 1961.

.W

hangling qualities experience with several vtol research

aircraft .

all of the vtol research aircraft

discussed in this paper have

successfully demonstrated conversion from

hovering to airplane flight and

vice versa . however, control about one

or more axes of these aircraft
has been inadequate in hovering flight .
furthermore, ground interference
effects have been severe in some cases
and have accentuated the inadequacy
of control in hovering and very low
speed flight .
stalling of wing surfaces has resulted
in limitations in level-flight
deceleration and in descent, particularly
for the tilt-wing aircraft, which
in this case is a very rudimentary type .
minor modifications to the wing
leading edge have, however, produced
surprisingly large and encouraging
reductions in adverse stall effects .
height control in hovering and in
low-speed flight has proved to
be a problem for the aircraft not having
direct control of the pitch of
the rotors . the other systems have shown
undesirable time lags in
development of a thrust change .

.I 1170

.T

structural loads surveys on two tilt-wing vtol configurations .

.A

ward,j.f.

.B

nasa tn.d729, 1961.

.W

structural loads surveys on two tilt-wing vtol configurations .

the results of two structural-loads

surveys are summarized . the

first loads program discussed concerns

the airframe vibratory loads

encountered during flight tests of the

vz-2 tilt-wing vtol aircraft

throughout the operational range from

hover to cruise flight . the

primary sources of airframe vibration

were wing-stall buffeting and

tail buffeting in descents . the second

loads program discussed

concerns the initial results of a

structural-loads survey conducted as

part of the wind-tunnel test of a

large-scale tilt-wing research model .

this loads program deals with the

steady wing loads measured throughout

simulated transition from hover to

cruise .

.I 1171

.T

the stability under axial compression and lateral pressure

of circular cylindrical shells with a soft elastic

core .

.A

seide,p.

.B

j. ae. scs. 1962.

.W

the stability under axial compression and lateral pressure

of circular cylindrical shells with a soft elastic

core .

the stability under axial compression and lateral pressure of a finite circular-cylindrical shell with an elastic core is treated by means of donnell's equations . the stability criterion is investigated in detail for the general cylinder under axial compression or lateral pressure and for a particular cylinder under combined loading . comparisons are made with available experimental data .

.I 1172

.T

elastic stability of circular cylindrical shells stabilized

by a soft elastic core .

.A

goree,w.s. and nash,w.a.

.B

exp. mech. 2, 1962.

.W

elastic stability of circular cylindrical shells stabilized

by a soft elastic core .

the effect of a soft elastic core upon the

buckling strength of a thin, circular, cylindrical shell is

investigated experimentally . two types of loading are

considered .. (a) axial compression, and (b) uniform

radial-band loading, where the width of the band is small

compared to the length of the shell .

for each type of loading it is shown that the

strengthening effect of the elastic core becomes more significant

with the increasing values of the radius-thickness ratio .

for example, it is shown that for the geometric and

elastic constants considered it is possible, with the presence

of the core, to increase the axial buckling stress by as

much as 65 percent over the values found for those

without an elastic core . the elastic core is even more

effective in stabilizing the shell against buckling due to band

loading, the peak pressure required to buckle the filled

specimen being 7.30 times that required to buckle the

unfilled shell .

.I 1173

.T

the buckling of cylindrical shells under longitudinally varying loads .

.A

v. i. weingarten

.B

space technology laboratories, inc., los angeles, calif.

now with aerospace corp., los angeles, calif.

.W

the buckling of cylindrical shells under longitudinally varying loads .

two problems illustrating the effect of nonuniformity of loading

on the buckling characteristics of circular cylinders are

investigated . the first problem deals with the effect of

linearly varying axial compressive stress, such as would be

produced by the weight of the propellant in a solid-propellant

engine case . the results indicate that the ratio of the maximum

critical compressive stress induced by the shear load to the

critical uniform compressive stress varies from 1.9 for

the curvature parameter z equal to 1.6 as z

becomes infinite . in particular, the increase in stress is

less than 20 per sq. ft. for z greater than 100 .

the stability of thin cylinders loaded by lateral external

pressure varying linearly in the longitudinal direction is also

investigated . the results indicate that for z greater than 100,

the buckling coefficients are proportional to square

root z .

.I 1174

.T

general theory of buckling .

.A

langhaar,h.l.

.B

app. mech. rev. 11, 1958.

.W

general theory of buckling .

various aspects of the theory of buckling are expounded

in many treatises (1 to 15) . the books of bleich (2)

and salmon (3) contain large bibliographies . goodier

developments in buckling theory . numerous references are

appended to hoff's article . nearly all publications on

buckling of shells, available in the u.s.a. (to 1956), are listed in

the bibliographies on shells by nash (18) . the section

ready guide to recent literature .

.I 1175

.T

stresses from local loadings in cylindrical pressure

vessels .

.A

bijlaard,p.p.

.B

asme trans. 77, 1953.

.W

stresses from local loadings in cylindrical pressure

vessels .

a short discussion is given of the possible methods for

computing the stresses caused in cylindrical shells by local

loadings . it is concluded that the method of developing

the loads and displacements into double fourier series

leads to formulas which are best suited for numerical

evaluation . with this method the pertinent expressions

for the displacements caused by radial loads are found by reducing the three partial differential equations of the shell theory to an eighth-order differential equation in the radial displacements, which is similar to, but not identical with, those derived by donnell and yuan . insertion of the fourier series for the radial displacements and the external loading in this equation leads directly to a double series expression of the radial displacement w in terms of the load factors of the radial load . this results in the pertinent expressions for the other displacements and for the bending moments and membrane forces . the cases of radial loading considered here and those which can be reduced to it are (a) a load uniformly distributed within a rectangle, tion, uniformly distributed over a short distance in the circumferential direction, (d) a moment in the circumferential direction, uniformly distributed over a short distance in the longitudinal direction . for all these loadings the load factors, which have to be used in the pertinent formulas for the displacements, bending moments, and membrane forces, are computed . for the case of tangential loading an eighth-order differential equation is derived in terms of the radial displacement and the tangential load . using this equation, formulas for the displacements, bending moments, and membrane forces for tangential loading within a rectangle are found .

.I 1176

.T

bending tests of ring-stiffened circular cylinders .

.A

peterson,j.p.

.B

naca tn.3735, 1956.

.W

bending tests of ring-stiffened circular cylinders .

twenty-five ring-stiffened circular

cylinders were loaded to

failure in bending . the results are

presented in the form of design curves

which are applicable to cylinders with

heavy rings that fail as a result

of local buckling .

.I 1177

.T

effects of rapid heating on strength of airframe components .

.A

pride,r.a.

.B

naca tn.4051, 1957.

.W

effects of rapid heating on strength of airframe components .

results of several experimental

investigations are presented which

indicate the effects of rapid heating

on the bending strength of multiweb

beams and ring-stiffened cylinders .

it is shown that thermal stresses

reduce the bending load carried at

buckling by both beams and cylinders .

the influence of thermal stress on

maximum load is found to depend

largely on the mode of buckling .

for beams that buckle locally, no

apparent effect of thermal stress on

the maximum load has been found .

a reduction in maximum load has been

observed for beams that buckle in

the wrinkling mode and for cylinders .

.I 1178

.T

buckling of ring-stiffened cylinders under a pure bending

moment and a nonuniform temperature distribution .

.A

anderson,m.s. and card,m.f.

.B

nasa tn.d1513, 1962.

.W

buckling of ring-stiffened cylinders under a pure bending

moment and a nonuniform temperature distribution .

thirteen stainless-steel ring-stiffened

cylinders were subjected to a pure

bending load and heated rapidly until

buckling occurred . for most of the

cylinders the heating was not uniform around

the circumference so that appreciable

axial thermal stresses were present .

elementary thermal stress theory was found

to be inadequate for the prediction of

these thermal stresses, but a method was

developed that would give satisfactory

thermal stress results . by properly

accounting for the thermal stress, the

buckling load could be correlated with a

theory for the buckling of an axially

compressed ring-stiffened cylinder that is

uniformly heated .

.I 1179

.T

a theory of asymmetric hypersonic blunt-body flows .

.A

swigart,r.j.

.B

aiaa jnl. 1963, 1034.

.W

a theory of asymmetric hypersonic blunt-body flows .

two-dimensional asymmetric and three-dimensional

inviscid blunt-body flows are analyzed

using a new method . the method is inverse, that

is, the shock-wave shape and freestream

conditions are taken as known, and the body shape and flow field are to be determined .

results at zero angle of attack are obtained as a special

case of the general problem . solutions at

zero angle are calculated for a variety of body shapes

at freestream mach numbers ranging from

infinity to 1.85 . the ratio of specific heats, is taken

as 1.4 . comparison with results obtained

using van dyke's and garabedian's numerical

solutions indicates that the method under

consideration is more accurate than the van dyke

method for determining stand-off distance .

solutions are obtained for parabolic and paraboloidal

shock waves at small angle of attack and

infinite freestream mach number,. assumes the

values 1.4, 1.2, 1.1, and 1.05 . for all cases,

the streamline that wets the body passes through

the shock wave slightly above the point

where the shock is normal and thus does not possess

maximum entropy . these results

provide counter examples to the conjecture that any

isolated convex body in a supersonic stream

is wetted by the streamline of maximum entropy .

.I 1180

.T

approximate analysis of the slot injection of a gas

in laminar flow .

.A

libby,p.a. and schetz,j.a.

.B

aiaa jnl. 1963, 1056.

.W

approximate analysis of the slot injection of a gas in laminar flow .

the laminar diffusion and combustion of a gas injected into a high-speed uniform stream by means of a wall slot are considered . the dorodnitzin-howarth transformation is employed to reduce the boundary layer equations to incompressible form,. the nonsimilar flow field is treated by a modified oseen approximation in conjunction with the integral method . thermal boundary conditions corresponding to an adiabatic wall and to constant wall enthalpy are discussed . the injection of homogeneous, heterogeneous, nonreactive, and reactive gases is treated . for the latter case, the models usually employed for chemical behavior, namely, frozen and equilibrium flow, are considered . the analysis is applicable to a wide variety of laminar flows, e.g., those involving cooling, thermal protection, skin-friction reduction, and supersonic deflagration . a numerical example of practical interest in connection with the venting of gaseous hydrogen boiloff from a rocket booster is presented .

.I 1181

.T

steady magnetohydrodynamic flow past a non-conducting

wedge .

.A

chu,c.k. and lynn,y.m.

.B

aiaa jnl.

.W

steady magnetohydrodynamic flow past a non-conducting

wedge .

this paper presents a study of the steady

two-dimensional magnetohydrodynamic flow of an

infinitely conducting fluid past a nonconducting

wedge with nonaligned flow and magnetic

field . the flows considered are in the /superfast/

or fully hyperbolic regime . the flows consist

of several regions of uniformity connected by

shocks and expansion waves . because of the

boundary condition on the magnetic field, the

magnetic field must be the same in the regions

above and below the wedge,. thus the flows in these

regions are coupled, unlike in the case of

ordinary supersonic gasdynamics . only small wedge

angles and weak waves (characteristics) are

considered . the problem thus is linearized, and explicit solutions are obtained which are

qualitatively similar to the nonlinear solutions .

some interesting and unexpected features

arise, and they are discussed in detail .

.I 1182

.T

an approximate solution for laminar boundary layer flow .

.A

robert l. kosson

.B

grumman aircraft engineering corporation, bethpage, n.y.

.W

an approximate solution for laminar boundary layer flow .

this paper presents an approximate solution for two-dimensional,

incompressible, laminar boundary layer flow with arbitrary pressure

gradient . von mises' form of the boundary layer equation is linearized

by making a change in the coefficient of one of the terms . the

linearized equation yields a solution that is accurate for the outer portion

of the flow but inaccurate near the surface . a separate inner solution

then is developed which is accurate at the surface and which joins with

the outer solution at some point within the boundary layer . the

method may be considered a major modification of one developed earlier

by von karman and millikan, with changes in both outer and inner

solutions, and the point at which the two solutions are joined . the changes

improve the accuracy of the method and in some respects simplify the

calculations . as examples, results are presented for flow with a

linear variation of velocity (including flat plate and stagnation point flow as special cases), flow with sinusoidal variation of

velocity, flow past a circular cylinder (heimenz' velocity

distribution), and flow past an ellipse (schubauer's data) . agreement

with theoretically exact solutions is good, and better than

results obtained using the pohlhausen method .

.I 1183

.T

laminar hypersonic trail in the expansion-conduction

region .

.A

lykoudis,p.s.

.B

aiaa jnl. 1963, 772.

.W

laminar hypersonic trail in the expansion-conduction

region .

the usual procedure in calculating the cooling

process in a wake behind a blunt object is to

assume a region of pure expansion up to a distance

where the pressure has reached its ambient

value, followed by a region where the mechanism

of pure heat conduction is operative . in

the present paper both mechanisms are assumed

to be valid simultaneously, and the result is

compared with previous calculations . the

following criterion is established .. the minimum

radius of a hemisphere-cylinder configuration, above which a simultaneous

conduction-expansion calculation is not needed, is given by

the approximation

where is the nondimensional value

of the enthalpy at the axis of the wake below

which the two methods of computation give the

same result, and m is the flight mach number .

.I 1184

.T

three dimensional effects in viscous wakes .

.A

steiger,m.h. and bloom,m.h.

.B

aiaa jnl. 1963,776

.W

three dimensional effects in viscous wakes .

three-dimensionality in wakelike or jetlike

free mixing may stem from initial geometric

configurations, nonuniformities in flow variables

over a cross section, or boundary conditions

along the flow . these may be generated by

bodies at angle of attack, nonaxisymmetric

bodies, mixing of nonaxisymmetric jets with

an outer flow, finite wings, or more artificial

means . this paper is devoted to studies bearing

on such configurations . the first section

deals with the general mathematical model, in

which the boundary layer approximations
are used, and with methods of solution .
laminar and turbulent flow, compressibility,
unsteadiness, and streamwise pressure gradients
are admitted initially . the flux forms of the
equations are given . algebraic integrals of the
energy equations and the diffusion (
frozen-flow) equations are obtained . a simplification
of the convective terms, roughly
corresponding to the oseen approximation, is used
in the asymptotic downstream region .
the second section contains explicit solutions
for specific configurations, in particular
for flows whose initial isovels are of elliptic shape .
these flows may be wakelike or jetlike .
compressibility is admitted,. however, the flows
must have uniform pressure and must be
steady . the final section deals with interpretation
and evaluation of the results .

.I 1185

.T

an integral method for calculating heat and mass transfer
in laminar boundary layers .

.A

culick,f.e.c.

.B

aiaa jnl. 1963, 783.

.W

an integral method for calculating heat and mass transfer

in laminar boundary layers .

an integral method previously used successfully

in several kinds of boundary layer problems

has been extended to treat simultaneous heat and

mass transfer in a binary-mixture laminar

boundary layer when the pressure is uniform .

the principal results are two pairs of dual

integral relations arising from solutions to the

integral concentration and energy equations .

one pair connects the surface mass transfer rate

and surface concentration of injected gas,. the

other relates surface temperature and heat transfer

rate in the presence of mass transfer .

only the cases of helium and air injection into an

undissociated air stream are discussed in

detail, but the method can be applied to problems

involving other gases . the approximate

results agree quite well with some numerical solutions

and with recent experimental results

for which no numerical solutions are available .

.I 1186

.T

lift of slender delta wings according to newtonian

theory .

.A

messiter,a.f.

.B

aiaa jnl. 1963,795

.W

lift of slender delta wings according to newtonian

theory .

an approximate system of equations is derived to

describe the inviscid flow past a flat slender

wing at angle of attack, in the limit and .

the aspect ratio is required to

approach zero at the same rate as the mach angle in

the flow behind the shock wave . only a

single parameter appears in the resulting equations,

and a similarity law therefore can be

written expressing a correction to the newtonian

normal-force coefficient . for the delta wing,

a correlation of experimental data according to the

similarity law is shown, and the first terms

of the solution are derived under the assumption

that the similarity parameter is small (

vertex angle much smaller than mach angle) .

.I 1187

.T

shock-induced boundary layer separation in overexpanded conical exhaust

nozzles .

.A

arens, m. and spiegler, e

.B

aiaa journal, march 1963. p.578 .

.W

shock-induced boundary layer separation in overexpanded conical exhaust

nozzles .

the flow in overexpanded supersonic nozzles is reviewed . although five essentially different flow regimes can be discerned, depending on the nozzle pressure ratio, the regime of most interest to the engine designer is the one characterized by oblique shock patterns in the nozzle and flow separation from the nozzle wall . it is shown that the pressure rise associated with the separation correlates well with the mach number at the separation point . a simple analytical formulation for the pressure rise required to separate the flow provides excellent agreement with experimental data over a wide range of nozzle operating conditions and allows prediction of overexpanded nozzle performance .

.I 1188

.T

factors affecting lift-drag ratios at mach numbers

from 5 to 20 .

.A

goebel,t.p. martin,j.j. and boyd,j.a.

.B

aiaa jnl. 1963, 640.

.W

factors affecting lift-drag ratios at mach numbers

from 5 to 20 .

yawed-cone working charts and an engineering method are presented and used to calculate

lift-drag ratios of flat-top conical wing-body

arrangements at mach numbers from 5 to 20 .

viscous interaction effects are considered, but

bluntness effects are neglected . correlations

of wind-tunnel data in the range

show that boundary layer

displacement corrections to surface pressure and skin

friction are required to calculate lift-drag ratios

by this method whenever

is greater than 0.2 . is the freestream mach

number and is the freestream reynolds

number based on body length . double- and

single-type shock patterns, transition from one

pattern to the other, and the variation of inner-shock

position with angle of attack are described .

lift-drag ratios are calculated at selected flight

design points for flat-top, conical body

arrangements with triangular and hyperbolic wing

planforms . the hyperbolic wing arrangement

offers a potential l d benefit at mach 5 but

not at mach 10 or above .

.I 1189

.T

nonequilibrium flow past a wedge .

.A

capiaux,r. and washington,m.

.B

aiaa jnl. 1963, 1, 650.

.W

nonequilibrium flow past a wedge .

an exact numerical solution is obtained for

the chemically reacting flow past a wedge . the

freestream is either in equilibrium or out of

equilibrium but nonreacting . the attached

shock wave is shown to be either concave, convex,

or straight, depending on the values of the

amount of dissociation in the freestream and

a parameter describing the amount of energy

contained in the freestream relative to the gas

dissociation energy . numerical examples are

presented illustrating these regimes . the flow

field is characterized by the presence of an

entropy layer and a relaxation layer, both easily

identifiable in the presentation of the numerical

results .

.I 1190

.T

flow of a gas near a solid surface .

.A

ziering,s.

.B

aiaa jnl. 1, 1963, 661.

.W

flow of a gas near a solid surface .

the state of a gas near a solid surface is investigated .

it is assumed that at a sufficiently

large distance from the surface the particle distribution

function is of the chapman-enskog

form . the half-range analysis previously employed for

parallel plate geometrics (symmetric

problems) can be adapted to the single-plate problem .

although the mathematical analysis

differs, the slip coefficients are essentially identical

with those obtained from the parallel

plate problem (e.g., couette flow) . detailed calculations

are presented for both hard sphere

and maxwellian molecules . the recent work of

bakanov and deryagin for hard sphere

molecules, which is based on incorrect approximations, is discussed .

.I 1191

.T

heat transfer to a hemisphere-cylinder at low reynolds

numbers .

.A

hickman,r.s. and giedt,w.h.

.B

aiaa jnl. 1, 1963, 665.

.W

heat transfer to a hemisphere-cylinder at low reynolds

numbers .

measurements of the local heat flux to
hemisphere-cylinder models in a supersonic rarefied
air stream are presented . two different steady-state
methods were developed, and five
individual models were used . data were obtained
throughout the mach number range of 2 to 6,
with reynolds numbers (based on conditions
behind the bow shock and model diameter)
varying from 38 to 1730 . the stagnation point data
indicated a gradual increase from continuum
boundary layer theory at the higher reynolds
numbers to about 10 above at the lower end
of the range investigated . pressure distribution
measurements on cooled and uncooled models
were found to agree well with modified newtonian
theory . local recovery factor
measurements showed a small rarefaction effect at the
lowest reynolds numbers .

.I 1192

.T

an integral method for calculation of supersonic laminar
boundary layer with heat transfer on yawed cone .

.A

yen,shee-mang and thyson,n.a.

.B

aiaa jnl. 1, 1963, 672.

.W

an integral method for calculation of supersonic laminar

boundary layer with heat transfer on yawed cone .

an integral method for calculating the three-dimensional

boundary layer over the surface of

a cone at angle of attack is investigated .

the numerical procedure of integration for that

method on the basis of a simplifying assumption

concerning the boundary layer development

along the cone generator is developed and illustrated

by applying the method to find the

solutions of integral equations for a specific example .

the results obtained for the example for the

range of circumferential angle of 40 investigated

are summarized and given as heat transfer

coefficients, coefficients of friction, and other

friction parameters . the distribution of heat

transfer coefficients checked with available

experimental data fairly well .

.I 1193

.T

some exact solutions for cavitating curvilinear bodies .

.A

ehrich,f.f.

.B

aiaa jnl. 1, 1963, 675.

.W

some exact solutions for cavitating curvilinear bodies .

a special case of cavitating flow solutions is postulated and transformed to a semi-infinite plane . the complete, exact solution then is synthesized by superposition of singularities . the solution is relevant to a general, two-parameter family of curvilinear bodies . the parameters are the flow angles at the two points of flow separation . the body reduces, in the special case, to the rayleigh solution for a flat plate . the equations of the cavity boundaries are given in explicit form . the body form and the stagnation streamline are given as the locus of the roots of a cubic equation . local static pressures and, hence, lift and drag, also may be calculated . the generated solutions constitute a technique involving simple computation for exact solutions of a special family of cavitating curvilinear bodies at finite angles of attack .

.I 1194

.T

magnetohydrodynamic flow past a thin airfoil .

.A

cumberbatch,e., sarason,l. and weitzner,h.

.B

aiaa jnl. 1, 1963, 679.

.W

magnetohydrodynamic flow past a thin airfoil .

the steady flow of a perfectly conducting
magnetohydrodynamic fluid past a thin
nonconducting airfoil is studied with the usual model
in which the fluid variables obey the lundquist
equations linearized about a constant unperturbed
flow . /hyperliptic/ flows, in which
hyperbolic and elliptic fields are superimposed,
are considered . results of grad, mccune and
resler, and sears and resler are extended and
considered in detail for the case of an arbitrarily
inclined unperturbed field . the general solution
contains four line singularities along the
characteristics through the ends of the body and
has two arbitrary constants . by a /
generalized kutta-joukowski condition,/ these constants
are fixed so that two of the line singularities
disappear . specifically, it is required that the
solution be locally square integrable . behavior
of the exponents of the singularities is investigated
by numerical computation and, in limiting
cases, analytically . the singular parts of some flows
are investigated numerically .

.I 1195

.T

experiments with two-dimensional, transversely impinging jets .

.A

dosanjh,d.s. and sheerlan,w.j.

.B

aiaa jnl. 1963, 1, 329.

.W

experiments with two-dimensional, transversely impinging jets .

experiments on the interaction of transversely impinging two-dimensional jet flows were performed in which a low pressure control jet flow interacted with a relatively high pressure power jet flow . the ratio of the control jet to the power jet supply chamber gauge stagnation pressure was adjusted at 0, 10, and 15 . shadowgraphs of the power jet alone, as well as the corresponding interacting jet flows, were recorded to establish the nature of and changes in the shock structure . the jet flows were traversed by a pitot tube to record the pitot pressure distributions at various locations downstream of the power jet exit . it was discovered that with the addition of only a small percent control jet flow, the normal shock front of the highly underexpanded power jet flow changed to an oblique shock structure and, downstream of the previous location of the normal shock which appeared in the power jet flow alone, the maximum recovery stagnation pressures were proportionally much higher . the mechanism for this behavior of the normal shock is proposed .

possible practical importance of this behavior

of interacting jet flows with reference to aerodynamic

noise, supersonic diffuser losses, etc., is

also pointed out . for the power jet flow alone it

was found that by considering the actual jet

boundaries as simply an extension of the actual

nozzle, the average axial flow quantities,

computed from the area-mach number relation using

the observed cross-sectional area of the jet

flow, agreed quite favorably with the experimental results .

.I 1196

.T

growth of the turbulent wake behind a supersonic sphere .

.A

murphy,c.h. and dickinson,e.r.

.B

aiaa jnl. 1, 1963, 339.

.W

growth of the turbulent wake behind a supersonic sphere .

experimental data are presented on the growth

of turbulent wakes up to 8000 calibers

behind and spheres traveling at supersonic

velocities . experimental determination

of the exponential coefficient in the growth law is

very difficult, if not impossible . data are

presented in the form of both .

in the representation, two

regions of different wake growths are observed .

by means of a quasi-steady state assumption,

the effect of drag deceleration is eliminated

and growth of the far wake compared with

theoretical predictions . the agreement with the

lees-hromas theory in this region was found to

be quite good .

.I 1197

.T

unsteady aerodynamic forces on slender supersonic aircraft

with flexible wings and bodies .

.A

yates,j.e. and zeydel,e.f.e.

.B

aiaa jnl. 1, 1963, 350.

.W

unsteady aerodynamic forces on slender supersonic aircraft

with flexible wings and bodies .

  the present paper derives generalized aerodynamic

forces for slender supersonic aircraft on

the basis of slender-body theory . particular

consideration is given to configurations which

are spanwise flexible . to treat configurations

with flexible wings and bodies, the slender

wing-body problem is first reduced to a simple body

problem whose solution is well known, and a

solution of the latter is obtained, utilizing the circle theorem or method of images and a known

solution of the airfoil equation for a double

interval . with this approach, it is not necessary

to apply conformal mapping techniques,

and the solution so obtained is valid for arbitrary

spanwise downwash distributions . on the

basis of slender-body theory, the velocity potential

and, subsequently, the generalized

aerodynamic forces are derived for a general class of

spanwise flexible wing-body configurations .

.I 1198

.T

the blunt-leading-edge problem in hypersonic flow .

.A

oguchi,h.

.B

aiaa jnl. u, 1963, 361.

.W

the blunt-leading-edge problem in hypersonic flow .

the present paper is mainly concerned with

the hypersonic flow over a flat plate with a

blunt nose . the analysis is based on the flow

model in which the flow field behind the shock

wave may be divided into two regions .. the

inviscid-hypersonic-flow region and the entropy

layer, across which the pressure has no appreciable

change . the equations for the entropy

layer can be reduced to those of the usual boundary-layer problem with the exception that the outer edge of the entropy layer, as well as the pressure remain unknown . these unknowns are determined so as to approximately match the entropy-layer solution with the inviscid hypersonic solution in which the shock wave has the shape of the power law of the distance from the leading edge . the assumed flow model is shown to be valid over a restricted range depending on the wall-to-stagnation temperature ratio and (where is the reynolds number based on half the thickness of nose t, m the freestream mach number, and c the chapman-rubesin constant . actual calculations have been carried out for the case with typical values of and the wall-to-stagnation temperature ratio . the calculated values for both the surface pressure and heat-transfer rate are compared with the experimental data . as regards surface pressure in particular, a satisfactory agreement with the data is obtained . the validity of the assumptions upon which the present analysis is based has been examined from the numerical results, and the region of the validity has been found to extend over a certain large range of the nondimensional distance from the

leading edge .

.I 1199

.T

theoretical investigations of a supersonic laminar

boundary layer with foreign-gas injection .

.A

freedman,s.i., radbill,j.r. and kaye,j.

.B

aiaa jnl. 1, 1963.

.W

theoretical investigations of a supersonic laminar

boundary layer with foreign-gas injection .

the phenomena arising from the uniform injection of helium,

air, argon, and iodine into the laminar boundary layer of a

supersonic stream of air in a tube were investigated theoretically .

the partial differential equations describing the energy, mass,

and momentum transfers through the boundary layer were

obtained, and a series solution was found for the case of uniform

injection through the tube wall . the results of the analysis are

in the form of axial distributions of wall temperature and

recovery factor and of radial distribution of concentration,

velocity, static, and stagnation temperatures . the gas mixture

was assumed to be a perfect gas . properties of the mixture were

calculated in accordance with the gibbs-dalton rule and the

mixing rules based on the kinetic theory of dilute gases .

transport properties for pure air were taken from the n.b.s.

tabulations . transport properties for the other gases were calculated
by kinetic-theory methods, employing a lennard-jones 6-12
model for the interaction potential . the theoretical predictions
for the recovery factor along the tube with air or argon injection
agree with experimental data to within one percent . the
theoretical predictions for helium injection indicate an 8-percent
rise in the recovery factor along the tube, while experiments
have shown only a 1-percent rise . these differences between
theory and experiment are attributed to inaccuracies in the
approximations to the transport properties of the binary mixtures .

.I 1200

.T

hypersonic viscous flow over a sweat-cooled flat plate .

.A

tien,c.l. and gee,c.

.B

aiaa jnl. 1963, 159.

.W

hypersonic viscous flow over a sweat-cooled flat plate .

this paper presents a theoretical analysis of the hypersonic
viscous flow over a sweat-cooled flat plate . the physical
system under consideration is the hypersonic laminar boundary
layer over a porous flat plate with homogeneous, normal
injection of a coolant into the external stream . a heat balance at
the porous surface is made between the heat transferred to the
surface and the heat absorbed by the coolant . the existence
of similar solutions requires a nonuniform distribution of coolant

injection . the method of solution consists of the integration of
three simultaneous first-order equations, the momentum and the
energy integral equations in the boundary layer, and the
tangent-wedge approximation in the inviscid layer . first-order
asymptotic formulas are given in both the strong and the weak pressure
interaction regions for the induced surface pressure, the
skin-friction coefficient, and the nusselt number . numerical
results for three specific cases are presented and discussed .

.I 1201

.T

a study of slender shapes of minimum drag using the
newton-busemann pressure coefficient law .

.A

miele,a.a.

.B

aiaa jnl. 1, 1963, 168.

