Aircraft Stability and Control Short Course

Savannah Section of AIAA

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966-4486

SESSION I - Introduction and Basic Aircraft Aerodynamics

+ Disclaimers

+ Course Content

+ Common Terms

+ Attendence and Short Quiz

+ Introduction to Aircraft Stability and Control

  - Some Good Books

  - Role of Stability and Control Staff in Aircraft Development

  - Simple Definitions of Stability, Control, and Maneuverability

  - Airworthiness, Flying Qualities, Handling Qualities

+ Break: Video of Maneuvering Flight with Tom Jones and Sukhoi 26

+ Basic Aircraft Aerodynamics

  - Gas is Fluid, the Potential Flow Model and Fluid Boundaries

  - Gas is Real, Viscous Behavior and Reynold's Scaling

  - Gas is Compressible, Shock Waves and Mach Number

  - Operation of the Airfoil, Dynamic Pressure, Normalized Forces and Moments

  - Airfoil Behavior with Angle of Attack

  - Varying and Enhancing Lift (Flaps)

  - Behavior of the Finite Wing

  - Downwash, Tip Vortices, and Induced Drag, Sectional Char.

  - Swept Wings and Compressibility

  - Low Aspect Ratio Wings

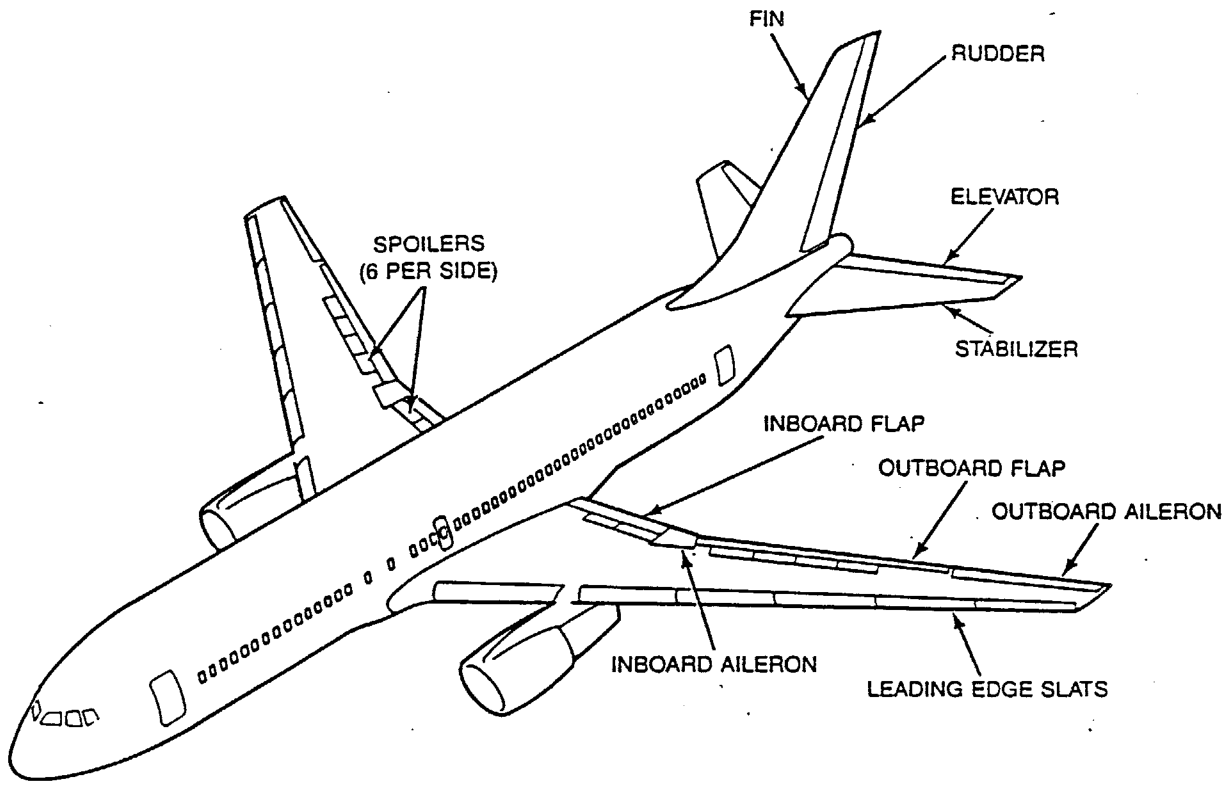
  - Wing-Body Aerodynamics

  - Ground Effect

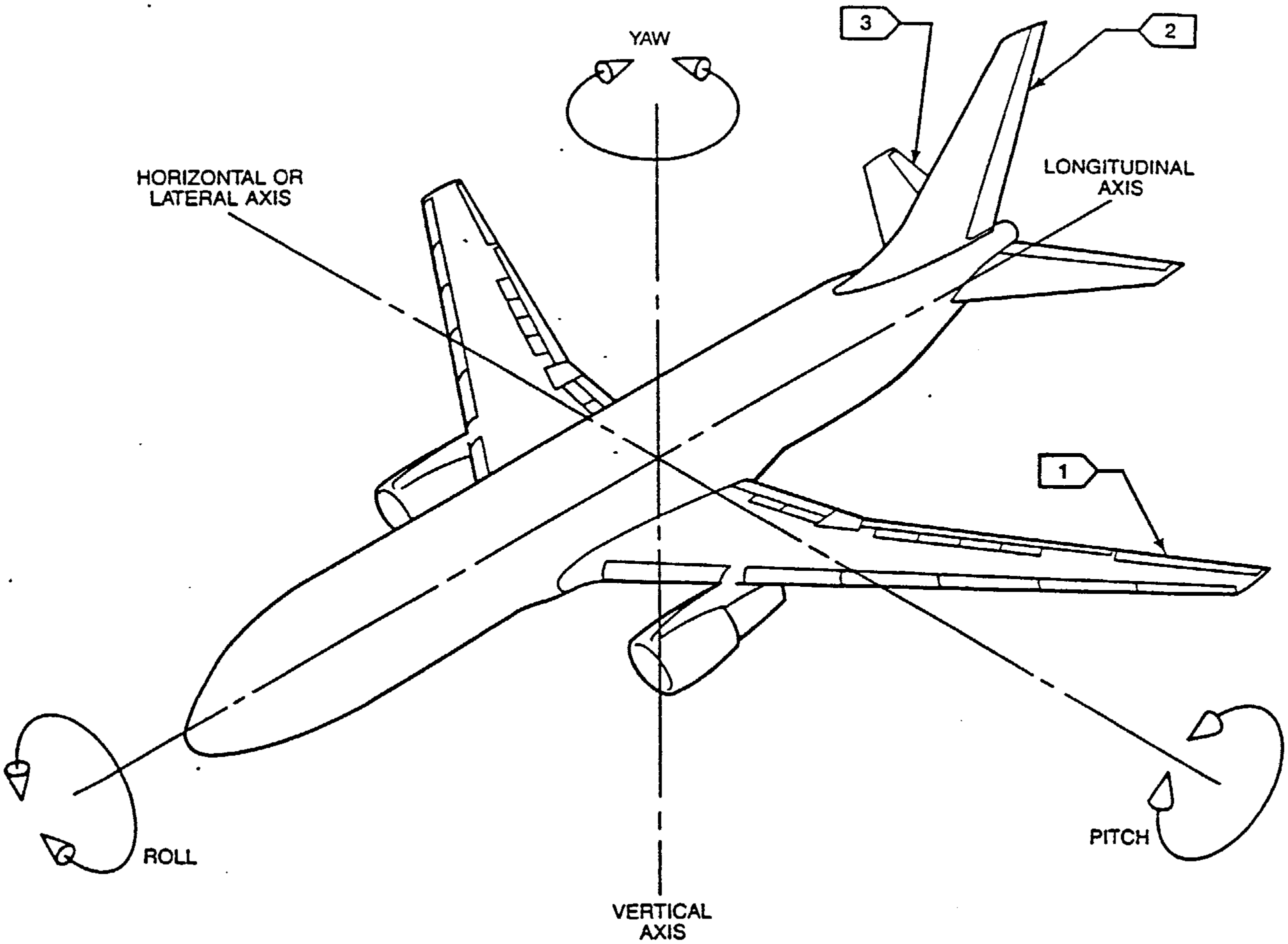
  - Standard Atmosphere, Airspeed Definitions