.W

a study of slender shapes of minimum drag using the
newton-busemann pressure coefficient law .

the problem of minimizing the drag of a slender, two-
dimensional or axisymmetric body in hypersonic flow at zero angle of
attack is considered under the assumption that the pressure
coefficient law is newton's impact law as modified by busemann in
order to include centripetal acceleration effects . after the
condition that the pressure coefficient be nonnegative is accounted
for and after arbitrary conditions are imposed on, in addition
to the thickness and the length, the enclosed area and the moment

of inertia of the contour in the two-dimensional case and the wetted area and the volume in the axisymmetric case, the minimal problem is formulated as a problem of the mayer type and solved by the combined use of the euler-lagrange equations, the transversality condition, the erdmann-weierstrass corner condition, and the properties of the switching function .

particular attention is devoted to the class of problems such that, among the four quantities being considered, two are prescribed while the remaining are free . for these problems, the extremal arc is composed of two subarcs .. one is characterized by a positive pressure coefficient and is called the regular shape,. the other is characterized by a zero pressure coefficient and is called the free layer . in this connection, the analysis shows the existence of two different types of solutions depending on whether the thickness is given or free .

if the thickness is given, the expression for the regular shape is a power law, and the transition from the regular shape to the free layer occurs in the second half of the body . in the two-dimensional case, the exponent of the power law is 1 if the length is given if the enclosed area is given, and 3 if the moment of inertia of the contour is given,. the transition point from the power body to the free layer is located at 50 percent of the length if the length is given, at 66 percent if the enclosed area is given, and at the axisymmetric case, the exponent of the power law is if the length is given, 1 if the wetted area is given, and if the volume is given,. the transition point from the power body to the free layer is located at 60 percent of the length if the length is given, at 70 percent if the wetted area is given, and at 80 percent if the volume is given .

on the other hand, for problems where the thickness is free, the equation governing the regular shape is not that of a power body, and the point of transition to the free layer is located in the first half of the body . in the two-dimensional case, the transition point is at 28 percent of the length if the length and the enclosed area are given, at 32 percent if the length and the moment of inertia of the contour are given, and at 45 percent if the enclosed area and the moment of inertia of the contour are given . in the axisymmetric case, the transition point is located at 35 percent of the length if the length and the wetted area are given, at 39 percent if the length and the volume are given, and at 46 percent if the wetted area and the volume are given .

for all of the cases considered, analytical expressions are obtained for the optimum shapes, the thickness ratios, and the drag coefficients .

.I 1202

.T

uniformly valid second-order solution for supersonic flow over cruciform surfaces .

.A

wallace,j. and clarke,j.h.

.B

aiaa jnl. 1, 1963, 179.

.W

uniformly valid second-order solution for supersonic flow over cruciform surfaces .

considered is the second-order supersonic flow over a cruciform configuration consisting of two intersecting rectangular wings of high aspect ratio . the practical interest is in application to supersonic inlets, wing-body junctions and vehicle fins . the fundamental interest centers about identification and adjustment of the severe local failures of the ordinary second-order theory . for wings with discontinuous slopes, discontinuous potentials occur across the planar shock and square-root singularities in the velocities occur at the intersection of these shocks with the cruciform surfaces . the problem is simple enough so that these interesting features stand out clearly .

a second-order solution uniformly valid to first order is constructed by adjustment of the ordinary second-order solution obtained first . the uniformly valid solution has two different series representations in the thickness parameter . one is the ordinary second-order series in ascending integral powers of the thickness parameter which is valid in the interior of the vertex-centered undisturbed mach cone, and the other is a series containing fractional powers which is valid adjacent to and upstream of this mach cone . the uniformly valid solution gives the detailed wave structure and shows a flow regime upstream of the vertex-centered undisturbed mach cone not predicted by the ordinary theory . the two solutions are otherwise identical .

the wave structure consists of a pyramidal arrangement of planar shocks adjacent to and upstream of the above cone, followed by weaker oblique expansion fans and finally by two extremely weak shocks coincident with the vertex-centered undisturbed

mach cone . as an example of the above, detailed results are

presented for the case of two intersecting wedges . application

of the techniques to other quasi-cylindrical problems is discussed .

.I 1203

.T

the propagation of a nonuniform magnetohydrodynamic

shock wave into a moving monatomic fluid .

.A

gundersen,r.m.

.B

j. ae. scs. 1962, 1421.

.W

the propagation of a nonuniform magnetohydrodynamic

shock wave into a moving monatomic fluid .

an initially uniform magnetohydrodynamic shock wave of

arbitrary strength propagates through a channel which consists of

two portions of which one has uniform cross-sectional area while

the other is of varying cross-sectional area . it is assumed that

the flow in the nonuniform section in front of the shock is initially

a uniform state and no perturbations (due to the area variations)

of this flow reach the shock until the area variation is encountered .

when the shock enters the nonuniform section, it is perturbed,

the shock strength altered and the subsequent flow is

nonisentropic . in addition to the perturbation due to the effect of the

area variations on the initially uniform upstream flow, there are

two further contributions--viz., a permanent perturbation

caused directly by the area changes and a transient
disturbance--which propagates with true sonic speed with respect to the flow
behind the shock, due to reflections of the permanent
perturbation at the shock . expressions for these various contributions are
obtained . the results presented include as special cases
propagation of a nonuniform conventional gas dynamic shock into a
moving nonconduction fluid and propagation of a nonuniform
hydromagnetic shock wave into a stationary fluid .

.I 1204

.T

experimental effect of bluntness and gas rarefaction
on drag coefficients and stagnation heat transfer on
axisymmetric shapes in hypersonic flow .

.A

bloxson,d.e. and rhodes,b.v.

.B

j. ae. scs.1962.

.W

experimental effect of bluntness and gas rarefaction
on drag coefficients and stagnation heat transfer on
axisymmetric shapes in hypersonic flow .

inverted hemispheres, circular discs (normal to stream),
spheres, 26 total angle 0.368 blunt hemisphere cones, 18
total-angle sharp cones, and other axisymmetric shapes were run in a
hypervelocity wind tunnel . hypersonic drag coefficients at zero
angle of attack were measured in the air velocity range, 7,000-
efficient is defined as drag force . knudsen number is

defined as mean free path behind shock sphere shock detachment distance . in the case of nonsphere shapes, the knudsen number is defined as the knudsen number of a sphere with the same base diameter .

these drag coefficients cover the range of gasdynamics to free molecule flow and are given in graphical form . the drag coefficients were measured by means of a ballistic balance in millisecond intervals, and referenced to the drag coefficient of a sphere in the gasdynamics region, for a gamma of 1.4, of 0.92 .

tunnel stagnation conditions of pressure, temperature, density, and pressure drop with time were measured directly . in the tunnel test section, velocity, q density, total pressure, and static pressure were measured directly .

these experimental curves have been found useful in the analysis of complex shapes if the complex shapes can be easily broken down into simple components with small interactions between components .

heat-transfer distributions have also been obtained on these and other complex shapes in the hypervelocity wind tunnel, by means of a special paint which changes through several visible spectral orders within a heat transfer range of x10 for a single application . heat transfer rates, so obtained, have been performed in the hypersonic gasdynamic and slip flow regions and are presented for spheres . these data, in the vorticity interaction region, agree with the data of ferri and zakkay .

.I 1205

.T

effects of cooling on boundary layer transition on

a hemi- sphere in simulated hypersonic flow .

.A

dunlap,r. and kuethe,a.m.

.B

j. ae. scs. 1962, 1454.

.W

effects of cooling on boundary layer transition on

a hemi- sphere in simulated hypersonic flow .

an experimental investigation of the effects of cooling on boundary-layer transition on a 9-in. diameter hemisphere in simulated hypersonic flow is reported . the newtonian pressure distribution was obtained by use of a shroud and boundary layer cooling was achieved by internally cooling the model . transition was detected with hot wires and with a pitot tube at the surface . attained .

transition was observed in the subsonic and near-sonic flow region at and upstream of n = 45 . in this region the stagnation reynolds number at which transition occurred when the surface was highly polished was only slightly affected by cooling within the temperature range . thus, transition reversal does not occur on a polished spherical surface within the range of these tests, and we therefore conclude that the cooling did not cause the linear stability of boundary layer to decrease significantly .

an essential feature of transition studies with boundary-layer cooling is the close control of surface roughness . in the present experiments this control required, in addition to a highly polished

surface, the necessity for low water vapor dewpoint, the

avoidance of carbon dioxide condensation and the utilzation of every

available means for removing the dust from the airstream .

.I 1206

.T

magnetohydrodynamic mach cones .

.A

cumberbatch,e.

.B

j. ae. scs. 1962, 1476.

.W

magnetohydrodynamic mach cones .

features of the surfaces of main disturbance created by a small

object in steady motion through a conducting fluid are examined .

these surfaces are found by drawing tangent cones from the

object to the relevant wave-front diagrams . the outer wave

cone (when present) is smooth, but the two inner cones have

cross sections similar to the cusped figures of the inner

wave-front diagram . it is conjectured that the disturbance may be

concentrated along such line cusps . this has particular

relevance in the application of known two-dimensional results to

three-dimensional problems, say in the well-known techniques

of aerodynamics . in mhd the omission of the large disturbance

characteristics implicit in a two-dimensional solution may

invalidate its use in any practical three-dimensional problem .

.I 1207

.T

supersonic airfoil performance with small heat addition .

.A

mager,a.

.B

j.ae.scs. 1959, 99.

.W

supersonic airfoil performance with small heat addition .

an analytical method is presented which permits a very rapid evaluation of the acrodynamic effects arising from the addition of small amounts of heat near supersonic two-dimensional airfoils . this method applies to shockless inviscid flow without heat conduction . also, the mechanism by which the sesired heat addition is achieved is not considered .

it is shown that even small amounts of heat generate a substantial pressure rise and thus cause appreciable changes in the acrodynamic coefficients . the results of this analysis compare favorably with those obtained by a more accurate, but also more tedious, graphical method of characteristics .

two possible modes of application to an airplane design are considered from the energy requirements standpoint . in this connection, it is shown that the decrease of the required wing area resulting from heat addition may, in some cases, lead to savings in the rate of the fuel consumption . in general, however, one should not expect any substantial reduction in energy requirements resulting from the application of the wing heat addition .

.I 1208

.T

a linearized analysis of the forces exerted on a rigid

wing by a shock wave .

.A

ehlers,f.e. and shoemaker,e.m.

.B

j. ae. scs. 1959, 75.

.W

a linearized analysis of the forces exerted on a rigid

wing by a shock wave .

solutions are obtained in closed form for the pressures exerted

on a rigid half plane by an incident, plane acoustic shock wave .

the angle of incidence of the wave front is arbitrary and the half

plane is considered to be traveling at constant velocity, subsonic

of supersonic with respect to the acoustic medium . a

closed-form solution is obtained also for a rigid wedge which is

motionless with respect to the acoustic medium . the analysis is carried

out by transforming the wave equation to laplace's equation

by the busemann conical transformation and then applying

conformal mapping .

.I 1209

.T

aerodynamic processes in the downwash-impingement problem .

.A

vidal,r.j.

.B

j. ae. scs. 1962.

.W

aerodynamic processes in the downwash-impingement problem .

theoretical and experimental data relating to the downwash impingement problem are examined in order to arrive at a coherent understanding of the process of entrainment of ground particles in the flow . it is demonstrated that a key mechanism in the process is the interaction of nonuniform flow in the ground boundary layer with bluff ground particles . this interaction produces a lift force which, under typical conditions, equals or exceeds the particle weight .

in the interest of quantitative prediction of the conditions necessary for particle entrainment, four subsidiary problem areas in the impinging jet are examined . these are the viscous decay, the inviscid flow field, the ground boundary layer, and the forces on a bluff body in nonuniform flow . applicable theories are used in conjunction with experimental data to assess the accuracy and range of validity of the theories, and to define the stream conditions which will cause particle entrainment .

available data are applied to the establishment of criteria for particle entrainment in the vicinity of the impinging-jet stagnation point . these criteria show that entrainment occurs in a finite annular region on the ground plane, and that the particles most readily entrained are those with a diameter equal to about two-thirds the thickness of the ground boundary layer . the configuration size is shown to influence the process in that the onset of entrainment is fixed by the jet diameter and velocity, and the size of the ground particles . the criteria established

provide a quantitative estimate of the conditions causing

entrainment and provide a basis for scaling experimental results to

a variety of full-scale situations .

.I 1210

.T

on slender airfoil theory for nonequilibrium flow .

.A

ryhming,i.l.

.B

j.ae.scs. 1962, 1076.

.W

on slender airfoil theory for nonequilibrium flow .

an exact linear theory for nonequilibrium flow past a thin

airfoil is given . green's function technique is used to solve the

boundary value problem for the governing third order equation .

upon satisfying the boundary condition on the airfoil surface an

integral equation is obtained which has an exact solution . the

final expression for the velocity potential, given as an integral over

the source strength times the green's function, shows that the

solution is dependent not only on the slope variation of the airfoil

but also on its curvature variation . this turns out to be the case

for all free-stream mach numbers .

as an example, the supersonic flow past a wedge is considered .

.I 1211

.T

boundary layer transition at supersonic

speeds-three-dimensional roughness effects (spheres).

.A

van driest,e.r. and blumer,c.b.

.B

.W

boundary layer transition at supersonic

speeds-three-dimensional roughness effects (spheres).

further experiments carried out in the 12-in. supersonic wind

tunnel of the jet propulsion laboratory of the california

institute of technology to investigate the effect of three-dimensional

roughness elements (spheres) on boundary-layer transition on a

the local mach number for these tests was 2.71 . the data show

clearly that the minimum (effective) size of trip required to bring

transition to its lowest reynolds number varies as the one-fourth

power of the distance from the apex of the cone to the trip . use

of available data for other mach numbers indicates that the

mach-number influence for effective tripping is taken into

account by the simple expression .

some remarks concerning the roughness

variation for transition on a blunt body are made .

.I 1212

.T

effect of uniformly distributed roughness on turbulent

skin-friction drag at supersonic speeds .

.A

goddard,f.e.

.B

j. ae. scs. 1959, 1.

.W

effect of uniformly distributed roughness on turbulent skin-friction drag at supersonic speeds .

an experimental program was carried out in the 18-in. by 20-in. supersonic wind tunnel of the jet propulsion laboratory to determine the effect of uniformly distributed sand-grain roughness on the skin-friction drag of a body of revolution for the case of a turbulent boundary layer . the mach number range covered was 1.98 to 4.54, and the reynolds number varied from about 3 x 10 to 8 x 10 . some data were also obtained at a mach number of 0.70 .

at speeds up to a mach number of 5 and for roughness sizes such that the quadratic resistance law holds, the compressibility effect is indirect, and the skin-friction drag is a function of only the roughness reynolds number, exactly as in the incompressible case . it is shown that the entire compressibility effect is a reduction of the fluid density at the surface as the mach number increases . the critical roughness, below which the surface is hydraulically smooth, is,. this is equal to the thickness of the laminar sublayer for a smooth surface for both compressible and incompressible flow . over the range of roughness sizes considered here, there appears to be no wave drag associated with the drag due to roughness . the shift in the turbulent veocity profile for a rough surface at supersonic speeds is a function of only the roughness reynolds number and quantitatively follows exactly the same law as that for the incompressible case .

.I 1213

.T

heat transfer to slender cones in hypersonic flow,

including effects of yaw and nose bluntness .

.A

wittcliff,c.e. and wilson,m.r.

.B

j. ae. scs. 1962, 761.

.W

heat transfer to slender cones in hypersonic flow,

including effects of yaw and nose bluntness .

as part of a general study of the aerothermodynamic characteristics of flight of hypersonic vehicles, an investigation of laminar heat transfer to slender yawed cones has been conducted . experiments have been made in the cal 11- by 15-in. shock tunnel at mach numbers from 11 to 13 and at yaw angles up to were tested .

the heat-transfer rates are compared with theoretical predictions . the effects on the local heat-transfer rates of the boundary-layer displacement thickness, transverse curvature, yaw, nose bluntness, and the entropy sublayer are discussed . it is shown that, at zero yaw, the experimental data for the sharp cone are in good agreement with theory when boundary-layer displacement and transverse-curvature effects are included . for the yawed sharp cone, the heat-transfer rates along the most windward streamline are in good agreement with reshotko's theory for yaw angles up to 3 . at larger yaw angles, the

experimental heat transfer was found to be greater than that

predicted theoretically . however, at these yaw angles the

heat-transfer distribution on the windward side was in good

agreement with laminar-boundary-layer calculations based on an

assumption of local similarity . the zero-yaw tests of the blunted

cones showed qualitative agreement with cheng's shock-layer

theory for slender blunt-nose bodies .

.I 1214

.T

the drag of elongated bodies over a wide reynolds number

range .

.A

robertson,j.m. and clark,m.e.

.B

j. ae. scs. 1962, 842.

.W

the drag of elongated bodies over a wide reynolds number

range .

the resistance of bodies in motion through an incompressible

viscous fluid is predictable from stokes- or oseen-type solutions

in the creeping-motion range, while some test information is

available in the boundary-layer range . with the exception of

experimental results for spheres or circular cylinders and

analytical and experimental results for flat plates, almost no information

is available on other bodies, particularly in the intermediate

range of reynolds numbers extending from unity to a million .

experimental results as obtained from hydroballistic studies in water and glycerin-water solutions are presented for finned ellipsoids of fineness-ratio .4 over a 20,000-fold range and are correlated with available information on other bodies . although results do not extend down to the creeping-motion region where analytical predictions are available, comparison with the drag coefficient trends for spheres and flat plates indicates that an appropriate curve for the ellipsoid could be extended so as to cover the entire laminar-viscous range . less extensive results are presented on the drag of fineness-ratio 8 ellipsoids and on laminar-turbulent transition occurrences .

.I 1215

.T

the effect of slip particularly for highly cooled walls .

.A

rott,n. and lenard,m.

.B

j. ae. scs. 1962, 591.

.W

the effect of slip particularly for highly cooled walls .

it is found that for boundary conditions on the velocity slip and on the temperature jump which are not oversimplified in an unrealistic way, the effect of these phenomena on the heat transfer and the shear at a stagnation point is of the order of the ratio of the mean free path outside the boundary layer to the boundary-layer thickness, even for highly cooled walls . a simplified theory of this effect is given, which puts the physical reasons for the results in evidence and agrees closely with the more accurate calculations . it is concluded that the effects of slip and jump are not negligible in comparison with other low-reynolds-number corrections, even for very cold walls .

.I 1216

.T

pressure distribution in regions of step-induced turbulent separation .

.A

vasilu,j.

.B

j. ae. scs. 1962, 596.

.W

pressure distribution in regions of step-induced turbulent separation .

an analysis is made of the pressure distribution in the separated-flow region ahead of a step, using the concept of the turbulent mixing coefficient of crocco and lees and the /jet-flow/ model of chapman with some modification . on the basis of a variable mixing coefficient, a differential equation for the pressure distribution is derived, which gives the pressure rise as a function of the distance from the separation point . this equation contains the separation length as an unknown . a second equation is obtained by making a mass balance of the air entering and, leaving the /dead-air/ region ahead of the step . the pressure rise and the separation distance for a given mach number are determined by solving the two equations simultaneously .

the analysis yields results which are in close agreement with the experimental data on steps, obtained at princeton, particularly for m = 3.85 . for lower mach numbers, a maximum variation of 5 percent is found between theory and experiment . use of the velocity profiles of jets, as required by the jet-flow model, necessarily restricts the applicability of the present study to flows with thin boundary layers at the separation point .

.I 1217

.T

application of inequality constraints to variational problems of lifting re-entry .

.A

levinsky,e.s.

.B

j. ae. scs. 1962, 400.

.W

application of inequality constraints to variational problems of lifting re-entry .

inequality constraints are introduced into the variational formulation of the optimum re-entry problem for a lifting vehicle to prevent human and or structural tolerances from being exceeded . these constraints consist of minimum and maximum angle of attack, maximum load factor, and maximum convective heat transfer (equilibrium temperature) .

the equations have been programed for the ibm 704 computer, and sample trajectories are presented for which the total heat transferred to certain critical areas on the windward surface of the vehicle is minimized . these trajectories indicate the

dominant effect of the constraints on the optimum flight path,

which is shown to consist of both unconstrained and constrained

arcs .

.I 1218

.T

experimental lift and drag of a series of glide configurations

at mach numbers 12 .6 and 17 .5.

.A

geiger,r.e.

.B

j. ae. scs. 1962, 410.

.W

experimental lift and drag of a series of glide configurations

at mach numbers 12 .6 and 17 .5.

a series of semiballistic-type bodies consisting of three half

sphere cones of 0.3 bluntness ratio with half-cone angles of 8.6,

laboratory hypersonic shock tunnel at m = 17.5 and 12.4 .

in addition, a representative winged glide configuration

consisting of a sharp-edged, 60 swept delta wing with cone-segment

the range of angle of attack for the half sphere-cone tests was

the technique for force coefficient determination consists of

analyzing high-speed motion pictures of the motion of very light

balsa and isofoam plastic models which are literally free-flown

for several milliseconds in the test section of the shock tunnel .

because of viscous effects the newtonian prediction of half

sphere cone drag is consistently less than, but generally parallel these bodies is generally well predicted by the newtonian theory

except at small and moderate positive angles of attack where it is

generally less than newtonian . this lift deficiency appears to

increase with cone half angle . maximum lift-drag ratios fall

considerably short of the newtonian predictions . several

exploratory tests at mach 11.7 and low reynolds number (

approximately reduction in) on the 13 model produced an

approximate doubling of minimum drag and a 35 percent

decrease in (l d) max,. this demonstrates the importance of viscous

effects for blunt bodies in the reynolds number range of these

tests .

the sharp leading-edge, 60 sweep delta wing-body

configuration exhibited the same (l d) max, as the wing alone, about 2.80

at both positive and negative angles of attack .

.I 1219

.T

determination of lift or drag programs to minimize

re-entry heating .

.A

bryson,a.e.

.B

j. ae. scs. 1962, 420.

.W

determination of lift or drag programs to minimize

re-entry heating .

a study of single-pass re-entry from escape speed and from

circular satellite speed is made to determine the lift program for a

hypersonic glider and the drag-modulation program for a
non-lifting vehicle that minimize the heating of the vehicles within
acceleration or range constraints . a new method of numerical
solution is used, similar to kelley's /method of gradients,/ that
permits rapid convergence to the optimum lift program starting
with an original good estimate . this method avoids the
two-point boundary-value problem of the calculus-of-variations
formulation, and is applicable to any optimum-programing
problem . an acceleration-tolerance limit is introduced which
describes the human pilot's capability to withstand acceleration
more accurately than a simple acceleration limit .

.I 1220

.T

boundary layer transition in the presence of streamwise
vortices .

.A

tani,i. and komoda,h.

.B

j. ae. scs.1962, 440.

.W

boundary layer transition in the presence of streamwise
vortices .

results of an experimental investigation of instability leading
to transition in the subsonic boundary layer flow along a flat
plate are presented . a series of wings was placed outside the
boundary layer to produce streamwise vortices, which in turn
made the boundary layer three-dimensional--i.e., periodic in

thickness in the spanwise direction . hot-wire measurements were made to trace the downstream development of the disturbance or wave created by the vibrating ribbon . as the wave travels downstream, it is deformed into a three-dimensional configuration by the three-dimensionality of the boundary-layer flow, but it is eventually damped out so long as it remains small in intensity . it is only after the wave intensity exceeds a certain amount (which depends on the degree of boundary-layer three-dimensionality) that the nonlinear effect manifests itself by the rapid amplification of wave intensity, the rapid increase in wave three-dimensionality, and the distortion in mean velocity profile . the appearance of nonlinear development inevitably leads to the breakdown of laminar flow, and hence the onset of turbulence . there is present a mechanism by which the energy is transferred from one spanwise position to another so that the breakdown of laminar flow occurs as a consequence of three-dimensional development of the wave as a whole .

.I 1221

.T

steady flow of conducting fluids in channels under transverse magnetic fields, with consideration of hall effect .

.A

tani,i.

.B

j. ae. scs.1962, 297.

.W

steady flow of conducting fluids in channels under
transverse magnetic fields, with consideration of hall
effect .

an approximate method of solution based on a minimum
principle is presented for the steady laminar incompressible flow of an
electrically conducting fluid through a straight channel of
arbitrary cross section with conducting or nonconducting walls in the
presence of a uniform transverse magnetic field . the hall effect
is taken into account by making simplifying assumptions that the
gas is fully ionized and that both reynolds number and magnetic
reynolds number are small . numerical calculations are carried
out for the case of a rectangular channel .

.I 1222

.T

axisymmetric magnetohydrodynamic channel flow .

.A

hains,f.d. and holer,y.a.

.B

j. ae. scs. 1962, 143.

.W

axisymmetric magnetohydrodynamic channel flow .

the axisymmetric subsonic and supersonic flow fields, and the
skin friction and heat transfer of an electrically conducting
compressible fluid flowing in a channel of constant circular area
through a magnetic field are investigated when the magnetic
reynolds number is small . the inviscid-flow field for flow
through a dipole field is solved by the method of characteristics

in the supersonic case . for the subsonic case, linearized equations are derived for small values of the magnetic interaction parameter . numerical results are obtained by the relaxation method .

the inviscid-flow-field solutions are used as boundary conditions for the laminar boundary layer along the wall, in which axial pressure gradients form an important feature . the exact continuum-flow equations are reduced by an order-of-magnitude analysis to the boundary-layer equations, which are solved numerically by an integral method using a fourth-degree velocity profile and a fifth-degree stagnation-enthalpy profile .

pressure, temperature, and heat-transfer measurements are made with a shock tube under supersonic-flow conditions closely approaching those used in the numerical computations . general agreement is found between the theoretical and the experimental results .

.I 1223

.T

inviscid-incompressible-flow theory of static two-dimensional solid jets, in proximity to the ground .

.A

strand,t.

.B

j. ae. scs. 1962, 170.

.W

inviscid-incompressible-flow theory of static two-dimensional solid jets, in proximity to the ground .

the inviscid-incompressible-flow theory of static two-dimensional solid jets impinging orthogonally on the ground is presented using conformal mapping methods .
  it is shown that the thrust of a solid jet at constant power initially decreases as the ground is approached . the magnitude of the thrust out of ground effect is regained only at a very low height-to-jet width ratio (approximately 0.55) . the maximuin decrease is about 6 percent . the ground effect on solid jets is thus largely unfavorable .

.I 1224

.T

on the plk method and the supersonic blunt-body problem .

.A

vaglio-laurin,r.

.B

j. ae. scs. 1962, 185.

.W

on the plk method and the supersonic blunt-body problem .
  detailed analysis of the subsonic and transonic portious of the flow field about either very blunt or asymmetric configurations requires successive approximations,. these can be carried out in a systematic fashion only when an appropriate convergent perturbation procedure is available . the problem of producing successively refined sets of initial conditions for either /direct/ or /inverse/ analysis of the flow is formulated in the following terms .. given reasonable estimates for shock shape and pressure distribution on the body, can one determine the flow field

of interest to any desired degree of approximation by a

perturbation approach .qm

a procedure to this effect is developed which involves

stretching of coordinates in the spirit of the poincare-lighthill-kuo

are transformed along body, shock, and intermediate lines so as

to annul perturbations of the local resultant velocity,. (b) for the

integral method the coordinate along the boundary of each strip

is shifted so as to control perturbations of the velocity component

that determines the critical point .

the approach is justified by a study of the equations governing

the direct method, and by consideration of model transonic flow

problems for which closed form solutions are available . the

range of validity of the proposed procedure is assessed by

practical application and comparison with experiment,. results are

presented for a disk set normal to a low-termperature air stream

at m = 4.76, and for a highly asymmetric two-dimensional

configuration at m = 8 .

.I 1225

.T

the effect of adverse pressure gradients on the characteristics of

turbulent boundary layers in supersonic streams .

.A

george h. mclafferty and robert e. barber

.B

united aircraft corporation research laboratories

.W

the effect of adverse pressure gradients on the characteristics of

turbulent boundary layers in supersonic streams .
tests were conducted at mach numbers from 2.0 to 3.5 to determine the thickness and profile shape characteristics of turbulent boundary layers on two-dimensional and axisymmetric curved-surface models having adverse pressure gradients . the magnitude of the gradients relative to the boundary-layer thickness at the beginning of the gradient was varied by employing models having different radii of curvature and by changing the boundary-layer thickness at the beginning of the gradient . the overall pressure rise in most cases was greater than the value which would cause a turbulent boundary layer to separate if the pressure rise were created by an oblique shock wave .
an analytical investigation was also conducted so that the results of the experimental investigation could be applied to the prediction of cases outside the range of the experiments .
it is shown that boundary-layer momentum thickness can be predicted from the von karman boundary-layer momentum equation, but that measured values of boundary-layer profile shape are in poor agreement with values computed from procedures derived by extending conventional methods for predicting profile shape in subsonic flow . a new procedure for calculating boundary-layer profile shapes, developed in this paper, is shown to provide a good correlation between experimental and calculated values of boundary-layer profile shapes in adverse pressure gradients created by curved surfaces . this procedure is based on the experimental observation that the station at which high-energy free-stream flow actually mixes into a turbulent boundary layer in an adverse pressure gradient is well downstream of the station at which flow would have to mix in order to maintain a flat-plate profile .

.I 1226

.T

heat transfer in the laminar boundary layer with ablation

of vapor of arbitrary molecular weight .

.A

faulders,c.r.

.B

j. ae. scs. 1962, 76.

.W

heat transfer in the laminar boundary layer with ablation

of vapor of arbitrary molecular weight .

the reduction of heat transfer in the laminar boundary layer under the condition of vaporizing ablation is analyzed for arbitrary molecular weight of the vapor . primary assumptions are that the pressure gradient is zero, the individual components of the binary system are perfect gases, the prandtl number is constant, and the viscosity is proportional to temperature . variations through the boundary layer of the schmidt number for binary diffusion and the density-viscosity product, are included in the analysis . the wall temperature is held constant . numerical results are obtained for prandtl numbers of 0.75 and varying from 0.25 to 4.00, wall concentration of the foreign gas as high as 0.9 (corresponding to the high heat rates encountered during re-entry), and ratio of specific heats of foreign gas equal to that of air . kinetic theory is used to obtain schmidt number as a function of molecular weight and concentration .

the departure of schmidt number and prandtl number from

unity and the variation of reynolds analogy factor with prandtl
number, blowing parameter, wall concentration, and molecular
weight ratio are found to have relatively minor influence on the
heat block ratio at high rates of ablation . the primary factor
governing the influence of molecular weight ratio on the heat
block ratio is the variation of across the boundary layer .
little loss of accuracy is incurred, in the range of molecular weight
ratios considered here, by assuming schmidt and prandtl
numbers of unity as long as the variation is properly taken into account .
.I 1227
.T
pressure-gradient effects on the preston tube in supersonic
flow .
.A
naleid,j.f. and thompson,m.j.
.B
j. ae. scs. 1961, 940.
.W
pressure-gradient effects on the preston tube in supersonic
flow .
this paper is concerned with an experimental investigation of
the effects of a longitudinal pressure gradient in a supersonic
stream of air over a bounding surface on the performance of a
preston or impact-pressure tube at the surface . evidence is
presented which indicates that for the mach number considered and
for the range of pressure gradients covered, the preston tube
functions in a completely satisfactory manner for the determination

of local shear stress .

.I 1228

.T

leading-edge separation of laminar boundary layers

in supersonic flow .

.A

brower,w.b.

.B

j. ae. scs. 1961, 957.

.W

leading-edge separation of laminar boundary layers

in supersonic flow .

a brief description of the flow field is given for the interaction

of shock wave and laminar boundary layer on a compression

corner in supersonic flow . a special sub-case--that of

leading-edge laminar separation--is analyzed by extension of chapman's

laminar mixing-layer theory . results are tabulated for ranges

of mach number, and compression-corner angle .

a limited region of possible leading-edge laminar

separation with an attached leading-edge shock (or in certain

cases an expansion) followed by a second shock due to the

reattachment flow is found to exist . comparison with existing

experimental data is found to be satisfactory in several cases .

.I 1229

.T

the effect of sweep angle on hypersonic flow over blunt

wings .

.A

bogdonoff,s.m. and vas,r.e.

.B

j. ae. scs. 1961.

.W

the effect of sweep angle on hypersonic flow over blunt

wings .

a series of tests were carried out in the princeton university

helium hypersoule wind tunnel on blunt two-dimensional wings

at zero angle of attack with sweep angles up to 70 at mach

numbers from 7 to 15 . the leading edge reynolds number varied

from 3,000 to 25,000 . the measured pressure distributions were

compared with the simple summation of the theoretical inviscid

effect (based on blast wave theory using the normal mach

number) added to the viscous effect (calculated as if no sweep

were present) . for the unswept wing, the slope of the pressure

decay was reasonably well predicted by the theoretical

calculations . the viscous theory reasonably predicted the variation in

the pressure distribution due to changes in leading-edge

reynolds number . by subtracting the theoretical viscous effects, an

inviscid mach number dependence of the 2.2 power was found

as compared to the value of 2.0 predicted by the inviscid theory .

the same approach for the swept wing did not give consistently

satisfactory results . deviations avove and below the calculated

value by as much as 40-50 percent were measured and there

seemed to be no systematic variation with either mach number or

reynolds number . at a constant high reynolds number, it was found that the pressure distribution varied with the distance

along the wing with an exponent between about--0.53 and--0.58

except for a rather sharp decrease which occurred for the 70

sweep case . the pressure at a given station for a fixed mach

number and given leading edge thickness varied as the cosine of

the sweep angle to the 1.1 power as compared to the 1.3 power

predicted from general geometrical considerations .

.I 1230

.T

hypersonic nozzle expansion of air with atom recombination

present .

.A

nagamatsu,h.t., workman,j.b. and sheer,r.e.

.B

j. ae. scs. 1961, 833.

.W

hypersonic nozzle expansion of air with atom recombination

present .

an experimental investigation on the expansion of high-

temperature, high-pressure air to hypersonic flow mach numbers in

a conical nozzle of a hypersonic shock tunnel has been carried out .

the equilibrium temperature and pressure ranges after the

reflected shock wave were 1400 to 6000 k and 100 to 1000 psia .

static-pressure measurements, which are sensitive to the state

of the gas, were made along the axis of the nozzle for different

reservoir conditions . these results are compared with the

calculated equilibrium and /frozen/ data for the same geometry and initial reservoir conditions .

for reservoir pressures greater than 500 psia, the expansion of the air in the nozzle is essentially in equilibrium up to reservoir temperatures of about 4,500 k . for temperatures greater than almost frozen . at a given area ratio for the nozzle and reservoir pressure, the expansion process remains in equilibrium up to a certain reservoir temperature, and beyond this temperature the flow expansion deviates rapidly from the equilibrium process and approaches the frozen case .

.I 1231

.T

hypersonic flow over an elliptic cone: theory and experiment .

.A

chapkis,r.l.

.B

j. ae. scs. 1961, 844.

.W

hypersonic flow over an elliptic cone: theory and experiment .

by applying hypersonic approximations to ferri's linearized characteristics method, simple results were obtained for the shock shape and surface pressure distribution for an unyawed conical body of arbitrary cross section . calculations were carried out for an elliptic cone having a ratio of major to minor axes of, and a semivertex angle of about 12 in the meridian plane containing the major axis . an experimental investigation of the flow over this body conducted at a mach number of 5.8 in the galcit hypersonic wind tunnel showed that the surface pressure distribution at zero angle of attack agreed quite closely

with the theoretical prediction . on the other hand, the simple

newtonian approximation predicts pressures that are too low .

surface pressure distributions and schlieren photographs of

the shock shape were obtained at angles of attack up to 14 at

zero yaw, and at angles of yaw up to 10 at zero pitch . at the

higher angles of attack the newtonian approximation for the

surface pressures is quite accurate .

.I 1232

.T

the curtain jet .

.A

ehrich,f.f.

.B

j. ae. scs. 1961, 855.

.W

the curtain jet .

a detailed analytic study is made of the curtain jet, the

two-dimensional fluid wall used to contain support pressure on the

underside of ground effect machines . two variations of the jet

are studied in detail--the bifurcated jet, in which a portion of the

flow streams into the support pressure region, and the deflected

jet, in which none of the flow penetrates into the support pressure

region . kirchhoff-helmholtz free steamline analysis is used

to construct the flow field, and quantitative results are presented

for the effect of nozzle inclination and detailed geometry on flow

requirements and support pressure differential at varying altitudes .

.I 1233

.T

supersonic shear flow past an airfoil between two parallel

walls .

.A

rhyming,i.l.

.B

j. ae. scs. 1961, 861.

.W

supersonic shear flow past an airfoil between two parallel

walls .

the supersonic flow with assigned mach number gradient in

the span direction past a straight wing between two parallel

walls is studied using the small-disturbance theory . the

governing equation for the disturbance pressure on the airfoil, together

with the boundary conditions on the airfoil and at the walls, is

solved by the method of separation of variables . upon

separation the problem is reduced to a sturm-liouville eigenvalue

problem and to the solution of the telegraph equation .

as an application, a certain mach number profile is selected

and the resulting pressure distribution on a parabolic arc airfoil

is computed .

.I 1234

.T

direct calculation of pressure distribution on blunt

hypersonic nose shapes with sharp corners .

.A

holt,m.

.B

j. ae. scs. 1961, 872.

.W

direct calculation of pressure distribution on blunt

hypersonic nose shapes with sharp corners .

the method of belotserkovskii for calculating hypersonic

flow fields past a circular cylinder is extended to deal with

axially-symmetric flow past sharp-cornered nose shapes, in particular,

spherical segments and flat-headed cylinders . results on

spheres are also included . in the present paper belotserkovskii's

first approximation is considered, and comparison of calculated

pressure distribution and shock shape with experimental results

shows very good agreement .

.I 1235

.T

a theory of the two dimensional laminar bounary layer

over a curved surface .

.A

yen,k.t. and toba,k.

.B

j. ae. scs. 1961, 877.

.W

a theory of the two dimensional laminar bounary layer

over a curved surface .

the purpose of this paper is to present a theory to account for surface curvature effects on the two-dimensional boundary-layer flow which approaches a potential flow at free stream .

the problem of two-dimensional viscous flow is first formulated by using the streamlines and their orthogonal trajectories as the generalized coordinates . a boundary-layer approximation is applied to the navier-stokes equations and the gauss equation in the generalized coordinates to yield the boundary-layer equations . the conditions under which similar solutions of the boundary-layer equations exist are determined . by a simple transformation, the governing differential equation can be expressed in a form which reduces to the falkner-skan equation for zero surface curvature .

numerical results for a similar solution which corresponds to a flow over a curved surface with zero surface pressure gradient have been obtained . the velocity profiles in the boundary layer and the wall skin-friction distribution for concave and convex surfaces are presented . the wall skin friction for a convex wall is found to be higher than the blasius value for a flat plate . on the other hand, for a concave wall, the skin friction will drop below the blasius value as the curvature increases, but it appears to reach a minimum, and beyond this minimum point it will increase again . the same flow problem was treated by murphy by a different method of analysis . comparison of murphy's results with those obtained by the present method reveals some basic differences in the boundary-layer characteristics . in particular, murphy's results indicate that the wall skin friction for a convex surface is smaller than the blasius value, while for a concave wall it is higher .

.I 1236

.T

the stagnation point boundary layer in the presence

of an applied magnetic field .

.A

bush,w.b.

.B

j. ae. scs. 1961, 610.

.W

the stagnation point boundary layer in the presence

of an applied magnetic field .

similarity equations for axisymmetric compressible flow are

obtained, assuming that the magnetic field is uniform, normal to the

surface, and unaffected by the flow, and that the conductivity

varies as the nth power of the enthalpy . numerical solutions are

given for a number of values of n and of the field strength, and are

used to modify the estimates of heat-transfer made by the author

using inviscid theory (title source 26, 536-537, 1959) .

.I 1237

.T

foreign-gas injection into a compressible turbulent

boundary layer on a flat plate .

.A

ness,n.

.B

j. ae. scs. 1961, 645.

.W

foreign-gas injection into a compressible turbulent

boundary layer on a flat plate .

the distributed injection of a foreign gas into a compressible, turbulent boundary layer in the absence of a pressure gradient is considered . the analysis is performed within the framework of the binary-mixture concept, that is, the primary fluid flowing over the surface represents one component while the injected species represents the second .

calculations have been performed for the injection of helium into undissociated air . the results indicate an effect of mach number on surface shear and energy transfer when distributed light-gas injection normal to the surface exists . a comparison with experimental data indicates reasonable agreement over a range of mach numbers .

.I 1238

.T

the newtonian approximation in magnetic hypersonic

stagnation-point flow .

.A

lykoudis,p.s.

.B

j. ae. scs. 1961.

.W

the newtonian approximation in magnetic hypersonic

stagnation-point flow .

the hypersonic flow of an electrically conducting fluid around the stagnation region of a sphere carrying a radial magnetic field

is examined .

by assuming a newtonian pressure distribution and constant

density, the differential equation of the inviscid flow is integrated

and a simple closed-form solution is obtained .

it is found that the ratio of the stand-off distances of the shock

wave for the magnetic and nonmagnetic cases does not depend

explicitly on the magnetic parameter s (ratio of the ponderomotive

force to the free-stream inertia force) nor on the density ratio (the

value at the free stream divided by the value behind the shock

wave) but on the product s at least for values of between

and .

the velocity gradient on the body is also calculated and the

ratio of the magnetic to the nonmagnetic case is shown to depend

on the parameter .

the case of cylindrical shocks is also examined,. the same

general conclusions are drawn .

.I 1239

.T

body under lifting wing .

.A

chen,c.f. and clarke,j.h.

.B

j. ae. scs. 1961, 547.

.W

body under lifting wing .

an investigation is made of supersonic-aircraft configurations composed of a cambered body positioned a certain distance beneath an arbitrary lifting wing . the geometry of the wing is regarded as given and the geometry of the body may be given or optimum . expressions for the drag and lift are obtained from reverse-flow considerations,. these greatly implement such a study when interference cross flows must be cancelled . the drag advantage to be gained when a given body and wing assume a given orientation is studied . treated more extensively is the variational problem of determining the optimum wing incidence and optimum body shape, for the given volume and length, to yield the minimum drag for prescribed lift . numerical results are provided to indicate the significance of the large number of parameters appearing in the problem . of these, the gap between the wing and the body is found to be particularly important . it is found that at low gap moderate body distortions have a significant influence on the drag . drag reductions of up to 44 relative to the case of no interference have been found at a mach number of 2.24 in a configuration having a gap approximately equal to the maximum diameter of the body, and a wing chord of about three eighths of the length of the body . comparison is made with the conventional wing-body combination including the effects of skin friction, and it is concluded that the advantage suggested by the preceding considerations is not appreciably diminished . finally, it is shown that the configurations studied lead to bodies of fineness ratios much lower than are appropriate to conventional wing-body combinations .

tests were made on an arrangement consisting of a scars-haack body located under a lifting rectangular diamond-profile wing . the mach number was 1.6 and the reynolds number was 9.17 x 10 based on the body length . it was found that the measured lift developed on the wing due to the flow field of the body agrees very well with the theoretical value . downstream of the impinging shock from the wing, flow separation was observed on the exterior of the body but not in the interior . the separation is attributed not to the pressure rise across the shock but to the pressure field arising from the reflection from the body of the shock-induced cross flow . further observations suggest that the separation can be avoided by pitching the body or by kinking the body at the shock wave to accommodate the shock-induced cross flow .

.I 1240

.T

nonsimilar solutions of the compressible laminar boundary layer equations with applications to the upstream-transpiration cooling problem .

.A

pallone,a.

.B

j. ae. scs. 1961, 449.

.W

nonsimilar solutions of the compressible laminar boundary layer equations with applications to the upstream-transpiration cooling problem .

a new method is presented for predicting the boundary-layer

characteristics downstream of the porous region of an
injection-cooled surface . the method consists of a general scheme for
obtaining nonsimilar solutions of the compressible-
laminar-boundary-layer equations and is formulated along the following
lines . the viscous domain is divided into n curvilinear strips .
the governing equations are then integrated along the
coordinate normal to the body from the surface to the boundary of each
strip . as a result, one obtains a set of independent
integro-differential relations . the integration is carried out by
expressing the integrands as polynomials, the coefficients of which
are functions of the unknown values of the velocity and
temperature on the strip boundaries as well as of the imposed
boundary condition at the wall and at the outer edge . after the
integration is performed, a set of ordinary first-order differential
equations is obtained . the set of equations may be solved for
given initial conditions by a numerical integration scheme such
as the runge-kutta method . several numerical examples of
interest are presented .

.I 1241

.T

the turbulent boundary layer on chemically active ablating
surfaces .

.A

denison,m.r.

.B

j. ae. scs. 1961.

.W

the turbulent boundary layer on chemically active ablating surfaces .

incompressible turbulent-boundary-layer analysis is extrapolated analytically to the case of a compressible turbulent boundary layer with ablation or mass injection at the surface . the effects of chemical reactions such as dissociation and recombination as well as combustion are included . the analysis applies to blunt as well as sharp bodies which are either axisymmetric or two-dimensional . when the turbulent lewis and prandtl numbers are unity, it is found that, as in the laminar case, little detailed knowledge of the chemistry inside the boundary layer is required in most instances . the conditions at the surface and the outer edge of the boundary layer are often sufficient for prediction of heat and mass transfer .

comparison is made with experiments on the combustion of graphite under turbulent flow conditions . prediction of ablation rates within about 30 percent accuracy is obtained when empirical constants obtained from incompressible velocity profiles with no mass injection are used .

.I 1242

.T

some considerations on the laminar stability of time-dependent basic flows .

.A

s. f. shen

.B

u.s. naval ordnance laboratory

.W

some considerations on the laminar stability of time-dependent basic

flows .

as a stability criterion for infinitesimal disturbances in an

incompressible, parallel but time-dependent basic flow, it is proposed

to introduce the concept of /momentary stability,/ which is said

to prevail at the instant if the kinetic energy of the disturbances,

as a fraction of the kinetic energy of the basic flow, tends to

decrease . the significance of such a criterion is briefly discussed .

for special time-dependent basic flows which are described by similar

velocity profiles at all times (except for changes in amplitude),

in the inviscid limit only a change of the time scale is needed to

reduce the solution essentially to that for the steady case . the

disturbances may be of either the transverse-wave or the

longitudinal-vortices type . the result indicates a very strong destabilizing

influence of deceleration, which is likely to overshadow that of the

velocity profile under normal circumstances . the observations of fales

rotating cylinders) are believed to be largely due to the deceleration .

at finite reynolds numbers, the usual procedure of calculating the

stability solution on the basis of the instantaneous profile is further

shown to be valid only for extremely slow acceleration or deceleration .

even when the solution is acceptable, the condition for neutral

stability may not be used without reservation . to calculate

momentary stability properly, a procedure for a slowly varying but more

general profile is also described .

.I 1243

.T

supersonic boom of wing-body configurations .

.A

ryhming,i.l. and yoler,y.a.

.B

j. ae. scs. 1961, 313.

.W

supersonic boom of wing-body configurations .

the supersonic boom in steady, level flight of a wing-body configuration is due to the effects of body volume, wing volume, wing incidence or lift, and wing-body interference . the contribution in the far field of each of these factors can, in any given azimuthal plane, be represented as that of an equivalent body of revolution . this concept is developed to investigate the possibilities of using interference among the components of a wing-body configuration to reduce or suppress the boom due to lift . results of wind tunnel experiments are also presented and discussed in light of the theoretical indications .

.I 1244

.T

on the aerodynamic noise of a turbulent jet .

.A

cheng,sin-i.

.B

j. ae. scs. 1961, 321.

.W

on the aerodynamic noise of a turbulent jet .

a new model is advanced for analyzing the broad-spectrum noise of a turbulent jet . the shear layer bounding the turbulent jet is assumed to play an important role in modifying the / quadrupole sound radiation/ from the interior . to the sound-emitting small-scale turbulent eddies (with frequencies much higher than those of large-scale eddies), the laminar shear layer has an irregular contour, as if the large-scale turbulent motions were frozen . the linearized analysis is then applied to the laminar shear layer to relate the acoustic oscillations across it .

the concept of geometrical acoustics is generalized to represent the passage of an acoustic ray through a laminar shear layer . acoustic rays may be traced across the shear layer as transmission and refraction, but they may also be apparently /absorbed/ or /generated/ by the laminar layer . this /generation/ is visualized as the schematic representation, within the framework of geometrical acoustics, of the action of the reynolds stress in transferring energy from the shearing mean flow to the acoustic waves . such action of the reynolds stress can be neglected in ordinary acoustics when the acoustic medium is not moving at speeds comparable to the speed of sound in the medium . however, this action is of crucial importance in the aerodynamic noise of high-speed turbulent jets where the reynolds stress is the fundamental element of the radiating quadrupoles, according to lighthill .

those acoustic waves that become /stationary/ with respect to the local mean flow somewhere in the interior of the shear layer are significantly modified by the viscous action through the critical layer . the shear layer therefore serves as a selective amplifier of the acoustic waves passing through it .

kinematically, the shear layer brings about the preferred downstream

emission .. dynamically, the shear-layer augmentation

significantly increases the polar peak noise level . the acoustic power

output per unit solid angle for such downstream emissions

augmented by the shear layer (including the polar peak) varies as,

predicted by lighthill, but without lighthill's convective

corrections . on the other hand, the acoustic power output per

unit solid angle nearly normal to the jet, due to the transmitted

downstream-propagating waves, varies roughly as . heating

the jet gas increases the shear-layer augmentation and may

increase the polar peak noise level by several db . the silencing

action of the edge notches and edge teeth may also be

interpreted as due apparently to the result of possible distortion of

the shear-layer profiles .

.I 1245

.T

some aspects of nonequilibrium flows .

.A

sedney,r.

.B

j. ae. scs. 1961, 189.

.W

some aspects of nonequilibrium flows .

in this paper are discussed some of the general features of

nonequilibrium flow . in particular, vibrational relaxation is

discussed in detail . this case is somewhat simpler than dissociation and ionization but it illustrates some of the main new features of nonequilibrium flow . those aspects of two-dimensional and axisymmetric flow behind shock waves are examined analytically which yield significant information without requiring numerical solution of the governing equations .

the thermodynamics of a vibrational relaxing gas are discussed . the conditions for simulating flows are noted . crocco's theorem and the characteristic equations are derived . then a simple method of obtaining the initial gradients of the flow variables behind a shock is shown . these gradients are used in discussing two particular flows . an exact solution for flow over a cusped body is obtained . flow over a wedge near the tip and far from the tip is considered . it is found that far from the tip a boundary-layer type phenomenon occurs .

.I 1246

.T

solution of subsonic nonplanar lifting surface problems
by means of high-speed digital computers .

.A

kilakowski,l.j. and haskell,r.n.

.B

j. ae. scs. 1961.

.W

solution of subsonic nonplanar lifting surface problems
by means of high-speed digital computers .

the method proposed in this paper is based on an approximate solution of the integral equation which represents the potential flow about a finite wing, with no restrictions beyond those necessary for linearization . after assuming the usual series representation of the wing surface vorticity distribution, the solution is achieved by approximating portions of the kernels of the transformed integral equation by single and double fourier series and performing termwise integrations analytically . this is followed by the routine inversion of the aerodynamic influence coefficient matrix, after satisfying appropriate boundary conditions at selected control points . in this procedure the number of control point used is limited only by the storage capacity of the computer . control points may be located so as to cover the entire wing surface, with due regard to certain physical requirements, permitting the accurate representation of complicated mean surface shapes .

an evaluation of the proposed method is included . comparisons with other theoretical methods and electrical analogy tank results are used to substantiate the accuracy of the proposed method when applied to plane wings . a final evaluation involves a comparison of calculated surface pressure distribution with wind-tunnel measurements on a swept, tapered wing with a cambered and twisted mean surface . the agreement evidenced in the latter comparison has the same order of overall accuracy as similar comparisons on plane wing planforms . in either case, the results given by the proposed method are within the accuracy requirements for most aircraft design studies .

.I 1247

.T

the supersonic boom of a projectile related to drag

and volume .

.A

ryhming,i.l.

.B

j. ae. scs. 1961.

.W

the supersonic boom of a projectile related to drag

and volume .

the whitham theory predicting the far-flow field around a projectile is used to derive body shapes which produce extreme bow shock-wave pressure jump or /boom,/ subject to constraining conditions regarding the drag due to the bow shock and fineness ratio of the bodies . it is found that the minimum drag body is also the minimum boom body . the body-volume effect and the effect of discontinuities in slope of the body meridian section on the boom intensity is investigated .

as a general result of the investigation, it can be said that the boom of a projectile for given mach number and flight altitude is primarily determined by its length and fineness ratio . the maximum variation in the boom intensity for pointed bodies with given length and fineness ratio is of the order of 10 per cent . the geometry of the bodies is thus found to play a minor role .

.I 1248

.T

an analytic extension of the shock-expansion method .

.A

waldman,g.d. and probstein,r.f.

.B

j. ae. scs. 1961, 119.

.W

an analytic extension of the shock-expansion method .

the problem is considered of calculating approximately the inviscid rotational flow field and pressure distribution about a smooth two-dimensional airfoil with sharp leading and trailing edges in a uniform supersonic or hypersonic stream . the assumption of a perfect gas is made, and the basic flow pattern for the analysis is taken to be given by the simple isentropic shock-expansion method with straight characteristics . an elementary characteristics treatment is discussed to show when the simple shock-expansion method should be satisfactory for computing the surface pressure distribution, and under what circumstances it may be expected to break down . by utilizing characteristic variables the isentropic shock-expansion method is then formulated analytically, and an analytic result is obtained for the shock shape corresponding to this zero-order approximation . in the special case where hypersonic similitude is applicable, that is, for slender bodies and high mach numbers, the shock-shape expression for large distances is found to reduce to the result previously given by mahony, which for weak shocks and slender bodies in turn reduces to the simple-wave result first given by friedrichs .

employing the analytic form of the isentropic shock-expansion method as a zero-order approximation, an analytically consistent perturbation method is developed by expanding the dependent

flow variables in the exact partial differential equations in powers

of the reflection coefficient for simple waves interacting with an

oblique shock . the scheme by its nature helps to define those

regions in which shock expansion can be used, in addition to

taking into account in a perturbation sense the factors neglected

in simple shock-expansion theory, namely, the curvature and

reflection of the mach waves and the correct boundary

conditions at the shock wave . analytic solutions are obtained for

the first-order corrections, including the surface pressure

distribution . the necessary numerical computation of the integrals

involved is considerably simpler than a direct application of the

method of characteristics . to illustrate the method and its

accuracy, the zero-order shock shape and first-order pressure

distribution are calculated for a family of parabolic arc airfoils at

an infinite free-stream mach number . these results are

compared with rotational characteristic solutions where available,

and the present method is found to be in excellent agreement .

.I 1249

.T

plasma flow over a thin charged conductor .

.A

yoshihara,h.

.B

j. ae. scs. 1961, 141.

.W

plasma flow over a thin charged conductor .

the flow of a dense plasma over a wavy conducting wall of small amplitude is investigated where magnetic effects are negligible . these results are then used to analyze the flow over a thin conductor with cusped edges . it is found that the coulomb drag vanishes identically, while the fluid-pressure drag corresponds to the ackeret value for a neutral particle gas at the reduced-plasma mach number .

.I 1250

.T

high-speed viscous corner flow .

.A

bloom,m.h. and rubin,s.

.B

j. ae. scs. 1961, 145.

.W

high-speed viscous corner flow .

a boundary-layer integral method analysis is set up for compressible laminar flow in a symmetric corner with varying angle and streamwise pressure gradient . it represents an extension and modification of the constant density analysis of loitsianskii and bolshakov . the analysis is applied to the case of constant pressure, constant corner angle, and isothermal surfaces, for which the crocco velocity-enthalpy relation holds . although simplifying assumptions limit the quantitative accuracy outside the 60 to 120 angle range, some qualitative trends are probably correct outside this range . the limiting cases near 0 and 180 are not considered .

favorable agreement between some results obtained by the integral method and by other methods is demonstrated for the isothermal, constant-density case .

results show an increasingly sharp merger of the outermost isovels of streamwise velocity as the mach number increases . this sharp merging of the outer isovels is increased by increasing corner angle and by insulation of heating of the surfaces . within the interior of the viscous layer the spreading or contraction of the disturbed region of merging is influenced by surface heat-transfer conditions . surface shear and heat flux are decreased in the disturbed region, and are zero at the apex . for cases corresponding roughly to the higher mach numbers of wider corner angles, the /specific momentum-area/ exhibits the same decrease with mach number as its two-dimensional counterpart, whereas the /specific displacement-area,/ a measure of stream-tube dilation, increases more rapidly with mach number than the comparable two-dimensional parameter .

.I 1251

.T

viscous flow past a quarter infinite plate .

.A

stewartson,k.

.B

j. ae. scs. 1961, 1.

.W

viscous flow past a quarter infinite plate .

a study is made of the motion of an incompressible viscous fluid past a quarter-infinite plate, whose leading edge is

perpendicular to and whose side edge is parallel to the undisturbed

direction of the stream . it is assumed that the kinematic

viscosity is small . the first approximation is taken to be the

undisturbed motion, and successive approximations are obtained

by iferation . the second approximation is the blasius shear

layer necessary to satisfy the boundary conditions on the plate .

in turn, this layer leads to a velocity component normal to the

plate which needs a potential solution, in which the velocities

are 0, to match with the conditions at infinity . further,

the match at the edge of the blasius shear layer must be

completed to 0 by introducing a secondary shear layer . the

regions near the leading and side edges are considered separately,.

in particular, the neighborhood of the side edge needs special

care, because the determination of the chief terms is complicated

by the presence of powers of log . in particular it is shown

that the effect of the edge is to change the skin-friction coefficient

by a factor

.I 1252

.T

on the approach to chemical and vibrational equilibrium

behind a strong normal shock wave .

.A

dorrance,w.h.

.B

j. ae. scs. 161, 43.

.W

on the approach to chemical and vibrational equilibrium
behind a strong normal shock wave .

the concurrent approach to chemical and vibrational equilibrium of a pure diatomic gas passing through a strong normal shock wave is investigated . it is demonstrated that the equilibrium degree of dissociation behind the shock front, and hence the density, for the case where the vibrational degrees of freedom are frozen out can exceed the degree of dissociation, and hence the density, for the case where all degrees of freedom are in equilibrium . thus the necessary condition for a maximum of the density between the shock front and the position of full equilibrium flow downstream of the shock front is established . the sufficient condition that such a maximum be observable is shown to be that the approach to equilibrium of the vibrational degrees of freedom (or any other internal degrees of freedom) must lag the approach to dissociation equilibrium by a significant amount,. that is, there must be at least an order of magnitude difference in the respective relaxation times before such a maximum might be observed . an example calculation for a mach 13 strong shock wave in oxygen illustrates the appearance of such a maximum of the density and its dependency upon the relative values of the vibration and dissociation relaxation times .

.I 1253

.T

hypersonic viscous flow near the stagnation point in
the presence of magnetic field .

.A

wu,ching-sheng.

.B

j. ae. scs. 1960, 882.

.W

hypersonic viscous flow near the stagnation point in

the presence of magnetic field .

the present study investigates the hypersonic viscous flow

past blunt-nosed bodies with hydromagnetic interaction .

local-similarity solutions of flow field and temperature distribution are

near the stagnation-point region . the discussions may be

grouped into two parts .. the two-dimensional problem (circular

cylinder) and axisymmetric problem (sphere) .

numerical computations have been carried out for the sphere

problem for the /viscous-layer regime,/ with various magnetic

field strengths and electrical conductivities .