COMMON TERMS

CONTROL SURFACES



CONTROL AXIS DIAGRAM



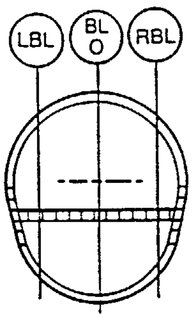
1 AILERON—THE HINGED SECTION OF THE TRAILING EDGE OF THE LEFT AND RIGHT WINGS THAT OPERATE IN SERIES TO PROVIDE LATERAL CONTROL. WHEN ONE AILERON IS RAISED, THE OPPOSITE IS LOWERED, PRODUCING ROLLING MOVEMENTS AROUND THE LONGITUDINAL AXIS.

2 RUDDER—THE HINGED OR MOVABLE AUXILIARY AIRFOIL ATTACHED TO THE VERTICAL FIN TO CONTROL YAW.

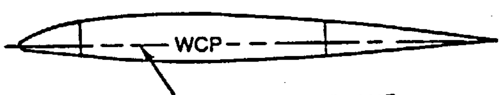
3 ELEVATOR—THE HINGED SECTION OF THE HORIZONTAL STABILIZER USED TO CONTROL PITCH.

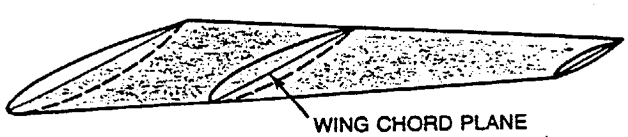
ROLLING MOVEMENTS ARE A FUNCTION OF BOTH AILERONS AND FLIGHT SPOILERS.

BUTTOCK LINE. A vertical reference line or plane parallel to the centerline of the airplane used to locate points or planes to the left or right of the airplane centerline.

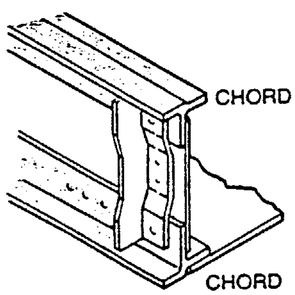


CHORD PLANE, WING. The plane that defines the planform of the wing and around which the airfoil is figured. The wing chord plane scribes a line from the extreme point of the leading edge to the extreme point of the trailing edge, thus giving a datum line to measure incidence and dihedral.

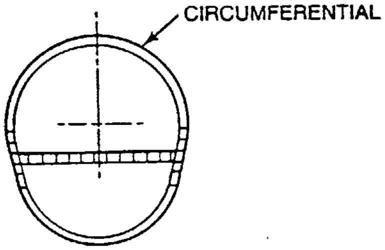




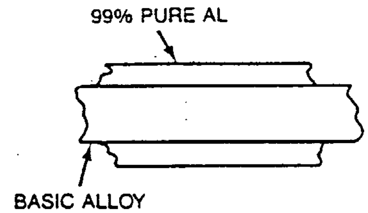
CHORD (structural). Sometimes called a cap. A strong member that forms the edges of beam structures or heavy frames.



CIRCUMFERENTIAL. A frame that is shaped to the fuselage.



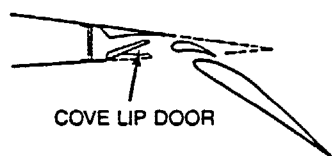
CLAD. A 99% pure aluminum layer, molecular-bonded to the basic alloy by rolling while heated.



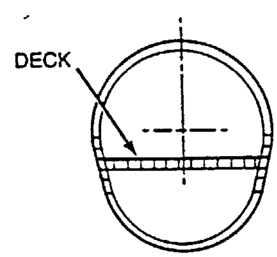
CLIP. Sometimes called bracket. Usually a small angle used to attach lightweight parts such as wiring clamps.

COMPOSITE MATERIAL. Composites are considered combinations of material differing in composition or form. The constituents retain their identities in the composite; that is, they do not dissolve or otherwise merge completely into each other although they act together. Normally, the components can be identified physically.