.I 1254

.T

combustion in the boundary layer on a porous surface .

.A

eschenroeder,a.q.

.B

j. ae. scs. 1960, 901.

.W

combustion in the boundary layer on a porous surface .

the position of the diffusion flame in a boundary layer with

uniform mixture injection from a porous wall parallel to a uniform

air stream is determined under the conditions of laminar, steady

flow with zero streamwise pressure gradient . under the
assumption of fast forward reaction rate, solutions of the boundary
layer forms of the conservation laws of acrothermochemistry are
obtained leading to a formula for the downstream velocity at
the flame in terms of composition and flow variables . the
rates of change of conditions at the wall in the streamwise
direction are assumed to be relatively small . methods of
treating complex reaction systems are described, and generalized
form of the reynolds analogy is developed .

.I 1255

.T

the flow about a charged body moving in the lower atmosphere .

.A

hunziker,r.r.

.B

j. ae. scs. 1960.

.W

the flow about a charged body moving in the lower atmosphere .

the flow about an electrically charged body traveling at high
speeds through the lower ionosphere is analyzed . a simple
gas model composed of electrons, ions, and neutral particles is
used and the hydrodynamic description given is based on
maxwell's transfer equations for a mixture .

the conditions under which local statistical equilibrium can
be assumed are discussed, and different approaches to determine
the gasdynamic force in the subsonic, supersonic, and hypersonic
cases are indicated . the reciprocal action of the electric field

of the flow on the body is also analyzed and a formula for the

resultant electric force is given . the total force on the body

is equal to the sum of the gasdynamic force and the electric force .

the negative potential acquired by a plane body is also

calculated . finally, the lack of validity of debye's linearization

in this case and the solution of the exterior nonlinear problem

which characterize the electric potential and the electron

distribution are discussed .

.I 1256

.T

fluctuating lift and drag acting on a cylinder in a

flow at supercritical reynolds numbers .

.A

fung,y.c.

.B

j. ae. scs.1960, 801.

.W

fluctuating lift and drag acting on a cylinder in a

flow at supercritical reynolds numbers .

the fluctuating lift and drag acting on a circular cylinder in a

flow of an incompressible fluid at large reynolds numbers were

measured . data on the root-mean-square values of the lift and

drag coefficients, the extreme values of these coefficients, and

their power spectra at various reynolds numbers are presented .

.I 1257

.T

an optical boundary-layer probe .

.A

daiber,j.w.

.B

j. ae. scs. 1960.

.W

an optical boundary-layer probe .

the applicability of the schlieren-photomultiplier technique to obtain quantitative density measurements in the laminar boundary layer induced by a traveling shock wave in a shock tube is investigated . tests were conducted at a mach number of 1.58 so that the data could be compared with the exact theoretical solution tabulated by mirels . the data obtained are in good agreement with the theory if the distance of the light beam above the floor of the shock tube is adjusted to fit the theoretical curve,. this would not be necessary if a larger shock tube were used . values of the transition reynolds number were also determined which are slightly less than those found by martin using an interferometer . it is shown that this technique is sensitive enough to detect changes in density that are only 0.000 per cent of atmospheric density .

.I 1258

.T

heat transfer, recovery factor and pressure distributions around a circular cylinder normal to a supersonic rarefied air stream .

.A

tewfik,o.k. and giedt,w.h.

.B

j. ae. scs. 1960, 721.

.W

heat transfer, recovery factor and pressure distributions

around a circular cylinder normal to a supersonic rarefied

air stream .

measurements of the heat transfer, recovery factor, and

pressure distributions around a circular cylinder normal to a

supersonic rarefied-air stream (total temperature 300 k.)

are described for the mach number range of 1.3 to 5.7, the

reynolds number range of 37 to 4,100 and at cylinder wall

average temperature levels of 90 k. and 210 k . study of the

results yielded .. (1) a correlation equation for the

stagnation-point nusselt number as a function of the reynolds number just

after the normal part of the detached bow shock wave,. and (2)

fourier series expressions for the heat-transfer coefficient and

pressure coefficient distributions in terms of the stagnation point

values .

in comparing these measurements with predictions based on

recent analytical studies, exceptionally good agreement for the

heat-transfer coefficient distribution was obtained with lees'

theory . in the mach number range of 3.55 to 5.73 the

pressure decreased less rapidly with distance from the stagnation point

than predicted by the modified newtonian theory .

.I 1259

.T

second-order theory for unsteady supersonic flow past slender pointed bodies of revolution .

.A

revell,j.d.

.B

j. ae. scs.1960, 730.

.W

second-order theory for unsteady supersonic flow past

slender pointed bodies of revolution .

an analysis is made of the second-order effects of thickness on

the unsteady aerodynamic forces on a slender pointed body of

revolution in supersonic flow . the theory is restricted to

harmonic oscillations for small angles of attack . the solution is

obtained by approximating the nonlinear terms in the second-order

potential equation by their first-order values and solving the

resulting inhomogeneous partial differential equation, subject to

more refined boundary conditions . the pressure equation is

likewise refined and integrated to give the second-order

corrections to lift and pitching moment coefficients . the analysis can

be considered as an extension of the second-order, slender body

theory of lighthill to the case of unsteady flow .

the results indicate appreciable reductions in unsteady lift

and damping moment coefficients when applied to slender cones .

the present theory is estimated to be reliable provided that

is less than 0.7 .

.I 1260

.T

on the response of the laminar boundary layer to small fluctuations of the free-stream velocity .

.A

rott,n. and rosenzweig,m.l.

.B

j. ae. scs. 1960, 741.

.W

on the response of the laminar boundary layer to small

fluctuations of the free-stream velocity .

the linearized treatment of small time-dependent disturbances

of a laminar boundary layer, initiated by lighthill, is extended in

several ways . in particular, the high-frequency expansion is

continued beyond the leading (stokes) term . several interesting

questions of /joining/ occur, which are discussed but left

unresolved . in addition, a practical method for obtaining the

response to the laminar boundary layer to an impulsive change in

velocity is presented . the methods are applied to the case in

which the basic steady flow belongs to the falkner and skan

family of similarity solutions .

.I 1261

.T

a method of calculating velocity distribution for turbulent

boundary layers in adverse pressure distributions .

.A

uram,e.m.

.B

j. ae. scs. 1960, 659.

.W

a method of calculating velocity distribution for turbulent boundary layers in adverse pressure distributions .

a new method of calculating the behavior of turbulent boundary layers in adverse pressure distributions is developed which permits direct determination of the velocity profile rather than the gross integral parameters normally used to infer the general character of the boundary layer . the method offers the simplicity of algebraic equations coupled with the use of charts rather than the laborious simultaneous solution of coupled differential equations required by existing methods . the method also affords, for the first time, a means of determining the total boundary-layer thickness, thus allowing calculation of the absolute as well as the nondimensional velocity distribution .

the velocity profile is considered to be composed of two regions--an inner region which is described by the law of the wall and an outer region which is described by a function depicting the deviation from that law . the deviation function involves two parameters which are uniquely dependent upon the skin-friction coefficient and a third parameter which, for practical purposes, can be considered a constant . since the entire velocity distribution was found to be almost uniquely dependent upon the local skin friction, serious doubt is cast upon the generally accepted /history concept/ which considers the outer region of the boundary layer to be dependent on integrated upstream conditions .

agreement between experimental velocity distributions and those calculated by the method presented here is generally very

good . the analysis and calculation procedures which are
presented are applicable to two-dimensional, pseudo-two-
dimensional, and axisymmetric conical flows .

.I 1262

.T

an extension of the linearized characteristics method
for calculating the supersonic flow around elliptic
cones .

.A

martellucci,a.

.B

j. ae. scs. 1960, 667.

.W

an extension of the linearized characteristics method
for calculating the supersonic flow around elliptic
cones .

the method of linearized characteristics as applied by ferri to
the flow about elliptic cones can be used to determine the surface
pressure distribution, even when only linear terms are kept in the
boundary conditions, provided an area rule requirement is
satisfied . in addition, the method can be applied for angles of attack
provided the elliptic body geometry is specified in a manner that
does not distort the cross section . the surface pressure
distribution obtained by this modified method is in reasonable
agreement with experiment over the range of mach numbers and
semidiameter ratios considered . experimental results for several
conical bodies are presented .

.I 1263

.T

turbulent heat transfer through a highly cooled, partially

dissociated boundary layer .

.A

rose,p.h., probstein,r.f. and adams,m.c.

.B

j. ae. scs. 1958, 751.

.W

turbulent heat transfer through a highly cooled, partially

dissociated boundary layer .

the problem of heat transfer from high-temperature air

through a turbulent boundary layer to a cold surface is

considered both analytically and experimentally . heat-transfer

data obtained in shock tubes are presented and correlated by a

semiempirical theory which includes the effect of atomic diffusion .

the distinguishing characteristics of turbulent boundary

layers with dissociation and large cooling are considered . it is

shown that the equations governing such flow, after certain

approximations, can be represented in a form similar to the

classical equations for a turbulent boundary layer .

an approximate theory is proposed for turbulent heat transfer

for a highly cooled boundary layer on portions of the body where

the pressure gradient is negligible in the case of blunted bodies

of revolution in high-speed flight .

experimental results obtained on the cylindrical portion of a

hemisphere-cylinder model are presented for conditions simulating flight speeds to 21,350 ft. sec., where up to 30 per cent of the molecules are dissociated . reynolds numbers of 2.5 x 10, based on local fluid properties external to the boundary layer, were achieved . the larger values of reynolds number and flight speed were not obtained simultaneously, due to structural limitations of the shock tubes,. however, the experiments were conducted in such a way that the important effects of each could be determined .

in the experiments the mach number external to the boundary layer varied between 1.7 and 2.2 . the corresponding mach number for blunted nonslender bodies in flight would have a maximum value between 2.5 and 4,. however, it is shown that these differences in mach number are not important for such bodies .

.I 1264

.T

boundary layer transition and heat transfer in shock tubes .

.A

hartunian,r.a., russo,a.l. and marrone,p.v.

.B

j. ae. scs. 1960, 587.

.W

boundary layer transition and heat transfer in shock tubes .

an experimental study is made of the wall boundary layer in a shock tube operated over a wide range of shock mach numbers and pressure levels in air, including those for which real-gas effects exist . transition distances are determined and correlated in terms of the transition reynolds number based on a characteristic length for this boundary layer . data from independent shock-tube studies are also included in this correlation .

the results indicate a weak dependence of transition reynolds number on shock strength up to moderate values of shock mach number, followed by a larger stabilizing tendency . comparison of these data with transition data obtained in the same manner in argon indicate that the increased cooling rates are largely responsible for the stabilization .

a dependence of transition reynolds number on the unit reynolds number is found at the lower shock strengths . specifically, higher transition reynolds numbers are achieved at larger unit reynolds numbers .

the phenomenon of transition reversal does not appear within the range of the experiments reported .

laminar- and turbulent-flow heat-transfer rates to the walls of the shock tube are determined experimentally . the results of the heat-transfer measurements substantiate existing theories in both the laminar- and turbulent-flow regimes .

.I 1265

.T

on the thrust hypothesis for the jet flap including

jet-mixing effects .

.A

yen,k.t.

.B

j. ae. scs. 1960, 607.

.W

on the thrust hypothesis for the jet flap including

jet-mixing effects .

this paper is concerned with the thrust generated by a jet

flap . it is shown that a /linear/ thrust hypothesis can be

obtained, provided linearized potential flow is assumed . in fact,

the linearized problem of a jet-flap system is found to be the

linear combination of a lift problem and a thrust problem . the

lift problem gives all the lift generated, but it is of interest to note

that the thrust problem would yield all the thrust developed by

the jet flap within the limitation of the linearized theory .

the mixing of the jet flap with the surrounding fluid is analyzed

by the momentum-integral method . the analysis substantiates

stratford's suggestion for obtaining an increase of thrust by

causing the jet to mix with the main stream in a region of high

suction . finally, some approximate formulas, relating the thrust

and the jet angle, are derived . the drag of the airfoil section and

other viscous effects are, however, not considered .

.I 1266

.T

minimum wing wave drag with volume constraint .

.A

strand,t.

.B

j. ae. scs. 1960, 615.

.W

minimum wing wave drag with volume constraint .

  a numerical method is developed for calculating the minimum

thickness drag for a given wing planform and volume using

linearized supersonic flow theory . the corresponding optimum

volume distribution is also determined . the results show that

considerable drag reduction is possible by improved volume

distribution .

.I 1267

.T

on supersonic flow past thick airfoils .

.A

kogan,a.

.B

j. ae. scs. 1960, 504.

.W

on supersonic flow past thick airfoils .

  the inviscid rotational supersonic flow behind the shock wave

attached to the sharp leading edge of an airfoil is studied by a

transformation of coordinates which introduces the crocco stream

function as an independent variable .

  using expansions in the power series of, an iterative process

is developed for the determination of pressure distribution along

the airfoil surface .

.I 1268

.T

stable combustion of a high-velocity gas in a heated boundary layer .

.A

turcotte,d.l.

.B

j. ae. scs.1960.

.W

stable combustion of a high-velocity gas in a heated

boundary layer .

it is generally recognized that stable combustion processes in heated boundary layers may be achieved by either of two conceptual mechanisms . in one mechanism it is pictured that the heat transfer to the wall quenches the propagating flame at a certain distance from the surface . the equality between the flow velocity and the normal burning velocity at this quenching distance determines the position of the propagating flame . in the second mechanism it is conceived that the hot surface provides a continuous source of ignition in much the same manner that the hot recirculation zone of a bluff body flame holder provides continuous ignition to the gas flowing around it . in this case it is the characteristic time during which the gas must be heated that determines the position of the flame .

all experimental work reported to date has been concerned with conditions where the first picture has apparently been applicable . in the present paper, experiment and analysis are given that show under what conditions the continuous ignition mechanism provides the appropriate model and also how the two models are related . to differentiate the two mechanisms an experiment was set up to study flame stabilization in high-velocity boundary layers over a wall heated in the form of a step function . with a turbulent boundary layer and a wall temperature above 1,700f., the characteristic time was found to be a systematic and reproducible variable . these observations led to the conclusion that a continuous ignition mechanism governs stabilization in heated turbulent boundary layers . a rational explanation is made for the transition from the low-speed mechanism known to be applicable in unheated turbulent boundary layers and heated laminar boundary layers to the ignition mechanism applicable in heated turbulent boundary layers .

as a further verification of the continuous ignition mechanism an apparent ignition energy was found . the logarithm of the heat added at the lower stability limit was found to be a linear function of the reciprocal of the limiting wall temperature . the activation energy derived from this arrhenius type of relation agreed reasonably well with the estimated value for the fuel used .

.I 1269

.T

a study of supersonic combustion .

.A

gross,r.a. and chinitz,w.

.B

j. ae. scs. 1960, 517.

.W

a study of supersonic combustion .

steady, stable, plain, and oblique detonation waves were created in a high-temperature, steady flow supersonic tunnel .

ignition conditions and properties across the wave were

measured . the local-wave fluid-dynamic properties agree well

with detonation theory . experimental data are presented in

detail and compared with other studies and theory .

experimental behavior of these detonations and their possible utility

are discussed .

.I 1270

.T

supersonic inlet dynamics .

.A

fraiser,h.r.

.B

j. ae. scs. 1960, 429.

.W

supersonic inlet dynamics .

an approximation of the differential equation for compressible

duct flow is presented . the equation is linear and of the second

order . the duct transfer function and response characteristics

are obtained by applying small-perturbation theory to the

differential equation . the resulting equations describe duct

natural frequency as a function of duct areas and volumes, and

damping ratio as a function of the slope of the steady-state mass

flow, pressure-recovery curve .

the calcualted response agrees, to a first approximation, with

measured response as obtained from tests of a fixed-geometry,

sugar-scoop inlet model with hypass for matching airflows .

testing was done in the 10 x 10 and 8 x 6 ft. supersonic tunnels

at nasa lewis flight propulsion laboratory . further

agreement was obtained during flight tests of the f8u-3 airplane .

.I 1271

.T

theory of supersonic propeller aerodynamics .

.A

ordway,d.e. and hale,r.w.

.B

j. ae. scs. 1960, 437.

.W

theory of supersonic propeller aerodynamics .

a supersonic propeller with blades attached to an infinite

cylinder as a hub is studied . the forward speed may be subsonic, but

the relative speed at each section is supersonic . the lightly

loaded blades are represented by a surface distribution of

appropriate /modified/ sources in a fashion similar to ordinary

supersonic thin-wing theory . these sources are found by

approximating the exact potential for a constant-strength

compressible source traveling along a helical path . the usual

relationship between the source strength and boundary condition is

found,. and subsequently the source distribution is given, to the

appropriate order, in terms of the blade geometry .

tip effects are considered by extending the theory of evvard

and krasilshchikova . the present investigation, however, is

restricted to those planforms for which no vortex sheet appears

off the tip . for points in the tip region, the potential is obtained

through the appropriate distribution of /modified/ sources in the
upwash region off the tip . by transforming to a curvilinear,
nonorthogonal coordinate system coincident with the modified mach
lines described by the infinities of the potential, an integral
equation for the required source distribution in the upwash region is
derived . without having to solve this equation, it is shown that
the potential for a point in the tip region can be obtained in terms
of an integration of known source distributions over the blade
surface only .

the case of a twisted flat plate of particular planform is treated,
and a sample calculation is made of the pressure distribution at
selected radial positions within the noncommunicating portion
of the blade, as well as over the entire tip region .

though this analysis is carried out explicitly for the supersonic
propeller, it could also be extended to calculate various rotary
derivatives for highspeed flight vehicles .

.I 1272

.T

oscillatory aerodynamic coefficients for a unified
supersonic hypersonic strip theory .

.A

rodden,w.p. and revell,j.d.

.B

j. ae. scs. 1960, 451.

.W

oscillatory aerodynamic coefficients for a unified
supersonic hypersonic strip theory .

this investigation presents a derivation of the oscillatory aerodynamic coefficients for wings with supersonic leading edges from the second-order, nonlinear, unsteady, supersonic flow theory of van dyke . the theory is considered applicable throughout the supersonic-hypersonic regime at mach numbers normal to the leading edge and reduced frequencies for which . the coefficients are modified for sweep, and a finite-span correction is suggested to increase the accuracy of strip-theory flutter analyses . the limiting values of the coefficients in steady flow are also discussed .

.I 1273

.T

magnetohydrodynamic effects on the formation of couette, flow .

.A

tao,l.n.

.B

j. ae. scs. 1960, 334.

.W

magnetohydrodynamic effects on the formation of couette, flow .

this paper is concerned with the problem of the formation of couette flow--i.e., the problem of how the velocity profile varies with the time tending asymptotically to that of the steady flow of an electrically conducting viscous fluid in the presence of a magnetic field . the governing equations and boundary conditions are established and discussed . the cases of both vanishing and

nonvanishing mean induced electric field strengths are solved in
terms of complimentary error functions as well as some
elementary functions . it is shown that the solutions are reducible to
that of the steady case as the time approaches infinity, and to that
of the nonmagnetic field as the hartmann number becomes zero .
some numerical calculations are given . the results indicate
that in the presence of a magnetic field the flow rate is reduced
depending on the magnitude of the hartmann number, and that
the magnetic field /assists/ the flow to reach its steady
condition .

.I 1274

.T

real gas effects in flow over blunt bodies at hypersonic speeds .

.A

nagamatsu, h.t., geiger, r.e. and sheer, r.e.

.B

j. aero. sc. april, 1960. p241 .

.W

real gas effects in flow over blunt bodies at hypersonic speeds .
a hypersonic shock tunnel has been developed to investigate the
aerodynamic characteristics of flow over bodies at conditions comparable to
those encountered by ballistic missiles and satellites re-entering the
atmosphere . some results for a shock velocity of over 50,000 ft. per
sec. in the shock tube portion of the facility are presented . static
pressure investigations were made in the nozzle to determine the flow
condition and the expansion process .
the results of the investigation of representative blunt bodies at

hypersonic mach numbers and nozzle stagnation temperatures up to
approximately 6,000degreek. are presented . these include body pressure
distributions, shock-wave shapes, detachment distances, and photographs
of the luminous gas region in the shock layer . it is seen that the
shock detachment distance is smaller at higher stagnation temperatures
due to the real gas effects . for the hemisphere the pressure
distribution was less than that predicted by the modified newtonian theory for
all stagnation temperatures . for a 50degree cone-hemisphere the
pressure distribution and the shock detachment distance were appreciably
affected by the real gas effects .
the observed shock-wave shape and the approximate boundary layer on a
flat plate are compared with the analytical prediction . some
preliminary results for the detached shock wave produced by a blunt two-
dimensional body in a low density flow at a mach number of 19.6 are presented
.I 1275
.T
flow about an unsteadily rotating disc .
.A
sparrow,e.m. and gregg,j.l.
.B
j. ae. scs.1960,252.
.W
flow about an unsteadily rotating disc .
an analysis is made of the unsteady laminar flow about a
rotating disc whose angular velocity may vary with time .
the deviation of the actual instantaneous state of the flow from
the quasi-steady state (instantaneous steady state) is

determined . from this, a simplified criterion, is

derived to define the conditions under which the flow can be

considered as quasi-steady for the purposes of shear stress and

torque computations . since a turbulent flow responds more

rapidly than a laminar flow, the quasi-steady criterion found here

should also serve for the turbulent situation .

.I 1276

.T

a three-dimensional linearized analysis of the forces

exerted on a rigid wing by a shock wave .

.A

ehlers,f.e. and shoemaker,e.m.

.B

j. ae. scs. 1960, 257.

.W

a three-dimensional linearized analysis of the forces

exerted on a rigid wing by a shock wave .

the pressure distribution on a moving flat plate induced by an

acoustic shock front striking the edge of the plate obliquely has

been found in terms of the two-dimensional solution of the authors

.I 1277

.T

a study of vortex cancellation .

.A

schaffer,a.

.B

j. ae. scs. 1960, 193.

.W

a study of vortex cancellation .

the cancellation of a vortex by means of another concentric vortex of equal strength but opposite spin is investigated . when such a cancellation occurs, there is a recovery of static pressure . the vortices are generated by means of two three-dimensional airfoils cantilevered from the duct wall, one being situated in the wake of the other . the airfoils have opposite effective angles of attack and therefore have trailing vortices of opposite spin, as required .

it is demonstrated experimentally that there exists an optimum angle of attack for the second airfoil which cancels the vortex from the first airfoil and restores uniform flow downstream of the two airfoils . a theoretical solution of this optimum angle of attack is presented, and it is found to depend upon the angle of attack of the first airfoil and upon the geometrical properties of the wings . the pressure recovery accompanying the vortex cancellation is also studied . theoretical considerations, based on the model of a vortex filament in the center of a circular tube show that a maximum of 62 per cent of the static pressure drop across the first airfoil can be recovered . this maximum is imposed, irrespective of skin friction and separation losses, by the irreversibility associated with establishing a vortex field . experimental pressure recoveries of 50 per cent are realized .

perhaps the primary value of the present study is the opportunity it provides to verify certain of the fundamental concepts of fluid mechanics which are brought into play when the trailing vortex system of a lifting wing is cancelled by a second wing .

.I 1278

.T

transition in a separated laminar boundary layer .

.A

lochtenberg,b.h.

.B

j.ae.scs.1960, 92.

.W

transition in a separated laminar boundary layer .

transition to turbulence was studied in a separated laminar

boundary layer on a flat plate 24 in. long and thick . steps

with a height of to were provided at a distance of 4 to

transition was observed through a hot-wire anemometer . the

author concludes that transition was always initiated by

tollmien-schlichting waves . two types of transition were observed . in one

type, bursts suddenly appeared in the wavy flow . the other type

consists of amplification, distortion, and breaking up of the waves .

which type of transition occurs depends on the value of the

following parameter .. boundary-layer displacement thickness times

step height times free-stream velocity squared divided by

kinematic velocity squared . the burst type has been observed for

values of this parameter larger than 4.2 x 10 . the separated

laminar boundary layer becomes unstable and develops waves when

the critical reynolds number based on boundary-layer

displacement thickness at the step location exceeds a value of 350 . some

conclusions on the development of separation bubbles on air foils are drawn from the present studies .

.I 1279

.T

sublimation in a hypersonic environment .

.A

scala,s.m.

.B

j. ae. scs. 1960, 1.

.W

sublimation in a hypersonic environment .

a priori knowledge of the response of materials subjected to a severe aerothermal environment is essential in the space age . the successful design of space and re-entry vehicles demands that the fundamental problem of the interaction between a material and dissociated air be properly formulated and solved . in this paper, the problem of sublimation in a hypersonic environment is considered .

in this study of hypersonic ablation, the pertinent conservation equations are derived and the simultaneous processes of diffusion, convection, and thermal exchange are analyzed for the vaporization of a refractory material which is subjected to the environmental conditions encountered during hypersonic reentry .

for simplicity, only the forward stagnation point of an axially symmetric body is treated . it is shown that the quantity, called the effective heat of vaporization, which includes all heat absorbing or heat blocking effects, is an increasing function of flight speed, independent of body size, except where

nonequilibrium vaporization effects or radiative effects appear .

.I 1280

.T

wings with minimum drag due to lift in supersonic flow .

.A

ginzel,i. and multhopp,h.

.B

j. ae. scs. 1960, 13.

.W

wings with minimum drag due to lift in supersonic flow .

it has been shown by r. t. jones that, in order to produce minimum drag, the given lift must be distributed over the wing surface in such a way that the sum of the downwash induced by this distribution and the downwash induced in reversed flow is constant over the wing surface . this combined downwash can be expressed by an integral which contains the load as a function of the spanwise and chordwise coordinate . the problem of finding the appropriate load distribution is thus reduced to the problem of finding the solution of a rather cumbersome integral equation .

the severe spanwise singularity of the kernel function is handled most easily, as in corresponding subsonic problems, by an approximate integration over interpolation polynomials . the chordwise load distribution is represented by a limited series development in legendre polynomials . the sigularity of the kernel function along the mach lines through any pivotal point can be avoided by a similar legendre development of the

combined induced downwash which is constant . the integral

equation is thus converted into a system of linear equations for the

unknown coefficients of the legendre functions of the load

distribution at a limited number of spanwise stations .

practical calculations are carried out on an electronic

computer . the solutions yield the optimum load distribution and the

local incidence (twist, camber, etc.) necessary to realize this

distribution . for many wing plan forms, considerable gains

over a plane wing appear possible .

.I 1281

.T

turbulent heat transfer on blunt-nosed bodies in two-dimensional

and general three-dimensional hypersonic flow .

.A

vaglio-laurin,r.

.B

j. ae. scs. 1960, 27.

.W

turbulent heat transfer on blunt-nosed bodies in two-dimensional

and general three-dimensional hypersonic flow .

recent results obtained for three-dimensional laminar

boundary layers are extended to the turbulent case . it is shown that

in the presence of highly cooled surfaces and of moderate mach

numbers of the outer stream, the crossflow and the pertaining

reynolds stresses in a general three-dimensional turbulent

boundary layer are negligible even for large transverse pressure

gradients . a correlation due to mager between two-
dimensional compressible and incompressible turbulent boundary
layers is extended to the problem in question . from a study of
the transformation and of its implications, a rapid method for
the analysis of the boundary-layer flow under the subject
conditions is established . in the absence of general three-
dimensional data, a comparison with experiments and with the
predictions of other known analyses is carried out for several
axisymmetric configurations,. the results of the method presented
here exhibit good agreement with the data . the range of
validity of the cold wall approximation for general three-dimensional
problems is estimated qualitatively on the basis of recent
measurements in laminar flow, the argument being that, for either
zero or favorable streamwise pressure gradients, smaller
three-dimensional effects are to be expected in a turbulent boundary
layer, as compared to a laminar layer .

.I 1282

.T

compressible flat-plate boundary-layer flow with an
applied magnetic field .

.A

bush,w.b.

.B

j. ae. scs. 1960, 49.

.W

compressible flat-plate boundary-layer flow with an
applied magnetic field .

the laminar boundary-layer equations are formulated and solved for a flat plate in high-speed compressible air flow where equilibrium dissociation and ionization are assumed and where there is an applied magnetic field having its component normal to the plate proportional to . the skin-friction and heat-transfer characteristics are determined for free-stream velocities of up to 17,500 meters sec. and magnetic fields of up to about

the results show that the skin friction and heat transfer at a given free-stream velocity decrease with increasing magnetic field strength, and the percentage reduction is constant along the length of the plate . they also exhibit the same hysteresis behavior as was first found in the case of magnetoaerodynamic couette flow,. however, for the flat plate the hysteresis effect disappears at a higher mach number . furthermore, it was found that the reduction in heat transfer with increasing field strength is opposite in behavior from that for couette flow .

.I 1283

.T

on shearing flow between porous coaxial cylinders .

.A

dunwoody,n.t.

.B

j. ae. scs. 29, 1962, 494.

.W

on shearing flow between porous coaxial cylinders .

the flow between concentric porous cylinders in relative axial motion with a pressure gradient is considered . the analysis is

restricted by the assumption that the velocity distribution is a
function of the radial coordinate only so that there is no net
injection or withdrawal of fluid at any station . this assumption
reduces the problem to a soluble system of ordinary differential
equations . an associated heat-transfer problem is also discussed
briefly .

.I 1284

.T

the transition to tubulence in a boundary layer on
a blunt cone in supersonic flow .

.A

rogers,r.h.

.B

j. ae. scs. 29, 1962, 501.

.W

the transition to tubulence in a boundary layer on
a blunt cone in supersonic flow .

experiments were made with a series of cones, each having an
included angle of 15 degrees, and having different tip radii (sharp
to blunt nose) . the cones were tested in streams, undisturbed
mach numbers of 3.12 and 3.81, and the position of transition to
turbulence was observed with a shadowgraph technique . for each
mach number the distance to transition (distance downstream from
the tip of the cone) increased with increase in tip radius, reached
a maximum at a certain tip radius, and then decreased with
increase in tip radius . a study indicates that a reynolds number
based on the momentum thickness, instead of the length from tip to

transition, is a more appropriate parameter for correlating the data .

the research scientist active in this field probably would derive

the most benefit from the original paper .

.I 1285

.T

experiments at hypersonic speeds on circular cones

at incidence .

.A

peckham,d.h.

.B

rae tn.aero.2863, 1963.

.W

experiments at hypersonic speeds on circular cones

at incidence .

pressure distribution measurements

on five circular cones with total

apex-angles ranging from 25 to 45 degrees

are described . the tests covered

a range of angles of incidence from 0 to

and 8.60 . the extent to which various

analytical and empirical theories

predict the measured pressures is assessed .

.I 1286

.T

equilibrium real-gas performance charts for a shypersonic

shock-tube wind-tunnel employing nitrogen .

.A

bernstein,l.

.B

arc cp.633, 1961.

.W

equilibrium real-gas performance charts for a shypersonic

shock-tube wind-tunnel employing nitrogen .

charts are presented covering

a wide range of reflected-shock

wind-tunnel operating conditions, using

nitrogen as the working gas . a

statistical-mechanical model of the gas

is assumed which takes account of

molecular vibration, electronic excitation

and dissociation . the gas is

assumed to be constantly in

equilibrium that is, the reaction rates are

taken to be infinitely fast . the equations

of motion are solved with the

aid of a digital computer, previously reported

results for the state of the

shock-processed gas in the shock-tube being used .

.I 1287

.T

progress report on an experiment on the effect of surface

flexibility on the stability of laminar flow .

.A

gregory,n. and love,e.m.

.B

arc cp.602, 1961.

.W

progress report on an experiment on the effect of surface

flexibility on the stability of laminar flow .

this paper describes the flexible

surfaces whose properties have

been examined and which have been tested on

an aerofoil in a wind tunnel .

the experiment has been rather inconclusive

as no drag reductions have been

found in turbulent flow, whilst the only

rearward movements of transition

occurred in conditions where the alteration

has been inhibited by the onset

of laminar separation . the limitations of

the experiment are discussed

carefully in order to clarify the next steps

which are to be taken with

more flexible surfaces with less damping .

.I 1288

.T

analysis of the fluid mechanics of secondary injection

for thrust vector control .

.A

broadwell,j.e.

.B

aiaa jnl. 1963, 1067.

.W

analysis of the fluid mechanics of secondary injection

for thrust vector control .

an analysis is made of the interaction of an injected

gas or liquid with a supersonic stream,

and the force induced on an adjacent wall is predicted .

the study deals only with the

freestream-injectant interaction,. the modifications to the

flow introduced by the boundary layer

are not considered . in the case of liquids, it is shown

that the momentum deficit of the

injectant relative to the freestream may play a larger part

in producing the side force than the

volume generation by vaporization and reaction . the

analytical results are compared with

those obtained from experiments in a wind tunnel and in nozzles .

.I 1289

.T

numerical technique to lifting surface theory for calculation

of unsteady aerodynamic forces due to continuous sinusoidal

gusts on several wing planforms at sobsonic speeds .

.A

murrow,h.n., pratt,k.g. and drischler,j.a.

.B

nasa tn.d1501, 1963.

.W

numerical technique to lifting surface theory for calculation of unsteady aerodynamic forces due to continuous sinusoidal gusts on several wing planforms at sobsonic speeds .

a numerical lifting-surface method has been used to calculate direct gust forces and moments on wings of several planforms . the gust velocities are continuous and vary sinusoidally in the stream direction and are also uniform across the wing span . the procedure has the advantage of rapid machine calculation and includes the effects of wing planform, nonsteady subsonic flow, and induced flow effects . the method provides for calculation of gust forces on a basis consistent with that for the calculation of forces due to motion and deformation . the results include the in-phase and quadrature components of the following quantities .. (a) spanwise distribution of section lift coefficient, (b) total lift coefficient, and (c) total pitching-moment coefficient . in addition, generalized gust forces on approximate fundamental cantilever bending modes (parabolic) are also included . results have been obtained for 60 and 75 delta wings, ratio 11.60, and an unswept wing of aspect

ratio 6.00 . conditions for which

calculations were made include two mach numbers

reduced-frequency range of 0 to 1.0 . the

direct gust forces and moments are in

forms suitable to be inserted in equations

of motion used in the calculation of

the dynamic responses of flexible lifting

vehicles to random turbulence and to be

compared with results from other methods .

.I 1290

.T

measured and calculated subsonic and transonic flutter

characteristics of a 45 sweptback wing planform in

air and in freon-12 in the langley transonic dynamics

tunnel .

.A

yates,e.c., land,n.s. and foughner,j.t.

.B

nasa tn.d1616, 1963.

.W

measured and calculated subsonic and transonic flutter

characteristics of a 45 sweptback wing planform in

air and in freon-12 in the langley transonic dynamics

tunnel .

 in order to investigate the

reliability of flutter data measured

in the langley transonic dynamics tunnel, an experimental and theoretical subsonic and transonic flutter study has been conducted in air and in freon-12 in this facility . the wing planform employed had an aspect ratio of 4.0, a taper ratio of 0.6, and 45 of quarter-chord sweepback . a sting-mounted full-span model was tested in addition to three sizes of wall-mounted semispan models . a wide range of mass ratio was covered by the tests in air and by flutter calculations made by the modified strip-analysis method of naca research memorandum l57l10 . a limited amount of data was obtained in freon-12 .

results of the tests in air and in freon-12 are in good agreement with the flutter calculations at all mach numbers . the test data compare favorably with previously published transonic flutter data for the same wing planform . the results indicate that flutter characteristics obtained in freon-12 may be interpreted directly as equivalent flutter data in air at the same mass ratio and mach number .

.I 1291

.T

atmosphere entries with spacecraft lift-drag ratios

modulated to limit decelerations .

.A

levy,l.l.

.B

nasa tn.d1427, 1962.

.W

atmosphere entries with spacecraft lift-drag ratios

modulated to limit decelerations .

an analysis has been made of

atmosphere entries for which the

spacecraft lift-drag ratios were

modulated to limit the maximum

deceleration . the parts of the drag polars

used during modulation were from

maximum lift coefficient to minimum

drag coefficient . five drag polars

of different shapes were assumed for

the spacecraft . the entries covered

wide ranges of initial velocity,

initial flight-path angle, initial

and maximum lift-drag ratio .

two-dimensional trajectory calculations were

made for a nonrotating, spherical

earth with an exponential atmosphere . the results of the analysis indicate for four of the five drag polars that, relative to the maximum deceleration of an unmodulated entry at maximum lift-drag ratio, the greatest reduction in maximum deceleration obtainable by modulation depends upon a single parameter . this parameter is the ratio of the value of the aerodynamic resultant-force coefficient at minimum drag coefficient to the value at maximum lift coefficient . thus, the reduction in maximum deceleration is independent of initial velocity, initial flight-path angle, initial maximum lift-drag ratio, and the shape of the drag polar . for the fifth drag polar, the reduction in maximum deceleration was found to depend upon the maximum lift-drag ratio . also, relative to the depth of a given deceleration-limited corridor, the greatest increase in corridor depth obtainable by modulation (for four of the five drag polars) depends upon the same ratio of aerodynamic resultant-force coefficients . the fractional

increase in corridor depth can be

expressed as an empirically determined

analytical function of this ratio .

.I 1292

.T

effect of jet pluming on the static stability of cone-cylinder-flare

configurations at a mach number of 9. 65 .

.A

hinson,w.f. and falanga,r.a.

.B

nasa tn.d1352, 1962.

.W

effect of jet pluming on the static stability of cone-cylinder-flare

configurations at a mach number of 9. 65 .

the effects of jet pluming on

normal force and pitching moment of

have been measured at a free-stream

mach number of 9.65 with reynolds

numbers based on model length of 500,000

to 600,000 . geometric variables

included nose bluntness, flare half-angle,

and nozzle geometry and exit displacement .

two test nozzles with design

mach numbers of 3.74 and 4.60 were operated

with compressed air to

simulate the initial jet-boundary shape of a particular solid-propellant

rocket motor operating between altitudes

of 165,000 and 215,000 feet .

the ratio of the jet pressure to free-stream

static pressure varied from

a jet-off condition to approximately 1,300

for the nozzle with design mach

number of 3.74, and from a jet-off condition

to approximately 280 for the

nozzle with design mach number of 4.60 .

the angle-of-attack range was

from 0 to approximately 6 .

the results indicate that as the

jet-pressure ratio was increased

the size of the jet plume increased,

and as a result the model static

stability was decreased . increasing

the angle of attack resulted in a

reduction in static instability during

the jet-on condition . increasing

nose bluntness resulted in a more forward

movement of the center of

pressure when jet-plume interference was not

present and a rearward movement

in the center of pressure when jet

interference was present . increasing

the nozzle-area expansion ratio and

displacing the nozzle exit downstream

of the flare base resulted in a more

rearward location of the center of

pressure .

.I 1293

.T

design of stiffened cylinders in axial compression .

.A

hedgepeth,j.m.

.B

nasa tn.d1510, 1962.

.W

design of stiffened cylinders in axial compression .

the problem of optimum design

of axially compressed cylinders

stiffened by rings and stringers is

discussed . particular

attention is devoted to configurations

suitable for large launch

vehicles . consideration is given to

the analytical techniques for

determining strength as well as

the procedures for optimization .

.I 1294

.T

non-linear shallow shell analysis by the matrix force

method .

.A

lansing,w., jones,i.w. and ratner,p.

.B

nasa tn.d1510, 1962, 753.

.W

non-linear shallow shell analysis by the matrix force

method .

the matrix force method of redundant

structure analysis is currently

being extended by various users to cover

a number of non-linear problems .

one of these is the non-linear analysis

of heated cambered wings, such as

might be used in advanced flight vehicles .

in this case the approach used

by the present authors is equally applicable

to shallow shells, the

formulation of the strain-displacement and

equilibrium relations being a

finite element equivalent to that used

by marguerre . the solution is

obtained by a combined iteration and step

by step procedure utilizing a

tangent flexibility matrix . divergence

in the calculations indicates

that the range of stable configurations

has been exceeded . cambered

plates subjected to several loadings are

given as examples,. for one, an

exact solution is available for comparison .

it is believed that the basic concepts

involved in this shallow

shell analysis can be extended to apply

to other, more general shell

instability problems, and that useful

solutions to the latter are

probably within the capability of present day digital computers .

.I 1295

.T

recent advances in nonequilibrium dissociating gasdynamics .

.A

li,t.y.

.B

ars jnl. 1961.

.W

recent advances in nonequilibrium dissociating gasdynamics .

the purpose of this paper is to review some recent

advances in the study of gasdynamic problems including

effects of chemical reactions . to provide a background for

the study the general concepts shall be outlined briefly . the

discussions of the recent developments are restricted to

inviscid flow problems only, neglecting viscosity, heat

conduction and diffusion . particular attention is directed to recent

advances in analyses of nonequilibrium dissociating gas flows . in the hypersonic flight regime, high stagnation

enthalpies sufficient to cause dissociation are realized . when

the time to reach equilibrium is comparable with the time it

takes for a fluid particle to pass through the flow, then there

exist regions of the flow field where nonequilibrium states are

encountered . a brief survey of both the linear and the

nonlinear methods of treatment of these nonequilibrium flows,

including some new developments that have not appeared

elsewhere, will be presented .

.I 1296

.T

non-equilibrium expansions of air with coupled chemical

reactions .

.A

eschenroeder,a.q., boyer,d. and hall,j.g.

.B

phys. fluids, 1962.

.W

non-equilibrium expansions of air with coupled chemical

reactions .

analysis and solutions of the streamtube gas

dynamics involving coupled chemical rate equations

are carried out . results are presented for airflows

along the surface of blunt bodies and through

hypersonic nozzles . speeds and altitudes corresponding

to re-entry were selected to obtain initial

conditions for the external flow calculations . conditions appropriate to hypersonic tunnel testing

were chosen for the nozzle flow calculations . composition

histories are shown for a kinetic mechanism

including 6 species and 14 reactions . gas-dynamic effects

of nonequilibrium processes qualitatively

resemble those reported earlier . however, the freezing

process is complicated by the coupling of the

nitric oxide shuffle reactions with the dissociation-recombination

reactions . in many cases of

hypersonic nozzle flows where the energy in nitrogen dissociation

is significant, the fast shuffle reactions

prevent nitrogen-atom freezing which would otherwise occur

if three-body recombination were the only

process operating . nitric oxide concentrations

undershoot the equilibrium values if the ratio of

nitric oxide to oxygen molecule concentrations

exceeds unity in the freezing region . this depletion

of nitric oxide leads to nitrogen-atom freezing .

.I 1297

.T

ionization nonequilibrium in expanding flows .

.A

eschenroeder,a.q.

.B

ars jnl. 1962, 32.

.W

ionization nonequilibrium in expanding flows .

approximate methods are developed for solving

the electron-ion kinetic equations in expanding

quasi one-dimensional airflows . results are obtained

for inviscid nozzle flows at conditions

appropriate to shock tunnel testing and are compared

with exact numerical solutions . effects on rf

transmission and d-c conductivity are examined .

since two-body deionization never fully freezes

in the flows considered here, the assumption of sudden

freezing gives an upper bound on the residual

ionization at large area ratios . the use of an

asymptotic form of solution with the freezing

criterion provides an improved estimate for such cases .

ionization nonequilibrium is also

considered for the plasma sheath associated with blunt

hypersonic bodies flying at high altitudes .

the influence of atomic ions is examined for typical

re-entry conditions .

.I 1298

.T

theory of radiation from luminous shock waves in nitrogen .

.A

hammerling,p., teare,j.d. and kivel,b.

.B

phys. fluids, 1959, 2, 422.

.W

theory of radiation from luminous shock waves in nitrogen .

the physical properties behind a normal shock in
nitrogen are calculated as a function of time .
these include the variation of temperature, composition,
ionization, and the intensity of radiation
from the n first negative band system . this calculation
incorporates a rate equation for the
dissociation of nitrogen, the conservation laws, an equation
describing vibrational relaxation, and a
method of coupling the vibrational relaxation with the
dissociation rate . the n radiation is
computed assuming excitation of the radiating state by
collision with vibrationally excited nitrogen
molecules . a particular case is considered for which
experimental data are available, and regions
sensitive to particular rates are indicated .

.I 1299

.T

hypersonic viscous shock layer .

.A

scala,s.m.

.B

ars jnl. 1959, 29, 520.

.W

hypersonic viscous shock layer .

a decade ago tsien (1) (as well as others) and, more
recently, adams and probstein (2) have attempted to
define the different regimes of gaseous interactions during high

altitude flight . in this note some results are presented which

are pertinent to the flight of hypersonic lifting vehicles

composed of axially symmetric and two-dimensional elements,.

sec. fig. 1 .

.I 1300

.T

some effects of bluntness on boundary layer transition

and heat transfer at supersonic speeds .

.A

moeckel,w.e.

.B

naca r.1312, 1957.

.W

some effects of bluntness on boundary layer transition

and heat transfer at supersonic speeds .

large downstream movements of transition observed when the

leading edge of a hollow cylinder or a flat plate is slightly

blunted are explained in terms of the reduction in reynolds

number at the outer edge of the boundary layer due to the

detached shock wave . the magnitude of this reduction is

computed for cones and wedges for mach numbers to 20 .

concurrent changes in outer-edge mach number and temperature occur

in the direction that would increase the stability of the laminar

boundary layer .

the hypothesis is made that transition reynolds number is

substantially unchanged when a sharp leading edge or tip is

blunted . this hypothesis leads to the conclusion that the

downstream movement of transition is inversely proportional to
the ratio of surface reynolds number with blunted tip or
leading edge to surface reynolds number with sharp tip or leading
edge . this conclusion is in good agreement with the
hollow-cylinder result at mach 3.1 .
  application of this hypothesis to other mach numbers yields
the result that blunting the tip of a slender cone or the leading
edge of a thin wedge should produce downstream movements of
transition by factors ranging from 2 at mach 3.0 to 30 at mach
the possible reduction in over-all heat-transfer rate and friction
drag for aircraft flying at high supersonic speeds .
  mach number profiles near the surfaces of blunted cones and
wedges are computed for an assumed shape of the detached
shock wave at flight mach numbers to 20 . the dissipation and
stability of these profiles are discussed, and a method is
described for estimating the amount of blunting required to produce
the maximum possible downstream movement of transition .
.I 1301
.T
compressible boundary layers on bodies of revolution .
.A
mangler,k.w.
.B
arc r.9740, 1946.
.W
compressible boundary layers on bodies of revolution .
  in a former paper (1) it has been shown that the

behaviour of the laminar boundary layer on a body of revolution can

be described mathematically by the same equations which are also

applied to the processes in the laminar boundary layer in the

two-dimensional flow along a body contour, the form of which is

determined by the shape of the body of revolution . a simple

relation exists between the two-dimensional boundary layers and

the axially symmetrical ones . the flow had been assumed to be

incompressible . in this report it shall be shown that this

relation is still valid when the compressibility is taken into

consideration . the distribution of velocity as well as that of

temperature in the laminar boundary layer of a body of revolution

can be calculated by solving the corresponding problem for the

two-dimensional flow around a suitable contour . the method is

made clear by the example of the supersonic flow towards a cone

tip,. this example has already been treated by another method

by hantzsche and wendt (2) .

.I 1302

.T

the development of the boundary layer in supersonic

shear flow .

.A

rogers,r.h.

.B

rae tn.aero.2738, 1961.

.W

the development of the boundary layer in supersonic

shear flow .

the development of the boundary layer in a velocity shear layer is discussed for two-dimensional flow and for axisymmetric flow of both compressible and incompressible fluids . it is shown that the solutions obtained by li and glauert for the two-dimensional flow of an incompressible fluid are applicable in the more general case after suitable transformations of coordinates have been made . new definitions are shown to be necessary, and are given, for the displacement and momentum thicknesses of such a boundary layer . reynolds numbers based on these thicknesses are given, and it is shown that any phenomenon which occurs at a constant value of such a reynolds number will occur at a point which, as the length scale of the flow increases, first moves down-stream and then moves slightly upstream . this is shown to be in qualitative agreement with experimental results on a blunt cone in a supersonic flow . a quantitative comparison of the theoretical and experimental values of displacement and momentum thicknesses is

attempted, and no disagreement is

obvious,. unfortunately the accuracy of the

experiments so far available is

insufficient to give positive confirmation

of the theory of this note .

.I 1303

.T

air pressure on a cone moving at high speeds .

.A

taylor,g.i. and maccoll,j.w.

.B

proc. roy. soc. a, 139, 1933, 278.

.W

air pressure on a cone moving at high speeds .

the cone is considered to be

moving at a velocity higher than that

of sound, so that there is in front

of it a shock wave, moving with the

same speed as the cone itself . in

the first part of the paper, the case is

investigated mathematically where

the flow is irrotational, and the

pressure, velocity and density of the air

stream are each constant over the

surfaces of cones coaxial with the

moving solid cone . the complete

solution is obtained in numerical

form, for cones of semi-vertical angle

of the paper, the results are compared

with experiment, both in respect of

pressure distribution as measured in a

wind tunnel, and also (for the 30

cone) by comparison with photographs

of bullets in flight . in the latter case

the theory should only be applicable

if the speed is 1.46 or more times the

velocity of sound, and it is in fact

found in the photographs, that the nature

of the wave alters at about this

velocity . the exact solution found, is

compared with an approximation

given recently by v. karman and moore .

this should be valid for thin

spindle-shaped bodies, and does in fact

agree well in the case of the cone

of 10 semi-vertical angle, but diverges

increasingly from the truth as

the angle is increased .

.I 1304

.T

newtonian flow over a surface .

.A

giraud,j.p.

.B

colston symposium on hypersonic flow, bristol, 1959.

.W

newtonian flow over a surface .

a general method is presented for

the study of a three-dimensional

hypersonic flow about a body of arbitrary shape when .

the manner of constructing a double

asymptotic development in

and is shown . formulae are given

which enable the first three terms of

this development to be obtained while neglecting .

the theory is then applied to the case of

a body of circular-cone shape . the

pressure is given as a triple development

in accordance with the preceding

parameters and the angle of attack,. this

development neglects . a. ferri's

vortical layer is brought into evidence .

a second application is devoted to

calculation of the total forces acting

upon bodies of revolution at angles of

incidence, while neglecting .

general formulae are

established for the coefficients of axial

force, normal force and moments .

the formulae are developed according

to the powers of incidence, the first

terms of each formula being of very simple form .

.I 1305

.T

a proposed programme of wind tunnel tests at hypersonic

speeds to investigate the lifting properties of geometrically

slender shapes .

.A

peckham,d.h.

.B

rae tn.aero.2730.

.W

a proposed programme of wind tunnel tests at hypersonic

speeds to investigate the lifting properties of geometrically

slender shapes .

a programme of tests at hypersonic

speeds on slender bodies is

described, which has the aim of

investigating how lift is generated, and

the compromises that may be enforced

by aerodynamic heating . the programme

is based on models of simple geometric

shape, from which lifting

configurations will later be built up .

.I 1306

.T

experiments on circular cones at yaw in supersonic

flow .

.A

holt,m. and blackie,j.

.B

j. ae. scs. 23, 1956, 931.

.W

experiments on circular cones at yaw in supersonic

flow .

pressure measurements made in the fort halstead

supersonic tunnel on two circular cones, of semiapex angles 15 and

coefficients are compared with corresponding values calculated

by theoretical methods, and the relative merits of these methods

are then discussed .

.I 1307

.T

laminar heat-transfer and pressure measurements at

a mach number of 6 on sharp and blunt 15 half-angle

cones at angles of attack up to 90 .

.A

conti,r.j.

.B

nasa tn.d962, 1961.

.W

laminar heat-transfer and pressure measurements at

a mach number of 6 on sharp and blunt 15 half-angle

cones at angles of attack up to 90 .

two circular conical configurations

having 15 half-angles were

tested in laminar boundary layer at a

mach number of 6 and angles of

attack up to 90 . one cone had a sharp

nose and a fineness ratio of

blunted nose with a bluntness ratio

of 0.1428 and a fineness ratio of 1.66 .

pressure measurements and

schlieren pictures of the flow showed

that near-conical flow existed up

to an angle of attack of approximately

near the base and the bow shock

wave was considerably curved .

comparison of the results with

simply applied theories showed that

on the stagnation line pressures may

be predicted by newtonian theory,

and heat transfer by local yawed-cylinder

theory based on the yaw angle

of the windward generator and the local

radius of the cone . base effects

increased the heat transfer in a region

extending forward approximately

circumferential pressure

distributions were higher than the

corresponding newtonian distribution

and a better prediction was obtained

by modifying the theory to match

the pressure at 90 from the windward

generator to that on the surface

of the cone at an angle of attack of 0 .

circumferential heat-transfer

distributions were predicted satisfactorily

up to about 60 from the

stagnation line by using lees' heat-flux

distribution based on the

newtonian pressure . the effects of nose

bluntness at large angles of

attack were very small in the region

beyond two nose radii from the

point of tangency .

.I 1308

.T

a guide to the use of the m. i. t. cone tables .

.A

roberts,r.c. and riley,j.d.

.B

j. ae. scs. 21, 1954, 336.

.W

a guide to the use of the m. i. t. cone tables .

  the second and third volumes of the m.i.t. cone tables have

been found to be unsatisfactory in two respects . they have been

criticized because of their inconvenient tabulation and because

the theory on which they are based is inadequate near the cone surface . the former is climinated by means of a coordinate

transformation . empirical evidence is presented to show that

the latter may be ignored in practice . the exact nature of

certain numerical errors in the table is also pointed out .

.I 1309

.T

hypersonic flows past a yawed circular cone and other

pointed bodies .

.A

cheng,h.k.

.B

j. fluid mech. 12, 1962, 169.

.W

hypersonic flows past a yawed circular cone and other

pointed bodies .

a detailed treatment of inviscid hypersonic

flow past a circular cone is given, for

small and moderate yaw angles, within

the framework of shock-layer theory .

the basic problem of non-uniform validity

associated with the singularity of

the entropy field is examined and a valid

first-order solution is obtained which

provides an explicit description of a thin

vortical layer at the inner edge of the

shock layer . analytic formulas for pressure

and circumferential velocity are

given consistent to the second-order

approximation including the non-linear

yaw effect .

the study of the entropy field (which

is not restricted to the hypersonic case)

also provides corrections to previous

work on the yawed cone and confirms the

validity of the linear yaw effect on pressure

field in the stone theory .

a related investigation of three-dimensional

flow fields is presented with

special reference to the flow structure near

the surface of a pointed, but

otherwise arbitrary body . the inviscid streamline

pattern on the surface is given by

the geodesics originting from the pointed

nose as a leading approximation of

shock-layer theory . associated with this

streamline pattern is a vortical sublayer

which exists generally at small as well as at

large angle of attack . at the base of

the sublayer, enthalpy and flow speed remain

essentially uniform .

.I 1310

.T

survey of inviscid hypersonic flow theory for geometrically

slender shapes .

.A

crabtree,l.f.

.B

rae tn. aero.2695, 1960.

.W

survey of inviscid hypersonic flow theory for geometrically

slender shapes .

a survey is made of existing

theories for the calculation of pressure

distributions on slender bodies at

hypersonic speeds . no account is taken

of boundary layer displacement effects

which are expected to become important

above a mach number of about 10 for a slender body .

first the breakdown of linearised

supersonic theory is demonstrated as

mach number increases above about 5, and

this is followed by a derivation of

the hypersonic similarity rule . this

section includes a description of the

piston-analogy .

next a physical interpretation

of hypersonic flow is outlined and a

simple derivation of the modified

newtonian pressure formula is given .

the equations of flow through

an oblique shock wave are simplified by

assuming a strong shock, and various
results are thereby derived . these
include the tangent-wedge and tangent-cone formulae .
this is followed by a description
of the newtonian approximation for
slender bodies, including the effect of
centrifugal forces, and the connection
with newtonian flow theory is emphasized for .
the shock-expansion method is
described in some detail for both
two-and three-dimensional bodies, and
finally some remarks are made about the
available data sheets and tables for
estimating pressures on cones and
ogive-cylinders in yaw .
the note does not claim to be
original, even in presentation . the aim
has been to prepare a reasonably complete
survey of available theory for
hypersonic flow over slender bodies, excluding
viscous and explicit real gas effects .
this will provide the background
for further work in which experimental
data will be analysed and in conjunction
with which it is hoped to produce
accurate design methods for estimating
pressures and forces on shapes intended

for sustained flight at hypersonic speeds .

.I 1311

.T

some simple solutions to the problem of predicting

boundary layer self-induced pressures .

.A

bertram,m.h. and blackstock,t.a.

.B

nasa tn.d798, 1961.

.W

some simple solutions to the problem of predicting

boundary layer self-induced pressures .

simplified theoretical approaches

are shown, based on hypersonic

similarity boundary-layer theory,

which allow reasonably accurate

estimates to be made of the surface

pressures on plates on which viscous

effects are important . the consideration

of viscous effects includes

the cases where curved surfaces,

stream pressure gradients, and

leading-edge bluntness are important factors .

.I 1312

.T

tabulated solutions of the equilibrium gas properties

behind the incidents and reflected normal shock-wave in a shock-tube .

.A

bernstein,l.

.B

i. nitrogen, ii. oxygen. arc cp.626, 1961.

.W

tabulated solutions of the equilibrium gas properties

behind the incidents and reflected normal shock-wave

in a shock-tube .

tabulated solutions are

presented for the equilibrium gas

properties behind the incident and

reflected normal shock-waves in the

shock-tube, for nitrogen and oxygen .

they cover the range of shock-wave

mach numbers up to 12 at intervals of

undisturbed gas pressure between 1 and

the thermodynamic model of the

gas used in the calculations

is described in some detail, as is the

method of solving the equations .

the limitations of the assumption of

thermodynamic equilibrium are discussed

with regard to shock-tube applications,

and the estimated accuracy of the

tables is indicated .

.I 1313

.T

on the flow in a reflected shock tunnel .

.A

holder, d.w. and schultz, d.l.

.B

arc r + m 3265, august 1960 .

.W

on the flow in a reflected shock tunnel .

the performance of a shock tunnel operated by the reflected-shock

technique is examined theoretically neglecting viscous effects and

high-temperature real-gas effects . particular attention is given to

disturbances to the flow at the nozzle entry caused by waves reflected from

the contact surface when the operating conditions depart from those for

that the first disturbance reflected from the contact surface is weak

enough to be tolerated only within a small range of primary-shock mach

number, m /e.g., 5 7 m 6 3 if the pressure at entry to the nozzle is to

remain constant to 10 per cent/ . within this range, running times much

longer than those obtained in 'straight-through' shock tunnels are

predicted, the limitation usually being imposed by the arrival of the

expansion wave originating at the diaphragm .

outside this range of mach number, the uniform-flow duration between the

arrival at the nozzle entry of the primary shock and the first

disturbance reflected from the contact surface is shown to be approximately

equal to the time between the arrival of the primary shock and the

contact surface in a 'straight-through' shock tunnel . at first sight it

appears, therefore, that the advantages of reflected-shock operation

are confined to a very narrow range of shock mach number, unless a

heated driver gas is used in order to vary the mach number for
further analysis suggests, however, that subsequent disturbances in the multiple wave reflection process between the contact surface and the end of the tube are relatively weak over a useful range of shock mach number . thus, if the flow after the arrival of the early reflected disturbances is used for test purposes, long running times seem possible in theory without severe restrictions to the shock mach number .
experiments have been made in a shock tube and a shock tunnel to provide data for comparisons with the results of the simple theory . if allowance is made for viscous effects on the motion of the contact surface, fair agreement is found for the disturbances reflected and transmitted by the contact surface, and for the arrival of the expansion wave
reflection process increases when the shock mach number is raised substantially above the 'tailored' value, and a limit to the usable flow duration may result .
a striking feature of the results is a fall of pressure at the end of the tube immediately after reflection of the primary shock . this is attributed to attenuation of the reflected shock resulting from its interaction with the boundary layer on the wall of the tube . further research is required to check this explanation, and to investigate the effects of reynolds number and of the cross-sectional shape and size of the tube . the effects of the tail and reflected head of the expansion wave originating at the main diaphragm are discussed . it is shown that the arrival of the reflected head at the nozzle entry may impose a severe limitation to the duration of uniform conditions at low shock mach number, and that the arrival of the tail may limit the flow duration at high shock mach number . unless means can be devised to

suppress the expansion wave, it is demonstrated that it is desirable to have alternative diaphragm positions in a tube required to operate over a range of shock mach number .
it is concluded that running times of order 10 milliseconds at a shock mach number of 4, falling to, perhaps, 1 millisecond at a shock mach number of 8 seem possible in a shock tunnel of reasonable size by using reflected-shock operation with unheated hydrogen driving air . because of the simplifying assumptions of the theoretical investigations, and the deficiencies of the apparatus used for the experiments, the present investigation must, however, be regarded as preliminary in character . further research is required to check and extend the findings, and topics particularly requiring investigation are listed in the paper .