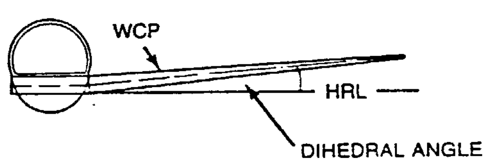
COVE LIP DOOR. A movable door on the under surface of the wing, hinged at the rear spar, that lifts upward when the flaps are lowered. These doors allow high-pressure air to flow through the main flap slots. Used on KC-135, 707, and 720 airplanes.



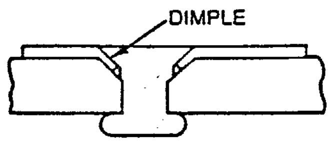
DECK. The horizontal floor in the control cabin or passenger cabin. The horizontal structure to support fuselage tanks in the B-52 (fuel deck).



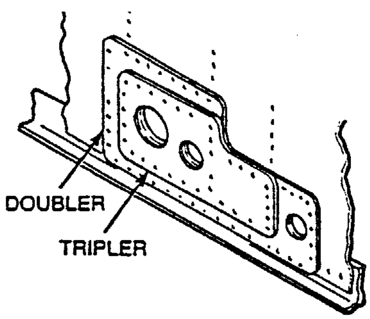
DIHEDRAL. The angle the wing chord plane makes with a horizontal reference plane.



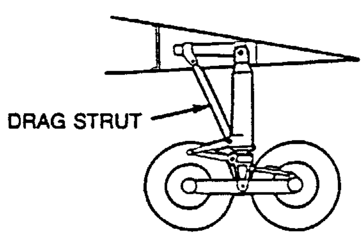
DIMPLE. A depression of the area around the edges of a hole in thin sheet to provide for a countersunk rivet.



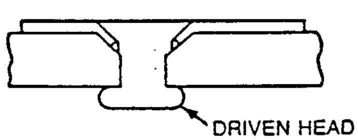
DOUBLER. A second sheet or plate installed next to the web or skin in a small area subject to high local loads to provide a double thickness of material. A tripler is a third sheet to provide three layers of material.



DRAG STRUT. A diagonal brace attached to the forward end of the landing gear trunnion and the lower end of the oleo strut. Absorbs drag loads during ground maneuvers and braking.

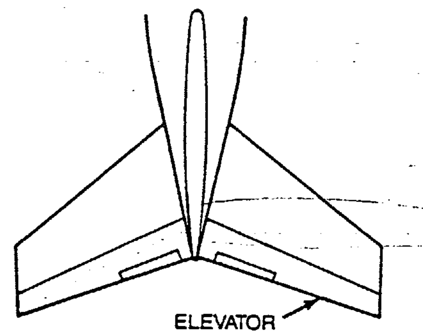


DRIVEN HEAD. The upset portion of a rivet shank that has been hammered flat by the bucking bar during installation.



DUTCH ROLL. A phenomenon peculiar to sweptwing aircraft. A continuous combination of yaw and roll.

ELEVATOR. The hinged section of the horizontal stabilizer used to control pitch.



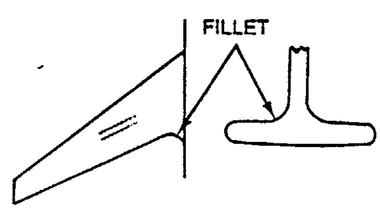
EMPENNAGE. The aft portion of an aircraft, usually consisting of a group of stabilizing planes or fins, to which control surfaces such as elevators and rudders are attached.

EXTRUSION. A part formed by squeezing the material through a die that has a hole cut to the desired cross-sectional shape of the part.

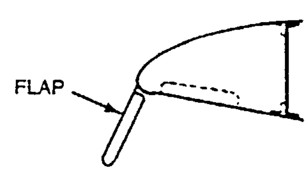
FAIRING. An auxiliary structural member shaped to provide a smooth flow of air and reduce drag.

FAYING SURFACE. A surface that fits, joins, or unites closely with an adjacent surface overlapping it.

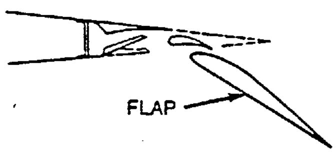
FILLET. A filler that smooths the angle formed by two intersecting surfaces and eliminates an abrupt change of direction. Used on forged or machined parts to prevent stress concentration at the "corner." Used aerodynamically to eliminate angular joints between components.