.I 1314

.T

production of high temperature gases in shock tubes .

.A

resler,e.l., lin,s.c. and kantrowitz,a.

.B

j. app. phys., 23, 1852, 1390.

.W

production of high temperature gases in shock tubes .

this paper is intended to set forth aerodynamic and thermodynamic calculations which are useful in the production of strong shock waves . the experimental production of strong shock waves is discussed . comparison of the experimental shock strengths with the theoretical calcualtions is made, and finally, some preliminary results of shock tube studies in high temperature

gases (up to 18,000k) are briefly surveyed .

.I 1315

.T

performance estimates for the rae 6in . high-pressure

shock tube .

.A

woods,n.a.

.B

arc 20,440, 1958.

.W

performance estimates for the rae 6in . high-pressure

shock tube .

estimates are made of the performance

of the rae 6'' high pressure

shock tube, with various driver gases, over

a range of pressure ratios giving

shock mach numbers from 6 to 22 . the

calculations are based on a simplified

model of shock tube flow, in which the

working fluid (argon-free air) is

assumed to be always in chemical

equilibrium, and the driver gas (either

hydrogen, or the products of combustion

of a hydrogen-oxygen mixture) is

assumed to behave as an ideal gas with

constant specifiic heats .

the results are presented in
graphical form and comprise charts
normal shock waves in argon-free air
shock wave mach number and diaphragm
pressure ratio under various initial conditions, and
of the shock-induced flows, both
in the uniform-sectioned shock tube, and
when expanded in a divergent nozzle

.I 1316

.T

temperature measurements of shock-waves by spectrum-line reversal, ii a double beam method .

.A

clouston, j.g., gaydon, a.g. and hurle, i.r

.B

proc. roy. soc. a252, p143, september 1959 .

.W

temperature measurements of shock-waves by spectrum-line reversal, ii a double beam method .

the sodium-line reversal method, as previously described, using a photomultiplier and oscillograph, has been modified . two light beams are now employed, and interference filters are used in front of the photomultipliers instead of a spectrograph . in one beam the background source is viewed directly, through the shock tube, and in the other beam the background source is viewed through the shock tube by a mirror system with a neutral filter interposed to reduce its effective brightness temperature . with a suitably chosen temperature for the background, one oscillograph trace indicates absorption and the other

indicates emission of the sodium lines . it is thus possible, from the

records of a single shock, to determine the temperature history behind

the shock wave to about 20degreec . nitrogen and oxygen again show

relaxation effects near the front . temperatures in argon tend to come

low, owing to radiative disequilibrium,. excitation processes in argon

are discussed . with this system it is possible to determine

temperatures rather higher than that of the background source . some work has

also been done, with a single-beam method, using a carbon arc as

background and following reversal of the indium blue line . temperatures

up to 3600degreek have been measured in shocks through nitrogen, but

the time resolution is not so good .

.I 1317

.T

shock-tube testing time .

.A

anderson, g.

.B

j. aero. sc. vol. 26, march 1959, p184 .

.W

shock-tube testing time .

in a theoretical investigation of attenuation effects of the shock wave

the conservation of mass equation led to an explanation of the

difference between the ideal theoretical test time and the experimentally

obtained time . a numerical example is given .

.I 1318

.T

stagnation temperature measurements in a hypersonic gun tunnel using the

sodium-line reversal method .

.A

stollery, j.l.

.B

arc 22854, may 1961 .

.W

stagnation temperature measurements in a hypersonic gun tunnel using the

sodium-line reversal method .

the sodium line reversal /s.l.r./ method has recently been used to

measure transient temperatures in the 1400degreek to 3000degreek range, for

example ref.6 reports measurements of gun flash temperatures . in the

gun tunnel, stagnation temperatures in the above range can be generated

flow between a blunt body and its bow shock wave . the gas temperature

in this zone is close to the stagnation value .

.I 1319

.T

real gas effects in flow over blunt bodies at hypersonic speeds .

.A

nagamatsu, h.t., geiger r.e. and sheet, r.e.

.B

arc 21, 083, june 1959 .

.W

real gas effects in flow over blunt bodies at hypersonic speeds .

a hypersonic shock tunnel has been developed to investigate the

aerodynamic characteristics of flow over bodies at conditions comparable to

those encountered by ballistic missiles and satellites re-entering the

atmosphere . some results for a shock velocity of over 50,000 ft/sec in
the shock tube portion of the facility are presented . static pressure
investigations were made in the nozzle for different stagnation
conditions in order to determine the flow condition and the expansion
process .
the results of the investigation on representative blunt bodies at
hypersonic mach numbers and nozzle stagnation temperatures up to
approximately 6000degreek are presented . these include body pressure
distributions, shock wave shapes, detachment distances, and photographs
of the luminous gas region in the shock layer . it is seen that the
shock detachment distance is smaller at higher stagnation temperatures
owing to the real gas effects . for the hemisphere the pressure
distribution was less than that predicted by the modified newtonian theory for
all stagnation temperatures . for a 50degree cone-hemisphere the
pressure distribution and the shock wave detachment distance were
appreciably affected by the real gas effects .
the experimentally obtained shock wave shape and the approximate
boundary layer on a flat plate are correlated with the analytical
prediction . some preliminary results for the detached shock wave produced
by a blunt two-dimensional body in a low density flow at a mach number
of 19.6 are presented .
.I 1320
.T
divergence of plate airfoils of low aspect ratio at
supersonic speeds .
.A
hancock,g.

.B

j. ae. scs. 1959, 495.

.W

divergence of plate airfoils of low aspect ratio at supersonic speeds .

in part (1), as a first approach to a theoretical investigation of low aspect ratio rectangular plate wings of constant thickness, the two assumptions are made that .. (a) the spanwise form of the structural distortion is known, leaving the chordwise distortion arbitrary,. and (b) the aerodynamic forces are approximations of the supersonic linearized theory . the form of the chordwise distortion is then deduced from the differential equation representing the state of neutral equilibrium for small displacements at the critical divergence speed .

secondly, this problem is investigated using measured structural flexibility coefficients together with theoretical aerodynamic coefficients .

thirdly, the usual series solution based on the rayleigh-ritz approach is discussed, using the same assumptions as in the first method .

all the results of these methods are consistent and indicate that the transonic regime at m = 1 is the most critical for divergence .

in part (2), it is established that sweeping the leading edge of a plate airfoil of constant thickness increases its stability . for angles of sweep less than 30, the critical conditions occur when the leading edge is sonic, but for angles greater than 30 the critical conditions occur when m = 1 .

.I 1321

.T

effects of a flexible boundary on hydrodynamic stability .

.A

benjamin, t.b.

.B

j. fluid mech. vol. 9, 4, 1960. p513 .

.W

effects of a flexible boundary on hydrodynamic stability .

purpose of paper is to examine theoretically the use of coatings of elastic materials to prevent transition from laminar to turbulent flow . theory is extension to flexible boundary of the small-disturbance tollmien-schlichting stability theory and makes use of /tietjens function/ and other functions that occur in solution of orr-sommerfeld equation . it is shown how solutions for flexible wall can be obtained from solutions for rigid boundary .

outline and discussion is given first for tollmien-schlichting stability theory for rigid wall, then for theory for flexible boundary . theory is given both for a nondissipative and a dissipative flexible boundary . behavior of flexible medium itself is also examined .

practical requirements are discussed . for example, a conclusion is that to avoid tollmien-schlichting instability, the wave velocity of surface waves in absence of flow should coincide with tollmien-schlichting wave velocity at wavelength of /most dangerous/ tollmien-schlichting waves . moreover, damping should be large enough to prevent surface waves from developing but not so large that tollmien-schlichting waves are permissable . author states that a boundary that is both soft and light, one whose elastic constants are of same order as the dynamic pressure of the flow, may be practical for use at high speeds . this surface should have a small damping to avoid tollmien-schlichting type of instability and a large enough wave speed without flow to avoid surface wave instability . although paper is somewhat sketchy in places, it gives comprehensive coverage of stability of laminar flow over a flexible wall .

.I 1322

.T

qualitiative solutions of the stability equation for a boundary layer in contact with various forms of flexible surface .

.A

nonweiler, t.r.f.

.B

arc, 22670, march 1961 .

.W

qualitiative solutions of the stability equation for a boundary layer in contact with various forms of flexible surface .

an appropriate form of the boundary layer stability equation is developed for the condition where the fluid is in contact with an isotropic and homogeneous elastic medium, and various approximate analytical solutions obtained for certain types of surface, so as to reveal at least qualitatively the origin and characteristics of neutral oscillations . in the worked solutions the elastic medium is treated as nondissipative, and the interior boundary is supposed either fixed, or free of stress, or exposed to fluid .. the boundary layer, also, is treated as that over a flat-plate in an incompressible fluid .

the results obtained show that the presence of such a resiliant surface introduces the possibility of a number of other modes of oscillation schlichting waves . most of these modes have speeds of propagation determined largely by the properties of the elastic material, and their presence may well be effectively a matter of 'non-viscous' flow stability -dash a subject not treated here . the tollmien-schlichting mode has its minimum reynolds number increased by the presence of the surface, but if the interior boundary is free there may be an upper limit as well . indeed, a sufficiently thin free surface, or one of low rigidity, apparently eliminates neutral oscillations of this mode altogether, only at the expense, however, of the introduction of a mode of flexural waves .

.I 1323

.T

an investigation of the use of an auxiliary slot to

re-establish laminar flow on low drag aerofoils .

.A

cumming,r.w., gregory,b. and walker,w.s.

.B

arc r + m.2742, 1950.

.W

an investigation of the use of an auxiliary slot to

re-establish laminar flow on low drag aerofoils .

the use of an auxiliary slot on a laminar-flow aerofoil has been investigated to check whether laminar flow can be re-established by suction at the rear of the region of deposited dirt, flies, etc .

results indicate that in the absence of unfavourable pressure
gradients, it is possible to re-establish a laminar
boundary layer by removing a little more than the whole turbulent
layer reaching the slot, and preliminary estimates
suggest that with efficient ducting it should be possible to achieve
a reduction in overall effective drag coefficient by
this means .

.I 1324

.T

the effect on transition of isolated surface excrescences
in the boundary layer .

.A

gregory,n. and walker,w.s.

.B

pt. 1, arc r + m 2779, 1950.

.W

the effect on transition of isolated surface excrescences
in the boundary layer .

the effect of isolated surface excrescences in a
laminar boundary layer in producing disturbances
which may lead to turbulent flow has been examined
experimentally by several methods . photographs of some
of the flow patterns visualised by smoke and china-clay
techniques are given .

the critical heights of pimple which just give rise to
spreading wedges of turbulent flow have been measured
on a flat plate and on two aerofoils at several angles of

incidence . the results are analysed and are presented in a

form which enables approximate estimates to be made of the

protuberances permissible on laminar-flow surfaces at

full-scale flight reynolds numbers . the estimates suggest

that at an altitude of 30,000 ft the critical pimple height

is 0.004 in. for a speed of 350 m.p.h., whilst 0.002 in.

may be permissible at all subsonic speeds . at sea-level,

however, the tolerances are approximately halved .

.I 1325

.T

experiments on the use of suction through perforated

strips for maintaining laminar flow . transition and

drag measurements .

.A

gregory,n. and walker,w.s.

.B

arc r + m 3083, 1955.

.W

experiments on the use of suction through perforated

strips for maintaining laminar flow . transition and

drag measurements .

wind-tunnel tests are described in which suction

is applied at perforated strips, as an alternative to

porous strips or slots, in order to maintain a laminar boundary

layer . a test was first carried out on a single row of

perforations on a cambered plate, as a preliminary to the main

tests which were performed on strips of multiple rows of perforations drilled through the surface of a low-drag-type aerofoil 13 per cent thick and of 5-ft chord .

up to a wind speed of 180 ft sec it has been ascertained that suction may be safely applied to extend laminar flow provided the ratio of hole diameter to boundary-layer displacement thickness is less than 2, the ratio of hole pitch to diameter is less than 3 and there are at least three rows of holes in the strip . with less than three rows, the criteria are much more restrictive . it is possible to extend laminar flow by suction through perforations whose diameters and pitches exceed these values slightly, but only with the risk that excessive suction quantities will produce wedges of turbulent boundary layer originating at the holes .

a uniform distribution of suction through the holes was necessary . this was successfully obtained by two methods, the use of cells and throttle holes, and with tapered holes . in particular, tests were carried out on some panels supplied by handley page, ltd., in which the cells and tapered holes had been constructed by commercial methods, and the suction distribution proved satisfactory .

the resistance of some of the cellular arrangements was measured . it was found that when the suction quantities were the minimum required to maintain laminar flow, the additional losses in total head of the sucked air due to the resistance of the throttle holes could be made small compared with the loss in total head of the sucked boundary layer .

.I 1326

.T

interaction of secondary injectants and rocket exhaust

for thrust vector control .

.A

newton,j.f. and spaid,f.w.

.B

ars jnl. 32, 1962, 1203.

.W

interaction of secondary injectants and rocket exhaust

for thrust vector control .

tests were conducted with 1300- to 1500-lb thrust

solid rocket motors in order to investigate the

side-force generation mechanisms associated with

the injection of a secondary fluid into the

expansion cone of a solid propellant rocket nozzle for

thrust-vector control . the nozzles were 15

conicals with a nominal expansion ratio of . all

firings were conducted in zero-flow ejectors .

freon-12, water, and gascous nitrogen were used as

the injectant . nozzle-wall pressure profiles,

side thrust, and the nozzle-wall shock interface were

recorded . the general character of the

pressure disturbance was defined . the major portion of

the side force was generated by the pressure

disturbance downstream of the injector . the

axial-thrust augmentation generated by the injectant was calculated . the effects of nozzle-expansion

ratio and injector location on the side force

were clearly illustrated .

.I 1327

.T

on the propagation and structure of the blast wave .

.A

sakurai,a.

.B

j. phys. soc. japan, 8, 1953, 662.

.W

on the propagation and structure of the blast wave .

concerning blast waves with front surfaces of plane, cylindrical and

spherical shape, the propagation velocity u and the distribution of

hydrodynamical quantities are discussed . the solutions are constructed

in the form of power series in (c u), where c is the sound velocity of

undisturbed fluid . especially r, the distance of shock front from the

charge, is represented as,

where r is the characteristic length related to the energy of explosion,

j and are constants, and a=0,1,2 correspond to plane, cylindrical and

spherical case, respectively . in this paper the first approximations

for a=0,1 are discussed (the case a=2 has been discussed by g. i.

taylor) . the solution is obtained numerically for the case of the

adiabatic index . the approximate solution is also considered . using

these solutions, is found to be .

the second approximation will appear in part 2 to be published

subsequently .

.I 1328

.T

the production of aerodynamic forces by heat addition

on external surfaces of aircraft .

.A

willmarth,w.w.

.B

rand corp. rm.2078, 1957.

.W

the production of aerodynamic forces by heat addition

on external surfaces of aircraft .

within the framework of linearized

flow theory an equivalence between

a fluid mass source, a heat source,

and streamwise body forces is

developed . the equivalence between

the fluid mass source and heat

source was first noticed by hicks(2)

and later by chu . (3) using the

equivalence the flow field produced

by heat addition and by

magnetohydrodynamical body forces can be computed .

examples for a two-dimensional

flat plate, a delta wing, an axially

symmetric slender body, and a wedge-shaped

afterbody are computed at

subsonic and supersonic speeds .

the efficiency of lift or thrust
production by surface heat addition is
very low at subsonic speeds . at
supersonic speeds the efficiency is
compared with the efficiency of a
conventional turbojet-powered aircraft
configuration . it is found that
the efficiency of lift or thrust production
by heat addition on
two-dimensional bodies is approximately the same
as that for a
turbojet-powered two-dimensional body . the
efficiency is somewhat higher at low
supersonic mach numbers and behaves
as, decreasing to a
constant value as increases .
on the other hand, the efficiency of
thrust production by heat addition
increases linearly with mach number
when heat is added on the rear surface
of an axially symmetric afterbody
of parabolic shape .
.I 1329
.T
some aspects of non-stationary airfoil theory and its
practical application .
.A

sears,w.r.

.B

j. ae. scs. 8, 1951, 104.

.W

some aspects of non-stationary airfoil theory and its

practical application .

this paper consists of three notes on the theory of two-

dimensional thin airfoils in non-uniform motion ..

oscillating airfoil are collected from an earlier paper and are

presented in convenient forms for practical application .

rigid airfoil passing through a vertical-gust pattern having a

sinusoidal distribution of intensity . the lift is determined as a

function of the reduced frequency (which in this case is

proportional to the ratio of the airfoil chord and the wave length of the

gust pattern) and is presented in the form of a vector diagram .

it is shown that the lift acts at the quarter-chord point of the

airfoil at all times .

calculation of the amplitude of torsional oscillation of a fan blade

operating in the wake of a set of pre-rotation vanes . in a

numerical example the amplitude is found to be small even when

the vanes are spaced so that the exciting frequency coincides

with the natural frequency of the fan blade .

.I 1330

.T

on some fourier transforms in the theory of non-stationary

flows .

.A

garrick,i.e.

.B

proc. 5th int. cong. app. mech. 1938, j. wiley, 590.

.W

on some fourier transforms in the theory of non-stationary

flows .

the growth of lift on a airfoil

starting impulsively from rest to a uniform

velocity has been given by wagner (1925) . the

steady-state lift due to circulation on an

airfoil oscillating sinusoidally and moving with

uniform velocity has been given by theodorsen

the present paper based essentially on the

material of n. a. c. a. report no. 629 by the

author, discusses some reciprocal relations of

the nature of fourier transforms existing

between the functions of wagner and theodorsen .

kussner (1936) has already shown that wagner's

function may be derived from theodorsen's

function . by means of a superposition principle it

is possible to utilize these fundamental

functions to treat general problems in transient

expression which is accurate to within 2

percent is given for wagner's function . this

expression leads to a good approximate expression

for theodorsen's function in terms of the

exponential integral, instead of hankel functions .

an analogy is drawn between transient

hydrodynamic flows and transient electrical flows .

kussner (1936) has introduced a function

describing the growth of lift on an airfoil

entering a sharp edged vertical gust region . this

function bears a certain relation to wagner's

function which is briefly discussed .

.I 1331

.T

calculated responses of a large sweptwing airplane

to continuous turbulence with flight-test comparisons .

.A

bennett,f.v. and pratt,k.g.

.B

nasa tr r.69, 1960.

.W

calculated responses of a large sweptwing airplane

to continuous turbulence with flight-test comparisons .

calculated responses of symmetrical

airplane motions, wing

deformations, and wing loads due to gusts are

shown to compare favorably with

available flight-test results . these

calculated responses are based on

random-process theory, five degrees of

freedom, lifting-surface

aerodynamics, and one-dimensional vertical

turbulence . the extent to which
various degrees of freedom contribute
to the responses is examined and
in this connection the relative effects
of static and dynamic
aeroelasticity are determined .

.I 1332

.T

calculated spanwise lift distributions, influence functions
and influence coefficients for unswept wings in subsonic
flow .

.A

diederich,f.w. and zlotnick,m.

.B

naca r.1228, 1955.

.W

calculated spanwise lift distributions, influence functions
and influence coefficients for unswept wings in subsonic
flow .

spanwise lift distributions have been calculated for nineteen
unswept wings with various aspect ratios and taper ratios and
with a variety of angle-of-attack or twist distributions, including
flap and aileron deflections, by means of the weissinger method
with eight control points on the semispan . also calculated
were aerodynamic influence coefficients which pertain to a
certain definite set of stations along the span, and several
methods are presented for calculating aerodynamic influence

functions and coefficients for stations other than those stipulated .

the information presented herein can be used in the analysis
of untwisted wings or wings with known twist distributions, as
well as in aeroelastic calculations involving initially unknown
twist distributions .

.I 1333

.T

aerodynamic forces on wings in non-uniform motion .

.A

jones,w.p.

.B

arc r + m.2117, 1945.

.W

aerodynamic forces on wings in non-uniform motion .

the problem of determining the aerodynamic forces
acting on wings of finite span in non-uniform motion
in an incompressible, inviscid fluid is investigated . the underlying
theory is outlined in 2, and some known results
for the case of an aerofoil of infinite span are included in 3 . it is
shown in 4, by the use of operational methods,
that the growth of lift function k (s) corresponding to a sudden unit
change of incidence can be derived from the lift
function corresponding to simple harmonic translational motion .
from results given by the writer for rectangular
wings (1943) and tapered wings (1945) in simple harmonic motion
the corresponding values of k (s) are determined .
the growth of lift function k (s) for a wing penetrating a uniform

vertical gust can then be estimated as shown in 4

and 5 . by the use of approximate formulae for the growth of lift

curves given in fig. 2, the aerodynamic forces

corresponding to damped and growing translational oscillations are

derived .

certain integrals involved in the theory are evaluated in

appendix 1, and in appendix 2 the method of determining

k (s), when k (s) is known, is discussed in detail .

it is suggested that the aerodynamic forces acting on wings

of finite span for any type of motion can best be derived

from a knowledge of the forces corresponding to purely

divergent motion, which can be calculated by the methods

outlined in this report .

.I 1334

.T

calculated spanwise lift distributions and aerodynamic

influence coefficients for swept wings in subsonic

flow .

.A

diederich,f.w. and zlotnick,m.

.B

naca tn.3476, 1955.

.W

calculated spanwise lift distributions and aerodynamic

influence coefficients for swept wings in subsonic

flow .

spanwise lift distributions have been calculated for 61 swept wings with various aspect ratios and taper ratios and with a variety of angle-of-attack or twist distributions, including flap and aileron deflections, by means of the weissinger method with eight control points on the semispan . also calculated for these plan forms were aerodynamic influence coefficients which pertain to a certain definite set of stations along the span . the information presented herein can thus be used both in the analysis of untwisted wings or wings with known twist distributions and in aeroelastic calculations involving initially unknown twist distributions .

this paper supplements and is intended to be used in conjunction with naca tn 3014, where the same type of information, calculated in the same way, is presented for 19 unswept wings .

.I 1335

.T

use of freon-12 as a fluid for aerodynamic testing .

.A

huber,p.w.

.B

naca tn.1024, 1946.

.W

use of freon-12 as a fluid for aerodynamic testing .

the thermodynamic properties of freon-12 have been investigated to determine the possibilities of the use of this gas as a fluid for aerodynamic testing . the

values of velocity of sound
in freon-12, which are less
than one-half those in air,
are presented as functions
of temperatures and pressure,
including measurements at
room temperature . the density
of freon-12 is about
four times that of air . changes
in state of freon-12
may be predicted by means of the
ideal gas law with an
accuracy of better than 1 percent
at pressures below
freon-12 is shown not
to condense during an adiabatic
expansion from normal
conditions up to a mach number
of 3 . the values of the
ratio of specific heats
for freon-12 are lower than
that for air, and therefore
an additional parameter is
introduced, which must be
considered when comparisons
are made of aerodynamic tests
using freon-12 with those

using air .

the time lag of the

vibrational heat capacity

of freon-12 to a change in

temperature has been

measured and found to be

of the order of 2 x 10 second

at atmospheric temperature

and pressure . this time

is so short that no important

energy dissipations

should result in most

engineering applications .

.I 1336

.T

studies of the use of freon-12 as a wind tunnel testing

medium .

.A

vondoenhoff.

.B

naca tn.3000, 1953.

.W

studies of the use of freon-12 as a wind tunnel testing

medium .

a number of studies relating to

the use of freon-12 as a

substitute medium for air in aerodynamic

testing have been made . the
use of freon-12 instead of air makes
possible large savings in
wind-tunnel drive power . because of the
fact that the ratio of specific
heats is approximately 1.13 for freon-12
as compared with 1.4 for air,
some differences exist between data
obtained in freon-12 and in air .
methods for predicting aerodynamic
characteristics of bodies in air
from data obtained in freon-12, however,
have been developed from the
concept of similarity of the streamline
pattern . these methods,
derived from consideration of two-dimensional
flows, provide substantial
agreement in all cases for which comparative
data are available . these
data consist of measurements throughout
a range of mach number from
approximately 0.4 to 1.2 of pressure
distributions and hinge moments on
swept and unswept wings having aspect
ratios ranging from 4.0 to 9.0,
including cases where a substantial
part of the wing was stalled .

the freon charging and recovery
system used for the langley
low-turbulence pressure tunnel is described .
.I 1337
.T
study of effects of sweep on the flutter of cantilever
wings .
.A
barmby,j.g.
.B
naca r.1014, 1951.
.W
study of effects of sweep on the flutter of cantilever
wings .
an experimental and analytical investigation of the flutter
of sweptback cantilever wings is reported . the experiments
employed groups of wings swept back by rotating and by
shearing . the angle of sweep ranged from 0 to 60 and
mach numbers extended to approximately 0.85 . a theoretical
analysis of the air forces on an oscillating swept wing of high
length-chord ratio is developed, and the approximations
inherent in the assumptions are discussed . comparison with
experiment indicates that the analysis developed in the present
report is satisfactory for giving the main effects of sweep, at
least for nearly uniform cantilever wings of high and moderate
length-chord ratios . a separation of the effects of finite span
and compressibility in their relation to sweep has not been made

experimentally but some combined effects are given . a

discussion of some of the experimental and theoretical trends is

given with the aid of several tables and figures .

.I 1338

.T

investigation to determine effects of center of gravity location on the

transonic flutter characteristics of a 45degree sweptback wing .

.A

jones, g.w. and unangst, j.r.

.B

naca rm l55k30, 1956 .

.W

investigation to determine effects of center of gravity location on the

transonic flutter characteristics of a 45degree sweptback wing .

an experimental investigation has been conducted in the 26-inch langley

transonic blowdown tunnel to determine effects of center-of-gravity

location on the transonic flutter characteristics of a 45degree

swept-back-wing plan form of aspect ratio 4.0 and taper ratio 0.6 .

solid-construction models of the plan form with streamwise naca 65a004

airfoil sections and center-of-gravity locations at approximately 34

percent chord, 46 percent chord, and 58 percent chord, respectively, were

fluttered at several mach numbers between 0.8 and 1.35 .

it was found that, for streamwise mach numbers from 0.8 to 1.0, the

variation with mach number of the ratio of experimental flutter speed to

a calculated incompressible flutter speed was not affected by

center-of-gravity location . however, for mach numbers from 1.0 to 1.35, there

was an increase in flutter-speed ratio with mach number which was different for each center-of-gravity position . data from wings with

successively more forward center-of-gravity locations showed

successively larger values of flutter-speed ratio at mach numbers from 1.0 to

.I 1339

.T

calculation of flutter characteristics for finite-span swept or unswept

wings at subsonic and supersonic speeds by a modified strip analysis .

.A

yates, e.c.

.B

naca rm l57l10, 1958 .

.W

calculation of flutter characteristics for finite-span swept or unswept

wings at subsonic and supersonic speeds by a modified strip analysis .

a method has been developed for calculating flutter characteristics of

finite-span swept or unswept wings at subsonic and supersonic speeds .

the method is basically a rayleigh type analysis and is illustrated with

uncoupled vibration modes although coupled modes can be used . the

aerodynamic loadings are based on distributions of section lift-curve

slope and local aerodynamic center calculated from three-dimensional

steady-flow theory . these distributions are used in conjunction with

the /effective/ angle-of-attack distribution resulting from each of the

assumed vibration modes in order to obtain values of section lift and

pitching moment . circulation functions modified on the basis of

loadings for two-dimensional airfoils oscillating in a compressible flow are

employed to account for the effects of oscillatory motion on the

magnitudes and phase angles of the lift and moment vectors .

flutter characteristics have been calculated by this method for 12 wings of varying sweep angle, aspect ratio, taper ratio, and center-of-gravity position at mach numbers from 0 to as high as 1.75 . comparisons of the results with experimental flutter data indicate that this method gives generally good flutter results for a broad range of wings .

.I 1340

.T

method of controlling stiffness properties of a solid-construction model wing .

.A

land, n.s. and abbott, f.t.

.B

naca tn 3423, 1955 .

.W

method of controlling stiffness properties of a solid-construction model wing .

a simple method is presented for controlling the bending and torsional stiffnesses of a solid-construction model wing . the method consists of weakening the wing by drilling holes through the wing normal to the chord plane . aerodynamic continuity is maintained by filling the holes with a relatively soft material . the important parameters controlling the stiffnesses are the amount of material removed by drilling, the ratio of hole diameter to wing thickness, and the plan-form pattern of the holes . data are given which may be used for predicting the stiffness of a model wing weakened in this manner .

.I 1341

.T

investigation of wing flutter at transonic speeds for six systematically

varied wing plan forms .

.A

jones, g.w. and dubose, h.c.

.B

naca rml53g10a, 1953 .

.W

investigation of wing flutter at transonic speeds for six systematically

varied wing plan forms .

an investigation of the effects of systematic variations in wing plan

form on the flutter speed at mach numbers between 0.73 and 1.43 has been

conducted in the 26-inch langley transonic blowdown tunnel . the angle

of sweepback was varied from 0degree to 60degree on wings of aspect

ratio 4, and the aspect ratio was varied from 2 to 6 on wings with

experimental flutter speed and the reference flutter speed calculated on

the basis of incompressible two-dimensional flow . this ratio,

designated as the flutter-speed ratio, is plotted as a function of mach

number for the various wings . it is found that the flutter-speed ratio

increased rapidly past sonic speed for sweep angles of 45degree and

less, indicating a favorable effect of mach number . for sweepback of

mach number range of the tests . reducing the aspect ratio had a

favorable effect on the flutter-speed ratio which was of the order of 100

percent higher for the aspect-ratio-2 wing than for the aspect-ratio-6

wing . this percentage difference was nearly constant throughout the

mach number range, indicating that the effect of mach number was about

the same for all aspect ratios tested .

.I 1342

.T

the calculation of aerodynamic loading on surfaces of any shape .

.A

falkner, v.m.

.B

r + m 1910, august 1943 .

.W

the calculation of aerodynamic loading on surfaces of any shape .

the object of the report is to establish a routine method for the calculation of aerodynamic loads on wings of arbitrary shape . the method developed is based on potential theory and uses a general mathematical formula for continuous loading on a wing which is equivalent to a double fourier series with unknown coefficients . in order to evaluate the unknown coefficients the continuous loading is split up into a regular pattern of horseshoe vortices, the strengths of which are proportional to the unknown coefficients and to standard factors which are given in a table . the total downwash at chosen pivotal points is obtained by summing the downwashes due to the individual vortices, a process which is simplified by the use of specially prepared tables of the properties of the horseshoe vortex . by equating the downwash to the slope of the wing at each pivotal point, simultaneous equations are obtained, the solution of which defines the unknown coefficients .

the first layout involves a total of 76 vortices over the wing, and a second layout, involving a total of 84, is shown to be of superior accuracy . the effect on the solution of the number of pivotal points is investigated and it is concluded that by a suitable choice, it is unnecessary to use a large number . results for a rectangular wing at with those obtained by other workers and it appears that there may be errors in published results in at least one of these cases . immediate development includes the application to the calculation of the characteristics of actual sweptback wings, including rotary derivatives, and future development includes also applications in wind tunnel design and technique .

.I 1343

.T

formulas for the supersonic loading, lift and drag of flat swept back wings with leading edges behind the mach lines .

.A

cohen,d.

.B

naca r.1050, 1951.

.W

formulas for the supersonic loading, lift and drag of flat swept back wings with leading edges behind the mach lines .

the method of superposition of linearized conical flows has been applied to the calculation of the aerodynamic properties, in supersonic flight, of thin flat, swept-back wings at an angle of attack . the wings are assumed to have rectilinear plan forms, with tips parallel to the stream, and to taper in the conventional sense . the investigation covers the moderately supersonic speed range where the mach lines from the leading-edge apex lie ahead of the wing . the trailing edge may lie ahead of or behind the

mach lines from its apex . the case in which the mach cone
from one tip intersects the other tip is not treated .
  formulas are obtained for the load distribution, the total lift,
and the drag due to lift . for the cases in which the trailing edge
is outside the mach cone from its apex (supersonic trailing edge),
the formulas are complete . for the wing with both leading and
trailing edges behind their respective mach lines, a degree of
approximation is necessary . it has been found possible to give
practical formulas which permit the total lift and drag to be
calculated to within 2 or 3 percent of the accurate
linearized-theory value . the local lift can be determined accurately over
most of the wing, but the trailing-edge-tip region is treated only
approximately .
  charts of some of the functions derived are included to
facilitate computing, and several examples are worked out in outline .

.I 1344

.T

atmospheric entries with vehicle lift-drag ratio modulated
to limit deceleration and rate of deceleration vehicles
with maximum lift-drag ratio of 0. 5.

.A

katzen,e.d. and levy,l.l.

.B

nasa tn.d1145, 1961.

.W

atmospheric entries with vehicle lift-drag ratio modulated
to limit deceleration and rate of deceleration vehicles

with maximum lift-drag ratio of 0. 5.

an analysis has been made of

atmosphere entries for which the

vehicle lift-drag ratio was modulated

to maintain specified maximum

decelerations and or maximum deceleration

rates . the part of the

vehicle drag polar used during modulation

was from maximum lift

coefficient to minimum drag coefficient . the

entries were at parabolic velocity

and the vehicle maximum lift-drag ratio

was 0.5 . two-dimensional

trajectory calculations were made for a

nonrotating, spherical earth with an

exponential atmosphere . the results

of the analysis indicate that for a

given initial flight-path angle,

modulation generally resulted in a

reduction of the maximum deceleration

to 60 percent of the unmodulated value or

a reduction of maximum deceleration

rate to less than 50 percent of the

unmodulated rate . these results were

equivalent, for a maximum

deceleration of 10g, to lowering the undershoot

boundary 24 miles with a resulting

decrease in total convective heating to

the stagnation point of 22 percent .

however, the maximum convective heating

rate was increased 18 percent,. the

maximum radiative heating rate and total

radiative heating were each

increased about 10 percent .

.I 1345

.T

the use of aerodynamic lift during entry into the earth's

atmosphere .

.A

lees,l.

.B

space tech. lab. r. gm-tr-0165-00519, 1958.

.W

the use of aerodynamic lift during entry into the earth's

atmosphere .

by employing aerodynamic lift during entry into the earth's

atmosphere at either orbital or /escape/ velocity, the range of

allowable entry angles for a prescribed peak deceleration is

greatly increased, while the total heat energy transferred to the

vehicle can be held to about the same value as for a nonlifting

vehicle . only modest lift-drag ratios are required beyond peak g

to prevent the deceleration from exceeding the peak value, or to

prevent the vehicle from skipping out of the earth's atmosphere .

thus, the difficult guidance and control problem is greatly alleviated,. in particular, for return from the moon or other

planets the necessity for multiple-pass drag braking is eliminated .

.I 1346

.T

modulated entry .

.A

grant,f.c.

.B

nasa tn.d452, 1960.

.W

modulated entry .

the technique of modulation, or variable coefficients, is discussed and the analytical formulation is reviewed . representative numerical results of the use of modulation are shown for the lifting and nonlifting cases . these results include the effects of modulation on peak acceleration, entry corridor, and heat absorption . results are given for entry at satellite speed and escape speed . the indications are that coefficient modulation on a vehicle with good lifting capability offers the possibility of sizable loading reductions or, alternatively, wider corridors,. thus, steep entries become practical from the loading standpoint . the amount of steepness depends on the acceptable heating penalty . the price of sizable fractions of the possible gains does not appear to be excessive .

.I 1347

.T

approximate analysis of atmospheric entry corridors

and angles .

.A

luidens,r.w.

.B

nasa tn.d590, 1961.

.W

approximate analysis of atmospheric entry corridors

and angles .

a simple closed-form solution

for the achievable corridor depths

and entry angles as a function of

g-load limit, entry velocity, and

vehicle aerodynamics and thermodynamics

is developed for two modes of

vehicle operation, constant angle

of attack and modulated angle of attack .

for constant angle of attack,

operation at maximum negative lift

coefficient on the overshoot bound,

and at an angle of attack between

zero and that for maximum lift-drag

ratio on the undershoot bound, gives

the deepest corridor . for modulated

angle of attack, operating at

maximum negative lift coefficient on the

overshoot bound and modulating

the angle of attack from maximum

positive lift coefficient to zero on

the undershoot bound give the deepest
corridor . the modulated angle of
attack gives corridor depths two to
four times larger than the fixed
angle of attack . for both cases the
corridor depth is increased by
increasing maximum lift-drag ratio,
increasing g limit, and decreasing
entry velocity .

consideration of hot-gas radiation
places a limit on the maximum
angle of attack for either mode of
operation . if a maximum free-stream
reynolds number limit must be placed
on the vehicle to ensure a laminar
boundary layer, the deep atmospheric
penetrations associated with
configurations with high lift-drag ratio
may be ruled out . both of these
thermodynamic considerations reduce
the acceptable corridor depth below
the value calculated from aerodynamic
considerations alone .

.I 1348

.T

radiative heat transfer during atmosphere entry at
parabolic velocity .

.A

yoshikawa,k.k. and wick,b.h.

.B

nasa tn.d1074, 1961.

.W

radiative heat transfer during atmosphere entry at

parabolic velocity .

stagnation point radiative heating

rates for manned vehicles entering

the earth's atmosphere at parabolic

velocity are presented and compared

with corresponding laminar convective

heating rates . the calculations

were made for both nonlifting and lifting

entry trajectories for vehicles

of varying nose radius, weight-to-area

ratio, and drag . it is concluded

from the results presented that radiative

heating will be important for

the entry conditions considered .

.I 1349

.T

effects of simulated rocket jet exhaust on stability

and control of a research type airplane configuration

at a mach number of 6. 86 .

.A

fetterman,d.e.

.B

nasa tm.x127, 1959.

.W

effects of simulated rocket jet exhaust on stability

and control of a research type airplane configuration

at a mach number of 6. 86 .

an investigation has been

undertaken in the langley 11-inch

hypersonic tunnel at a free-stream

mach number of 6.86 to determine

the jet-interference effects at

high jet-static-pressure ratios on the

stability and control of a

research-type airplane configuration .

compressed-air tests with a jet

exhausting from the base of the

fuselage were conducted over a reynolds

number range of 0.57 x 10 to

and over a jet-static-

pressure-ratio range of 0 to 1460 . the results

of these tests indicated that

the operation of the jet induced a

sizable separated-flow region over

the vertical- and horizontal-tail

surfaces which could be approximately

duplicated at low angles of attack

by use of metal jet-boundary

simulators . the results of force tests,

during which these metal

jet-boundary simulators were used,

indicated that this separated-flow

region caused a large reduction in

the longitudinal stability and

control and a smaller reduction in the

lateral and directional stability

and control . by extending the

divergent section of the nozzle and thus

reducing the jet-static-pressure

ratio, these losses were diminished .

.I 1350

.T

effects of jet billowing on stability of missile-type

bodies at mach 3. 85 .

.A

salmi,r.j.

.B

nasa tn.d284, 1960.

.W

effects of jet billowing on stability of missile-type

bodies at mach 3. 85 .

the interference effects of a

billowing jet on the forces and

moments of two missile-type bodies

were investigated in the nasa lewis

2-by 2-foot mach 3.85 wind tunnel .

to simulate a rocket jet, pressurized

nitrogen was exhausted from an

annular sonic nozzle .

the results indicate that for

both models the stability parameter

moment coefficient with angle of

attack) in the region of zero angle

of attack was favorably influenced

by the interference resulting from

separation due to jet billowing .

schlieren photographs are presented

that show the separation due to the

jet billowing at various pressure

ratios and angles of attack .

.I 1351

.T

exploratory tests of the effects of jet plumes on the

flow over cone- cylinder flare bodies .

.A

falanga,r.a.

.B

nasa tn.d1000, 1962.

.W

exploratory tests of the effects of jet plumes on the

flow over cone- cylinder flare bodies .

schlieren photographs have been

taken of the flow over
cone-cylinder-flare bodies to study the
extent of boundary-layer separation
due to the presence of rocket jet
plumes . tests were made of three
cone-cylinder-flare configurations in
the langley 11-inch hypersonic tunnel at
a mach number of 9.65 and in the
langley unitary plan wind tunnel at a
mach number of 4.65 with two
additional configurations . the stream
reynolds number varied from
approximately 317,000 to 582,000 based on
model length . the conical flares
had half-angles of 7 or 13 and
contained one of two test nozzles with
a design mach number of 3.72 or 4.53 .
the test nozzles were operated with
compressed air and were designed to
simulate a solid-propellant rocket
motor operating at altitudes between
to free-stream static-pressure ratio
varied from jet off to 1,150 for the
test nozzle with a design mach
number of 3.72 and from jet off to
mach number of 4.53 . for most of
the tests the angle-of-attack range

was 0 to -4,. some additional tests

were made at 2 and 4 .

measurements taken from flow pictures

indicated that at zero angle

of attack on all configurations tested

with jet on the boundary layer

separates ahead of the flare-cylinder

juncture and the separation point

moves toward the cone-cylinder juncture

with an increase in pressure

ratio . increasing angle of attack

reduced the extent of boundary-layer

separation on the windward side as

did increasing the stream mach number

from 4.65 to 9.65 . other parameters

which tended to reduce the extent

of boundary-layer separation were ..

number, (b) decreasing stream reynolds

number, and (c) displacing nozzle

exit rearward .

.I 1352

.T

aerodynamic investigation of a parabolic body of revolution

at mach number of 1. 92 and some effects of an annular

supersonic jet exhausting from the base .

.A

love,e.s.

.B

naca tn.3709, 1956.

.W

aerodynamic investigation of a parabolic body of revolution

at mach number of 1. 92 and some effects of an annular

supersonic jet exhausting from the base .

an aerodynamic investigation

of a parabolic body of revolution was

conducted at a mach number of 1.92

with and without an annular supersonic

jet exhausting from the base .

measurements with the jet inoperative were

made of lift, drag, pitching moment,

radial and longitudinal pressure

distributions, and base pressures .

with the jet in operation,

measurements were made of the pressures

over the rear of the body with the

primary variables being angle of attack,

ratio of jet velocity to

freestream velocity, and ratio of jet

pressure to stream pressure .

the results with the jet inoperative

showed that the radial

pressures over the body varied appreciably

from the distribution generally

employed in most approximate theories .

the linearized solutions for lift,

pitching moment, and center of pressure

gave relatively poor predictions

of the experimental results . an analysis

of several theoretical methods

for calculating pressure distribution and

wave drag showed that some

methods gave results in considerable

disagreement with experimental values .

maximum effects of the jet were

obtained at the lower ratio of jet

velocity to stream velocity and the

highest ratio of jet pressure to

stream pressure . these effects amounted

to a slight decrease in

fore-drag, a reduction in lift, and a shift

of center of pressure in a

destabilizing direction .

.I 1353

.T

investigation of a two-step nozzle in the langley 11in .

.A

hypersonic tunnel .

.B

mclellan,c.h.

naca tn.2171, 1950.

.W

investigation of a two-step nozzle in the langley 11in .

hypersonic tunnel .

flow surveys have been made in

the first of several nozzles to be

investigated in the langley 11-inch

hypersonic tunnel . the nozzle was

designed by the method of characteristics

for a mach number of 6.98 . two

step expanded the air in the

horizontal plane to a mach number of 4.36

and the second in the vertical

plane to a mach number of 6.98 .

the test results showed that, although

a maximum mach number of

about 6.5 was obtained, the flow in the

test section was not sufficiently

uniform for quantitative wind-tunnel test

purposes . deviations from the

design flow were traced to the presence

of a thick boundary layer which

developed in the first step along the

parallel walls .

.I 1354

.T

investigation of the flow through a single stage two

dimensional nozzle in the langley 11in . hypersonic

tunnel .

.A

mclellan,c.h.

.B

naca tn.2223, 1950.

.W

investigation of the flow through a single stage two

dimensional nozzle in the langley 11in . hypersonic

tunnel .

flow surveys have been made in

the second of several nozzles to be

investigated in the langley 11-inch

hypersonic tunnel . the single-stage,

two-dimensional nozzle was designed

by the method of characteristics for

a mach number of 7.08 without

boundary-layer corrections .

the test results show that

reasonably uniform flow at an average

mach number of about 6.86 was

obtained in a central region of the stream

at the test section . this region

had a cross section nearly 5 inches

square and had a deviation from

uniform flow of less than 1 percent in

mach number and 0.3 in flow angle .

an increase in mach number of

about 3 percent occurred during test

runs of about 60 seconds duration

because of distortions of the boundaries

at the first minimum due to

nonuniform heating of the nozzle blocks

during the tests .

.I 1355

.T

boundary layer displacement effects in air at mach

numbers of 6. 8 and 9. 6.

.A

bertram,m.h.

.B

nasa tr.r22, 1959.

.W

boundary layer displacement effects in air at mach

numbers of 6. 8 and 9. 6.

measurements are presented for pressure

gradients induced by a laminar boundary layer on a

flat plate in air at a mach number of 9.6 and for the

drag of thin wings at a mach number of about 6.8

and zero angle of attack . the pressure

measurements at a mach number of 9.6 were made in the

presence of substantial heat transfer from the

boundary layer to the plate surface . the measured

pressure distribution on the surface of the plate was

predicted with good accuracy by a modification to

insulated-plate displacement theory which allows for the effect of the heat transfer and temperature

gradient along the surface on the boundary-layer

displacement thickness .

the total drag of thin wings with square and delta

plan forms was measured at a nominal mach number

of 6.8 over a reasonably wide range of reynolds

numbers . the total drag was found to be greater

than can be explained by adding a classical value of

laminar skin friction to the estimated pressure drag .

the difference is, in general, explained by the

increase in skin friction (20 to 40 percent) caused by

the boundary-layer-induced pressures .

.I 1356

.T

secondary flow fields embedded in hypersonic shock

layers .

.A

seiff,a.

.B

nasa tn.d1304, 1962.

.W

secondary flow fields embedded in hypersonic shock

layers .

when a ramp or other compression

surface is located in a locally

supersonic region behind a hypersonic

bow shock wave, it generates a

secondary shock wave . the ramp flow
disturbance may be viewed as an
embedded newtonian impact flow if the
embedded shock layer is thin .
examination of the applicability of newtonian
flow theory to cones and wedges
in uniform streams suggests that this
theory can be expected to give a
useful approximation to the surface
pressures .

a pressure equation based on this
concept predicts a number of
interesting things .. first, pressures
can differ from simple newtonian
theory by factors of 1 5 to 3,. for
example, on flare stabilizers on
blunt-nosed bodies of revolution, pressures
are lower than newtonian and diminish
with increasing flight speed in the
hypersonic speed range . the calculated
pressures vary over the flare surface
as a result of the nonuniformity of
its incident stream, and depend on the
axial location of the flare . in
the case of a flap mounted on a
large-angled blunt-nosed cone, the pressure
coefficients vary from 1 to 5 through

the variable entropy layer . a

pressure coefficient of 5 greater than

the maximum possible in newtonian

flow can occur because the compression

process is more efficient than a

single shock wave process . on areas

of the flap that protrude through

the main bow wave, the pressure

coefficient should revert to the simple

newtonian value .

equations are developed for the

initial slopes of the normal-force

and pitching-moment curves of a flare

stabilizer . in the simplest case

these differ from conventional newtonian

theory by the ratio of local

dynamic pressure to free-stream dynamic

pressure . this ratio takes values

as low as 0.1 in some of the examples considered .

.I 1357

.T

compressive buckling of simply supplorted plates with

longitudinal stiffeners .

.A

seide,p. and stein,m.

.B

naca tn.1825, 1949.

.W

compressive buckling of simply supplorted plates with

longitudinal stiffeners .

charts are presented for the analysis of the stability under

compression of simply supported rectangular plates with one, two, three,

and an infinite number of identical equally spaced longitudinal

stiffeners that have zero torsional stiffness .

.I 1358

.T

compressive buckling of simply supported plates with

transverse stiffeners .

.A

budiansky,b. and seide,p.

.B

naca tn.1557, 1948.

.W

compressive buckling of simply supported plates with

transverse stiffeners .

charts are presented for the analysis of the stability under

longitudinal compression of simply supported rectangular plates with

several equally spaced transverse stiffeners that have both torsional

and flexural rigidity .

.I 1359

.T

compression tests on circular cylinders stiffened longitudinally

by closely spaced z-section stringers .

.A

peterson,j.p. and dow,m.b.

.B

nasa memo. 2-12-59l, 1959.

.W

compression tests on circular cylinders stiffened longitudinally

by closely spaced z-section stringers .

six circular cylinders stiffened longitudinally by closely spaced

z-section stringers were loaded to failure in compression . the results

obtained are presented and compared with available theoretical results

for the buckling of orthotropic cylinders . the results indicate that

the large disparity that exists between theory and experiment for

unstiffened compression cylinders may be significantly smaller for

stiffened cylinders .

.I 1360

.T

simplified analysis of general instability of stiffened

shells in pure bending .

.A

shanley,f.r.

.B

j. ae. scs.16, 1949, 590.

.W

simplified analysis of general instability of stiffened

shells in pure bending .

although much work has been done to develop a theory for

the failure of shells by general instability, there is at present no

simple method by which the size of the frames may be determined for any given diameter, bending moment,.and frame spacing .

such a method is needed in determining the optimum design for

stiffened shells, to be used as a basis for weight analysis of

fuselages, and other shell structures . in an extension of the work

done for the rand corporation, a simple coefficient has been

determined for this purpose . since it appears that this method

may also be useful in design calculations, a brief description is

presented below .

.I 1361

.T

large deflections of structures subjected to heating

and external loads .

.A

turner,j.m.

.B

j. ae. scs. 1960.

.W

large deflections of structures subjected to heating

and external loads .

  the method of direct formulation of the stiffness matrix is

extended to include the effects of nonuniform heating and large

deflections . the purpose is to develop an analytical tool for the

treatment of actual structures .

  in the solution of aeroelastic problems the relations between

forces and deflections must be determined . the usual stiffness

matrix formulation of this relationship is limited to small

temperature changes and small deflections . for large temperature changes additional terms are required . also the problem becomes geometrically nonlinear when large deflections are involved . to overcome the inherent difficulties of the nonlinear problem for practical structures either an iterative or a step-by-step procedure must be used . the force-deformation relations necessary for this step-by-step or iterative approach are derived for an axially loaded member and for a plate element including the effects of thermal strains .

.I 1362

.T

non-linear analysis of heated, cambered wings by the matrix force method .

.A

warner lansing, irving w. jones, paul ratner

.B

grumman aircraft engineering corporation

.W

non-linear analysis of heated, cambered wings by the matrix force method .

various extensions of the matrix force method for complex structure analysis are presented and illustrated with the objective of expanding its range to handle the problems likely to be encountered in advanced vehicle wing design . methods are covered in detail for (1) determining the change in flexibility that occurs when thermal stresses are present, and also how large these stresses must be to cause buckling, (2) including the non-linear effect of large deflections by an iterative procedure, and (3) analyzing a wing that is initially slightly cambered and warped with either or both of the aforementioned effects present .

formulas are given for calculating the input matrix terms as are the

matrix equations and supporting theoretical discussion . an example

illustrates the nature and magnitude of the effects being examined .

.I 1363

.T

a characteristic type of instability in the large deflections

of elastic plates .

.A

ashwell,d.g.

.B

proc. roy. soc. a, 214, 1952, 98.

.W

a characteristic type of instability in the large deflections

of elastic plates .

part 1. from a general equation governing

the bending of thin elastic plates into certain

types of surfaces of revolution are derived

expressions for the behaviour of rectangular plates

with initial curvatures, subjected to pure

bending about one axis . it is found that such

plates exhibit the type of instability characteristic

of thin-walled structures which depend

for their stiffness on curvature . curves are

drawn showing the deformation suffered by

such plates, and an expression for the critical

bending moment at which instability occurs is obtained . experimental results show satisfactory

agreement .

  part 2. the analysis of part 1 is extended to deal

with the case of flat square or rectangular

plates loaded by distributed bending moments

applied to all four edges . curves are drawn

to describe their behaviour, and they are found

to exhibit the characteristic instability

displayed by thin-walled curved structures .

experimental verification is satisfactory .

.I 1364

.T

an experimental investigation of the interaction between shock waves and

 boundary layers .

.A

gadd, g.e., holder, d.w. and regan, j.d.

.B

proc. of the roy. soc. of london, ser. a., vol. 226, 1954, pp. 227-253

.W

an experimental investigation of the interaction between shock waves and

 boundary layers .

an account is given of an investigation into the interaction between the

 boundary layer on a flat plate and a shock wave produced either

externally, by a wedge in the supersonic mainstream, or from within the

boundary layer, by a wedge held in contact with the plate . a wide range

 of free-stream mach numbers, boundary-layer reynolds numbers, and shock

 strengths has been covered, shock strength being defined as the ratio

of the static pressure downstream of the shock to the static pressure
upstream of it . variations in these parameters can have large effects
on the interaction, and there are also large differences between cases
with externally generated shocks and cases where the shock is generated
from within the boundary layer . the investigation has thrown light on
the physical mechanisms involved . it is found that many of the major
features of the interaction arise because the boundary layer separates
from the surface ahead of the shock wave . the conditions under which
separation occurs and the behaviour of the separated boundary layer thus
have important effects, in terms of which, for example, the differences
between the interactions observed with laminar and with turbulent
boundary layers may be explained .

.I 1365

.T

approximate calculation of the laminar boundary layer .

.A

thwaites,b.

.B

aero. quart. 1, 1949-1950.

.W

approximate calculation of the laminar boundary layer .

after analyzing a large class of boundary-layer
velocity-profiles, the author discovered that the functions l(m) and
h(m) for all such cases differ only slightly from each other
over the whole range of positive and negative pressure
gradients . here l, m and h are defined by
being the

velocity-component in the x direction and u the value of u at the

edge of the boundary-layer and and the displacement

and momentum thickness, respectively . based on this

discovery, an approximate method is proposed by constructing

two universal curves l(m) and h(m) for all conceivable

boundary-layer flows found in practice . once these are

chosen, karman's momentum-integral can be written in the

form, v being the kinematic

viscosity coefficient, and can be integrated numerically . as

examples, both howarth's and hartree's

tained is considered good for practical purposes .

.I 1366

.T

the compressible laminar boundary layer with heat transfer

and arbitrary pressure gradient .

.A

cohen,c.b. and reshotko,e.

.B

naca r.1294, 1956.

.W

the compressible laminar boundary layer with heat transfer

and arbitrary pressure gradient .

an approximate method for the calculation of the

compressible laminar boundary layer with heat transfer and

arbitrary pressure gradient, based on thwaites' correlation

concept, is presented . the method results from the application

of stewartson's transformation to prandtl's equations, which

yeilds a nonlinear set of two first-order differential equations .
these equations are then expressed in terms of dimensionless
parameters related to the wall shear, the surface heat transfer,
and the transformed free-stream velocity . thwaites' concept
of the unique interdependence of these parameters is assumed .
the evaluation of these quantities is then carried out by utilizing
exact solutions recently obtained .
with the resulting relations, methods are derived for the
calculation of the two-dimensional and axially symmetric
laminar boundary layer with arbitrary free-stream velocity
distribution . mach number, and surface temperature level .
the combined effect of heat transfer and pressure gradient
is demonstrated by applying the method to calculate the
characteristics of the boundary layer on thin supersonic surfaces
and in a highly cooled, convergent-divergent, axially symmetric
rocket nozzle .

.I 1367

.T

a theoretical investigation of the effects of mach number, reynolds
number, wall temperature and surface curvature on laminar separation in
supersonic flow .

.A

gadd, g.e.

.B

a.r.c. 18,494, june 1956 .

.W

a theoretical investigation of the effects of mach number, reynolds number, wall temperature and surface curvature on laminar separation in supersonic flow .

laminar separation in supersonic flow is investigated by an extension of stratford's method . it is assumed that separation is of the usual practical type, taking place upstream of the shock wave or other agency provoking it . the results of the analysis agree well in most respects with experiment .

.I 1368

.T

three dimensional viscous wakes .

.A

steiger,m.h. and bloom,m.h.

.B

j.fluid mech. 14, 1962,233.

.W

three dimensional viscous wakes .

the velocity fields of three-dimensional viscous wakes are examined with the use of the boundary-layer approximations, osoen's linearization of the convective terms, and the assumption of constant fluid properties . transform methods yield solutions for general types of initial conditions . as an illustration, the axial velocity distribution of a wake whose initial isovels (lines of constant velocity) are of elliptic shape and their decay to axial symmetry are demonstrated .

both laminar and turbulent flows are considered .

.I 1369

.T

steady motion of a sphere., oseen's criticism and solution .

.A

.B

.W

steady motion of a sphere., oseens's criticism and solution .

the formula of stokes for the resistance experienced slowly

moving sphere has been employed in physical researches of fundamental

importance, as a means of estimating the size of minute globules of

water, and thence the number of globules contained in a cloud of

given mass . consequently the conditions of its validity has been much

discussed both from the experimental and from the theoretical side .

.I 1370

.T

some remarks on the flat plate boundary layer .

.A

lewis,j.a. and carrier,g.f.

.B

q. app. math. 7, 1949, 228.

.W

some remarks on the flat plate boundary layer .

the authors discuss the solutions for the flow of a viscous

incompressible fluid near the leading edge of a semi-infinite

flat plate without pressure gradient . the oseen linearization is employed which approximates the equations of motion and continuity by where are the coordinate directions, the corresponding velocity components and the uniform free stream velocity which is parallel to the plate . defining a perturbation stream function by the differential equation to be solved is with boundary conditions far from the plate and when y=0 and . the authors discuss the problem by applying the two-dimensional fourier transform and obtain an explicit solution for the velocity gradient at the plate which is in disagreement with the result of the blasius solution . from this the authors conclude that it would be more appropriate to use a velocity other than in the linearization of the equations of motion and suggest replacing by where . this choice does not affect the solution far from the plate but gives on the plate and in comparison with blasius solution indicates that c=0.35 . the solution of the modified oseen equation with this value of c then seems acceptable as the approximate solution in the region intermediate between the stokes flow and the free stream . on the basis of these considerations, the authors suggest an iteration procedure for obtaining the exact solution for the above problem as well as a solution for the plate of finite length .

.I 1371

.T

axisymmetric free mixing with swirl .

.A

steiger,m.h. and bloom,m.h.

.B

pibal r.628.

.W

axisymmetric free mixing with swirl .

viscous laminar axially-symmetric

free-mixing with small, moderate,

and large swirl is investigated by a

boundary layer type of analysis with

integral methods . moderate and small

swirls are formally the same,

differing only in the order of their associated

radial pressure gradients . neither

induces significant axial pressure gradients,.

consequently their effect on the

axial flow is negligible . for moderate and

small swirl an interesting feature

is the swirl decay . in both compressible

and incompressible flow, it is

shown that jet swirl decays more rapidly

than wake swirl whereas both swirls

decay more rapidly than the non-uniformity

in axial velocity . large swirl

generates axial pressure gradients as well

as large radial pressure gradients,

and therefore alters the streamwise flow .

examples calculated for

incompressible flow, show that the wake is

lengthened by large swirl . it is

expected that this effect will be diminished

in the presence of higher free-stream

mach numbers which lead to decreased

densities, due to decreased

centrifugal effects, decreased radial pressure

gradients, and decreased axial

pressure gradients .

.I 1372

.T

on axially symmetric, turbulent, compressible mixing in the presence

of initial boundary layer .

.A

gdalia kleinstein

.B

polytechnic institute of brooklyn, farmingdale, n.y.

.W

on axially symmetric, turbulent, compressible mixing in the presence

of initial boundary layer .

recent experimental results have shown that the

mixing of heterogeneous gases having an initial velocity

ratio close to unity occurs faster than is predicted by classical eddy-viscosity theory . the theoretical analysis of two uniform streams of different gases but of nearly equal velocity, performed with the usual assumptions for eddy viscosity and prandtl number equal to a constant, shows that mixing will take place very slowly, i.e., at the rate corresponding to laminar diffusion . it has been suggested that the difference between analysis and experiment could be attributed to the presence of a boundary layer in the experiments . it is the purpose of this note to show that the use of the classical eddy-viscosity law, admitting the existence of a boundary layer, is not sufficient to explain the rapid mixing that is observed physically . instead, it is shown that rapid mixing can be explained on the basis of a different eddy-viscosity law, as was suggested in ref. 1 . these conclusions are obtained through application of the analysis presented briefly below .

.I 1373

.T

nose drag in free-molecule flow and its minimization .

.A

tan,h.s.

.B

j.ae.scs. 1959, 360.

.W

nose drag in free-molecule flow and its minimization .

the superaerodynamic nose drag of a body in a free-molecule flow involves two parameters .. the speed ratio s between ordered and random molecular motions (modified mach number), and the temperature ratio between the solid surface and undisturbed gas . simplifications of the drag formula are obtained at hypersonic as well as low-subsonic extremes . to minimize the drag on a nose of specified length and base radius, the ordinary method of calculus of variation was found inadequate . a generalized approach has, accordingly, been developed, and the specification of end conditions is discussed at length .

results of the present investigation indicate that in all cases an optimum nose requires a flat tip . the optimum nose curve for the hypersonic extreme does not depend on the temperature ratio, but that for the low-subsonic extreme varies in the following manner .. for a hot body the curve is convex,. for a cold body, concave . an optimum solution exists in a restricted range of specification only . with prescribed tip and base radii the admissible nose length is bounded below for the cases of hypersonic and low-subsonic hot body and bounded above for the case of low-subsonic cold body . a vanishing tip radius leads to an infinitely long nose in the former and a vanishing nose in the latter case .

optimum nose curves for several temperature ratios at the low-subsonic extreme, as well as the one for hypersonic extreme, are presented . it is observed that at the low-subsonic extreme, with, the hot-body solution asymptotically approaches the hypersonic solution--i.e., a slender conventional warhead with a flat tip,. whereas with, the cold-body solution asymptotically approaches the minimal-surface solution--i.e., tip radius, a flat disc .

.I 1374

.T

theoretical analysis of turbulent mixing of reactive

gases with application to supersonic combustion of

hydrogen .

.A

libby,p.a.

.B

ars jnl. 32, 1962, 388.

.W

theoretical analysis of turbulent mixing of reactive

gases with application to supersonic combustion of

hydrogen .

the turbulent mixing of an axisymmetric jet of a

reactive gas is considered . by assuming a

convenient model for the compressible eddy viscosity,

the momentum equation is reduced to a form

amenable to approximate solution . the resulting

velocity distribution in both incompressible and

compressible flows is in reasonable agreement with

experiment . the usual assumptions with

respect to chemical behavior, namely either frozen or

equilibrium flow, and to unity lewis numbers

and prandtl number are employed . the theoretical

results for chemical equilibrium are shown

to be in reasonable agreement with experimental data

from low speed hydrogen flames . a

numerical example of interest in connection with a hypersonic,

air breathing vehicle is carried out in detail .

.I 1375

.T

an approximate solution for the axisymmetric jet of a laminar

compressible fluid .

.A

gdalia kleinstein

.B

polytechnic institute of brooklyn

.W

an approximate solution for the axisymmetric jet of a laminar

compressible fluid .

an extension of the modified-oseen method of carrier, based on the

linearization of the viscous term of the von mises transformation,

is presented . the method is employed to determine the velocity field

associated with the laminar axisymmetric jet flow of a compressible

gas with an arbitrary but constant external flow . the approximate

solution is shown to be in good agreement with the exact numerical

calculation of pai .

in many boundary layer problems it is not possible to make the

assumption of flow similarity . the solution in these cases can be

obtained either by laborious finite difference techniques or by

resort to approximate solutions . carrier and lewis (1), and more

recently carrier (2), have suggested a method of obtaining approximate

solutions to problems involving convection and diffusion . this

method, termed by carrier /the modified-oseen method/, overcomes an essential difficulty of integral methods, namely, the generation of

reasonable profiles . it is well known that the integral method gives

accurate results only if the analytical profiles represent closely

the true profiles . according to the modified-oseen method the

convective operator in the original partial differential equation

is replaced by a linear one . the resulting equation for the

boundary layer problem is the heat conduction equation which can be

treated by well-known techniques .

it is the purpose of this paper to indicate a modification of this

procedure and to demonstrate its simplicity and accuracy by treating the

axisymmetric laminar flow of a compressible gas with arbitrary but

constant external flow . the modification is based on the use of the

von mises transformation with a subsequent linearization of the viscous

term, rather than the linearization of the convective term . pai's

problem (3), originally treated by a finite difference technique,

is considered to illustrate the effectiveness of this method .

.I 1376

.T

some applications in physics of the p-function .

.A

masters,j.i.

.B

j.chem.phys. 23, 1955, 1865.

.W

some applications in physics of the p-function .

the mathematical background and typical applications

in physics are presented for a recently tabulated

function . because of its properties, the p function should
prove to be a useful aid in the solution of certain
problems in applied mathematics involving surface
integrations in cylindrical coordinates . a tabulation of
the function in its normalized form is appended .
particular attention is paid to the application of the p
function to multiple scattering problems involving
circular symmetry .

.I 1377

.T

theoretical investigation of the flow field about blunt
nosed bodies in supersonic flight .

.A

vaglio-laurin,r. and ferri,a.

.B

j. ae. scs. 25, 1958, 761.

.W

theoretical investigation of the flow field about blunt
nosed bodies in supersonic flight .
  a numerical method ofr obtaining the solution to the inverse
problem of the flow behind a given detached shock to any desired
accuracy is presented . the cases of zero and small incidence
are considered . the combination of sets of such solutions
satisfying prescribed boundary conditions (body shapes) is described .
particular attention is devoted to the analysis of the sonic and
subsonic region of the flow field . convergence and stability of
the stepwise integration from the shock in the elliptic region are

discussed . numerical examples are also included .

.I 1378

.T

blunt-cone pressure distributions at hypersonic mach numbers .

.A

andrew f. burke and james t. curtis

.B

head, research section, and research aerodynamicist, respectively,

applied hypersonic research dept., cornell aeronautical laboratory,

inc., buffalo, n.y.

.W

blunt-cone pressure distributions at hypersonic mach numbers .

the static pressure distributions on the surface of a

blunted 7.5 degree-half-angle cone have recently been experimentally

determined in the cal 48 inch hypersonic shock tunnel . this

facility and the associated instrumentation are described in detail

in ref. 1 . these tests covered a mach number range of 8 to 18

at a reynolds number per foot of approximately 1 times 10 to the 5th

power., the models included one flat-faced cone and two hemispherically

blunted cones .

.I 1379

.T

hypersonic flight and the re-entry problem .

.A

allen, h.j.

.B

j. aerospace sce. 25, 217-227, 1958 .

.W

hypersonic flight and the re-entry problem .

paper reviews the possibilities and some of the main problems of three

types of long-range vehicle, the ballistic, the glide, and the skip

rocket . performance assessments are made on the basis of an ingenious,

if debatable, conversion of the vehicle characteristics to the breguet

formula . the problems of aerodynamic heating, aerodynamic loads and

stability are briefly discussed, and other aspects of hypersonic flight

free molecule flow-dash are touched upon . the results on the whole

favor the glide vehicle for manned flight . the subsequent discussion

on the paper includes references to multistaging and the use of

high-energy fuels .

.I 1380

.T

the problem of obtaining high lift-drag ratios at supersonic speeds .

.A

clinton e. brown and francis e. mclean

.B

langley research center, nasa

.W

the problem of obtaining high lift-drag ratios at supersonic speeds .

the importance of the lift to drag ratio is well

known to all aircraft designers since it gives, to a

great extent, the aerodynamic efficiency of the airplane .

aerodynamic efficiency, however, is only one

component of the grand compromise that a completed

airplane represents . at subsonic speeds, lift-drag ratios of well over 200 have been measured in wind tunnels on airfoil sections., but few powered aircraft have attained (lift to drag ratio) value of 20 . it is invariably true that the requirements of stability and control, structure, and flight operation all contribute to reducing the design (lift to drag ratio) considerably below those exotic values which can be predicted from unrestricted aerodynamic theory . if, however, a certain range or operating efficiency is required, there is most certainly a minimum if we examine the range equation we see that range is proportional to the lift-drag ratio, the thermopropulsive efficiency, and the logarithm of the initial to final weight ratio . the appearance of the lift-drag ratio as a linear factor in the range equation indicates that every attempt should be made to increase (lift to drag ratio)., however, the search for higher (lift to drag ratio) may lead to strange and unorthodox configurations . most frequently, such configurations are ruled out by the adverse effects of their geometry on the weight ratios . in the present paper, we will deal with the maximum lift-drag ratio problem for conventional configurations having a wing and a body in close proximity to each other . no attempt will be made to select a particular configuration as being the best . however, the promising direction to go from the aerodynamic view will be stressed with the understanding that the other factors may outweight the aerodynamics .

.I 1381

.T

effect of mach number on boundary layer transition

in the presence of pressure rise and surface roughness

on an ogive-cylinder body with cold wall conditions .

.A

carros,r.j.

.B

naca rm a56b15, 1956.

.W

effect of mach number on boundary layer transition

in the presence of pressure rise and surface roughness

on an ogive-cylinder body with cold wall conditions .

the effect of mach number variation

from 1.8 to 7.4 on boundary-layer

transition was investigated on a slender

fin-stabilized ogive-cylinder

body in free flight at a constant length

reynolds number of 13.8 million .

the wall to free-stream temperature ratio

was constant at a value of 1.0

below mach number 4.5 and at a value of

of the test showed that increasing mach

number had a very favorable effect

of increasing the extent of the laminar

boundary layer for a given surface

roughness . the transition data, when

plotted as a function of a factor indicative of heat transfer, showed that heat transfer was possibly responsible for a good deal of the increase in transition reynolds number with mach number .

transition was found to occur farther forward on the sheltered side of the body than on the windward side for angles of attack as low as 0.4 and for all mach numbers . the pressure rise along sheltered-side stream-lines was examined and it was found that the pressure-rise coefficient at the transition point, showed no variation with mach number . data from other sources for different test conditions, when reduced to values of pressure-rise coefficient, were also found to correlate well with that of the present investigation with the exception of data at low subsonic mach numbers . these present results also show that mach number, surface roughness, pressure rise, and length reynolds number all affected boundary-layer transition in the region of theoretical infinite laminar stability to

small two-dimensional disturbances as

calculated for a flat plate with

zero pressure gradient .

.I 1382

.T

the solution of the equations of the laminar boundary

layer for schubauer's observed pressure distribution

for an elliptic cylinder .

.A

hartree,d.r.

.B

arc r + m.2427, 1949.

.W

the solution of the equations of the laminar boundary

layer for schubauer's observed pressure distribution

for an elliptic cylinder .

the solution of the equations of the laminar

boundary layer has been carried out for the pressure

distribution for an elliptic cylinder of axial ratio 2.96 .. 1 with

its major axis in the direction of the incident stream .

the solution has been obtained by the method of hartree and

womersley . in applying this method the derivatives

parallel to the boundary are replaced by finite differences, so

that the partial differential equation of the boundary

layer is replaced by an ordinary equation relating the velocity

distribution through the boundary layer at one section

to that at another, at an interval upstream . by two independent
integrations covering the same range by finite intervals
of different sizes, it is possible to estimate the errors
involved in replacing the derivatives by finite differences,
and so to correct for these errors .
the process of solution requires the values of the pressure
gradient along the solid boundary, and there is a certain
tolerance in the derivation of the pressure gradient distribution
from a limited number of observed values of pressure .
an analysis of schubauer's pressure distribution is outlined, and
the results were used for the main solution calculated .
it is found that the solution, for the distribution of pressure gradient
so derived, does not give separation of the boundary
layer from the solid boundary, whereas the actual flow does separate .
it is found that the calculated solution is very sensitive to
the pressure distribution, and a comparatively small
modification of the pressure distribution gives a solution which
does indicate separation close to the point at which
separation is observed to occur . the solution with this pressure
distribution also gives very good agreement with the
observed velocity distribution through the boundary layer at
points upstream from separation .

.I 1383

.T

on the theory of laminar boundary layer involving separation .

.A

vonkarman,t. and millikan,c.b.

.B

naca r.504, 1934.

.W

on the theory of laminar boundary layer involving separation .

the paper presents a mathematical discussion of the laminar boundary layer, which was developed with a view of facilitating the investigation of those boundary layers in particular for which the phenomenon of separation occurs . the treatment starts with a slight modification of the form of the boundary layer equation first published by von mises . two approximate solutions of this equation are found, one of which is exact at the outer edge of the boundary layer while the other is exact at the wall . the final solution is obtained by joining these two solutions at the inflection points of the velocity profiles . the final solution is given in terms of a series of universal functions for a fairly broad class of potential velocity distributions outside of the boundary layer . detailed calculations of the boundary layer characteristics are worked out for the case in which the potential velocity is a linear function of the distance from the upstream stagnation point . finally the complete separation point characteristics are determined for the boundary layer associated with a potential velocity distribution made up of two linear functions of the distance from the stagnation point . it appears that extensions of the detailed calculations to more complex potential flows can be fairly easily carried out by using the explicit formulae given in the paper .

.I 1384

.T

a theoretical calculation of the laminar boundary layer

around an elliptic cylinder and its comparison with

experiment .

.A

millikan,c.b.

.B

j. ae. scs. 3, 1936, 91.

.W

a theoretical calculation of the laminar boundary layer

around an elliptic cylinder and its comparison with

experiment .

the author, in conjunction with th. von karman,

has recently given a new method of approximate

integration of the prandtl boundary layer equations,

which was developed in order to treat cases in which

separation of a laminar boundary layer might be

expected . the method was developed because some doubt

was felt as to the accuracy with which the well-known

pohlhausen analysis would describe conditions in the

neighborhood of such a separation point . numerical

calculations were carried out for certain cases involving

theoretical simplifications, and very considerable

discrepancies were found between the results of the new

and pohlhausen methods . the method was also used in developing a theory for the maximum lift coefficient

of certain classes of airfoils . this theory gave

satisfactory agreement with experiment but no direct

experimental check on the boundary layer analysis itself has

been given up to the present .

.I 1385

.T

air flow in a separating laminar boundary layer .

.A

schubauer,g.b.

.B

naca r.527, 1935.

.W

air flow in a separating laminar boundary layer .

  the speed distribution in a laminar boundary layer on

the surface of an elliptic cylinder, of major and minor

axes 11.78 and 3.98 inches, respectively, has been

determined by means of a hot-wire anemometer . the direction

of the impinging air stream was parallel to the major axis .

special attention was given to the speed distribution in

the region of separation and to the exact location of the

point of separation . an approximate method, developed

by k. pohlhausen for computing the speed distribution,

the thickness of the layer, and the point of separation, is

described in detail,. and speed-distribution curves

calculated by this method are presented for comparison with

experiment . good agreement is obtained along the forward part of the cylinder, but pohlhausen's method fails

shortly before the separation point is reached and

consequently cannot be used to locate this point .

the work was carried out at the national bureau of

standards with the cooperation and financial assistance

of the national advisory committee for aeronautics .

.I 1386

.T

analysis and calculation by integral methods of laminar

compressible boundary layer with heat transfer and

with and without pressure gradient .

.A

morduchow,m.

.B

naca r.1245, 1955.

.W

analysis and calculation by integral methods of laminar

compressible boundary layer with heat transfer and

with and without pressure gradient .

a survey of integral methods in laminar-boundary-layer

analysis is first given . a simple and sufficiently accurate

method for practical purposes of calculating the properties

layer in an axial pressure gradient with heat transfer at the

wall is then presented . for flow over a flat plate, the method

is applicable for an arbitrarily prescribed distribution of

temperature along the surface and for any given constant

prandtl number close to unity . for flow in a pressure gradient, the method is based on a prandtl number of unity and a uniform

wall temperature . a simple and accurate method of

determining the separation point in a compressible flow with an adverse

pressure gradient over a surface at a given uniform wall

temperature is developed . the analysis is based on an extension of

the karman-pohlhausen method to the momentum and thermal

energy equations in conjunction with fourth- and especially

higher degree velocity and stagnation-enthalpy profiles . from

the equations derived here, conclusions regarding the effect of

pressure gradient, mach number, and wall temperature on the

boundary-layer characteristics are derived and illustrated . in

particular the effects on skin-friction, heat-transfer coefficient,

separation point in an adverse pressure gradient, and stability

of the laminar boundary layer are analyzed .

.I 1387

.T

the buckling of a square panel under shear, when one

pair of opposite edges is clamped and the other pair

is simply supported .

.A

leggett,d.m.a.

.B

arc r + m.1991, 1941.

.W

the buckling of a square panel under shear, when one

pair of opposite edges is clamped and the other pair

is simply supported .

reasons for investigation.--for an efficient design
of spar with thin sheet web it is important to know
the load which will just cause the web to buckle . as stiffeners
divide the web into panels, it is required to find the
buckling stress of rectangular panels bounded on two sides by
spar flanges and on the other two sides by stiffeners .
boundary conditions which represent closely this type of edge
fixing are clamping (along the flanges) and simple support
critical shear stress for a square panel held in this way .

conclusions and further development.--it is found that the
value of the critical shear stress is almost midway between
its values when all four edges are clamped and all four edges
are simply supported .

the method of solution developed in this report is of very
general application, and can be used to investigate the
stability of rectangular panels when the loading is any combination
of shear and compression or tension, and the edges
are clamped or simply supported, and not necessarily all clamped
or all simply supported . by an easy extension the
method of solution can also be used to find the periods of transverse
vibration of rectangular panels for the same types
of loading and edge fixing .

.I 1388

.T

a process for the step-by-step integration of differential
equations in an automatic digital computing machine .

.A

gills,s.

.B

proc. cam. phil. soc. 47, 1951, 96.

.W

a process for the step-by-step integration of differential

equations in an automatic digital computing machine .

it is advantageous in automatic

computers to employ methods of integration

which do not require preceding function values

to be known . from a general theory given by

kutta, one such process is chosen giving

fourth-order accuracy and requiring the minimum

number of storage registers . it is developed into

a form which gives the highest attainable

accuracy and can be carried out by comparatively

few instructions . the errors are studied and

a simple example is given .

.I 1389

.T

numerical construction of detached shock waves .

.A

garabedian,p.r.

.B

j.math.phys. 36, 1957, 192.

.W

numerical construction of detached shock waves .

this article proposes a new method for solving the problem

of the detached shock wave . if the shock wave is assumed
known, a cauchy problem for a system of partial differential
equations arises . this has been solved by several authors
in the region where the system is elliptic (near the peak of
the shock wave) . considering the plane stationary case, the
author seeks an analytic continuation of the propagation
function (x,y) in the complex plane y = y1 + y2, x real .
in the plane (x, y2) the propagation function satisfies a
hyperbolic equation (near the peak of the shock wave) . a
new cauchy problem is solved and the solution of the original
problem obtained by analytic continuation . a numerical
example is treated with many details .

.I 1390

.T

on the numerical calculation of detached bow shock
waves in hypersonic flow .

.A

garabedian,p.r. and liebersten,h.m.

.B

j. ae. scs. 25, 1958, 109.

.W

on the numerical calculation of detached bow shock
waves in hypersonic flow .

a method is described for calculating examples of hypersonic
flow with a detached bow shock wave past a bluff axially
symmetric body . the form of the shock wave is assumed, and the
analysis is based on a cauchy problem for the stream function

in the subsonic region, where the motion is governed by a
partial differential equation of elliptic type . through analytic
continuation into the complex domain, the cauchy problem is
reformulated in such a manner that it becomes properly set in the
subsonic region . this leads to a stable scheme for computation
of the flow by finite differences . numerical examples at
freestream mach number 5.8 are presented in which the flow is
determined throughout the subsonic region, and, in particular, the
detachment distance, the location of the sonic line, and the
pressure distribution along the body are calculated . these results are
in excellent agreement with experimental data obtained at the
california institute of technology .

.I 1391

.T

shock wave and flow field development in hypersonic re-entry .

.A

probstein, r.f.

.B

ars. j. 31, 185-194, 1961 .

.W

shock wave and flow field development in hypersonic re-entry .
a study is made of when and how a shock wave and continuum-type flow
field develop in the nose region of a highly cooled blunt body re-
entering the atmosphere at hypersonic speed and in a free molecular flow
regime . the various types of flow regimes encountered down to low
altitude conditions are delineated, and the nature of the flow field and
behavior of some of the aerodynamic characteristics are discussed . it

is shown that for a highly cooled body, free molecule flow conditions
occur at a higher altitude than previously indicated . based on
available evidence, it is suggested that kinetic theory solutions, which are
essentially modified free molecule results, along with the navier-stokes
equations with no surface slip, serve to define all of the flow regimes
except for a narrow transitional layer regime which has a height of
less than one factor of 10 in free stream density change . it is also
suggested that the appearance of a definable shock wave occurs very
rapidly in terms of density change near the beginning of the
transitional layer regime, and that its location, as in continuum flow, is
governed principally by the body geometry, whereas its thickness is
determined by a local mean free path .

.I 1392

.T

the solution of small displacement, stability or vibration
problems concerning a flat rectangular panel when the
edges are either clamped or simply supported .

.A

hopkins,h.g.

.B

arc r + m.2234, 1945.

.W

the solution of small displacement, stability or vibration
problems concerning a flat rectangular panel when the
edges are either clamped or simply supported .

this report describes an energy method for
the exact solution of problems concerning the small

displacements, stability or vibration of a flat rectangular panel when the edges are either clamped or simply supported . the influence of stiffeners which are parallel to one pair of edges, and situated in pairs on opposite sides of the panel so that the neutral axis of each stiffener pair lies in the middle surface of the panel, is taken into account . the method is not only applicable to isotropic panels but also to aeolotropic panels when the material of the panel has two directions of elastic symmetry parallel to the edges .

the final solution of the problems depends on an infinite set of linear equations for small displacement problems or on an infinite determinantal equation for stability and vibration problems . the important feature of the analysis given is that it enables a direct approach to be made to these equations in any particular problem . it is not in general possible to obtain a direct solution of the final equations and it is necessary to approximate and consider a finite set of linear equations or a finite determinantal equation derived from the more important terms in the analytical expression for the transverse displacement of the panel . here, physical intuition and, if available, experimental data serve as a guide and the accuracy of the final results so obtained is gauged by the rate of convergence with the increase in the number of terms considered .

the general method of solution is applied first to the free vibration of a square panel when all the edges are clamped, and second to the buckling of a square panel under shear when three edges are clamped and one edge is simply supported .

.I 1393

.T

heat transfer near the forward stagnation point of

a body of revolution .

.A

sibulkin,m.j.

.B

j. ae. scs. 19, 1952, 570.

.W

heat transfer near the forward stagnation point of

a body of revolution .

in order to determine the temperature distribution over a

body moving through the atmosphere, a knowledge of the

local heat-transfer coefficients is required . for slender

sharp-nosed bodies, the heat-transfer coefficients are frequently

approximated by using the comparable flat-plate values .

however, for blunt-nosed bodies, flat-plate solutions are not

applicable near the forward stagnation point . since the greatest rate

of heat transfer may occur at the forward stagnation point, its

value should be investigated . in this note a theoretical solution

is given for the heat transfer near the forward stagnation point of

a body of revolution assuming laminar, incompressible, low-speed

flow . the comparable solution for two-dimensional flow has

been given by squire . in the case of a blunt-nosed body moving

with supersonic velocity, the flow behind the central portion of

the bow wave is subsonic, and it is possible that a low-speed

solution, using as /free-stream/ conditions those behind the center
of the bow wave, will apply near the stagnation point .

.I 1394

.T

stagnation point heat transfer measurements in hypersonic
low density flow .

.A

neice,s.e., rutkowski,r.w. and chann,k.k.

.B

j. ae. scs.27, 1960, 387.

.W

stagnation point heat transfer measurements in hypersonic
low density flow .
in hypersonic, low reynolds number flow around a blunt body,
the boundary-layer thickness approaches the shock-layer
thickness (shock standoff distance) within the region of
continuum flow . in this instance, the customary boundary-layer
approximations no longer apply . hoshizaki and probstein
have obtained solutions to the incompressible navier-stokes
equations in the stagnation region of a blunt body in this
hypersonic low reynolds number flow . the results indicate that
heat-transfer rates are substantially higher than those predicted by
incompressible boundary-layer theory . probstein indicated that
the actual heat-transfer rates would be correspondingly higher
than the predictions of fay and riddell . these findings are of
particular importance in the atmospheric entry phase of
recoverable satellites .

.I 1395

.T

low density stagnation point heat transfer measurements

in the hypersonic shock tunnel .

.A

wilson,m.r. and wittliff,c.e.

.B

ars jnl. 32, 1962, 275.

.W

low density stagnation point heat transfer measurements

in the hypersonic shock tunnel .

presents absolute heat-transfer measurements, using resistance

thermometer, for hemisphere-cylinder at mach numbers 9.2 to 11.2 .

results show vorticity-interaction and viscous-layer effects

increase heat transfer above values predicted by boundary-layer

theory . data are correlated using cheng's reynolds-number-

dependent parameter . investigation covers vorticity-interaction to

incipient-merged-layer regimes (free-stream unit reynolds numbers

.I 1396

.T

shear buckling of clamped and simply-supported infinitely

long plates reinforced by transverse stiffeners .

.A

cook,i.t. and rockey,k.c.

.B

aero. quart. 13, 1962, 41.

.W

shear buckling of clamped and simply-supported infinitely
long plates reinforced by transverse stiffeners .

the paper presents a solution to the buckling of infinitely long plates clamped along the edges, together with an extension of the solution obtained by stein and fralich for the case when the edges are simply-supported . it is shown that, as a consequence of increasing the edge support from that of a simple support to one in which both deflection and rotation are prevented, the rigidity which an intermediate transverse stiffener must possess in order to support the plate effectively is much reduced . agreement between the theoretical relationships and existing experimental data is good .

.I 1397

.T

critical shear stress of an infinitely long simply
supported plate with transverse stiffeners .

.A

stein,m. and fralich,r.w.

.B

naca tn.1851, 1949.

.W

critical shear stress of an infinitely long simply
supported plate with transverse stiffeners .

a theoretical solution is given for the critical shear stress of an infinitely long, simply supported, flat plate with identical, equally spaced, transverse stiffeners of zero torsional stiffness . results are obtained by means of the lagrangian multiplier method and are presented in the form of design charts . experimental results are included and are found to be in good agreement with the theoretical results .

.I 1398

.T

stability of rectangular plates under shear and bending

forces .

.A

way,s.

.B

j. app. mech. 3, 1936, a131.

.W

stability of rectangular plates under shear and bending

forces .

the author first discusses the problem of a plane, simply supported rectangular plate loaded by shearing forces in the plane of the plate on all four edges . there are two stiffeners attached one third and two thirds of the way along the plate . the critical load is calculated for various stiffener rigidities . also, the rigidity necessary to keep the stiffeners straight when the plate buckles is found . this stiffener rigidity is found to be slightly larger than that necessary for a plate with one stiffener and the same panel dimensions as the plate with two stiffeners .

the second problem discussed by the author is that of a plane, simply supported rectangular plate loaded by uniformly distributed edge shearing forces in the plane of the plate and linearly distributed tension and compression in the plane of the plate at the ends . the end forces vary from tension, at one corner to, at the other corner,

so that their resultant is a bending moment . the

presence of the edge shearing forces is found to diminish the

critical bending stress in this case . calculations are made

for various magnitudes of bending and shearing forces for

plates of various proportions .

.I 1399

.T

buckling of transverse stiffened plates under shear .

.A

wang,t.k.

.B

j.app.mech. 3,1947, a269.

.W

buckling of transverse stiffened plates under shear .

this paper presents an analysis of buckling of simply

supported rectangular plates reinforced by any number of

transverse stiffeners and subjected to shearing forces

uniformly distributed along the edges . two cases are

considered .. (a) the case of a plate with a finite length,.

ing stresses in both cases are expressed in similar forms,

that is, in equation (13), and

k in equation (24), respectively . design

curves are drawn as shown in figs. 2,3, and 5 .

.I 1400

.T

the buckling shear stress of simply-supported infinitely long plates with transverse stiffeners .

.A

kleeman,p.w.

.B

arc r + m.2971, 1953.

.W

the buckling shear stress of simply-supported infinitely

long plates with transverse stiffeners .

this report is an extension of previous theoretical

investigations of the elastic buckling in shear of flat

plates reinforced by transverse stiffeners . the plates are treated

as infinitely long and simply-supported along the long

sides . stiffeners are spaced at regular intervals, dividing the plate

into a number of panels of uniform size . the effect

ob bending and torsional stiffnesses of the stiffener upon the buckling

shear stress is calculated for the complete range

of stiffnesses, for panels with ratios of width to stiffener spacing of

graphical forms .