FLAP, LEADING EDGE. Hinged section of the under side of the leading edge that, when extended, prevents airflow separation over the top of the wing. Leading edge flaps hinge at the leading edge of the airfoil.



FLAP, TRAILING EDGE. Hinged section of the trailing edge of the wing that can be lowered and extended. When lowered, flaps increase airplane lift at low speeds.

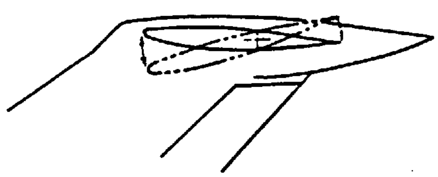


FLAP TRACK. A steel track on which the main landing flaps operate by means of rollers. The curvature of the flap track determines the deflection and position of the landing flaps when they are extended.

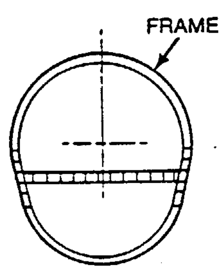
FLAT PATTERN. The overall shape or outline of a sheet metal part before bending operations.

FLYING TAIL. A horizontal stabilizer that is movable and controllable. The entire horizontal tail angle of incidence can be changed to trim the airplane.

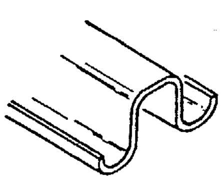
HORIZONTAL TAIL MOVEMENT, 727



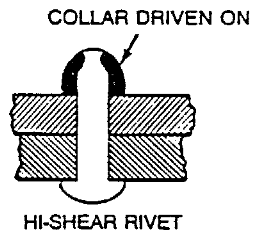
FRAME. A circumferential structural member in the body that supports the stringers and skin. Used in semimonocoque construction (see MONOCOQUE).



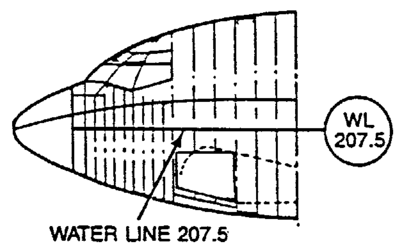
HAT SECTION. The cross-section shape of the stringers used in the fuselage. A common rolled shape that looks like a top hat with the brim curled up.



HI-SHEAR RIVET. Trade name for high-shear-strength steel fasteners used in the airplane where heavy loads are encountered. Installed with a swaged collar instead of being upset by a bucking bar. Used in shear applications.

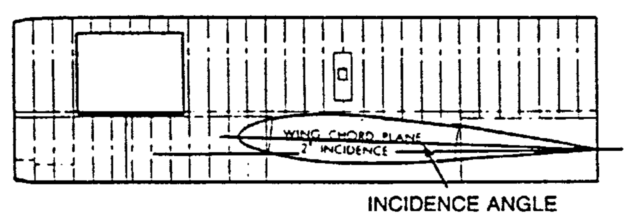


HRL (horizontal reference line). Will sometimes refer to a water line or can be a special horizontal line to locate a particular plane or points in the airplane's horizontal axis (see WATER LINE).

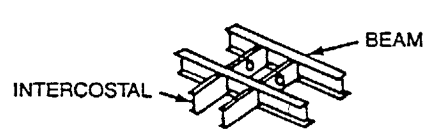


INBOARD. A term applying to the inside. An item nearest to the fuselage (antonym: OUTBOARD).

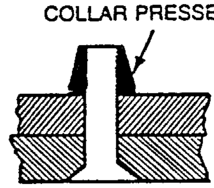
INCIDENCE, ANGLE OF. The fixed angle at which the wing chord plane is set relative to the horizontal datum line of the aircraft. Sometimes erroneously called the angle of attack; angle of attack rightfully refers to the angle of the entire aircraft to relative wind. The angle of attack can be changed by the elevators on the horizontal tail surfaces.



INTERCOSTAL. A small stabilizing beam between and at right angles to larger beams or bulkheads.

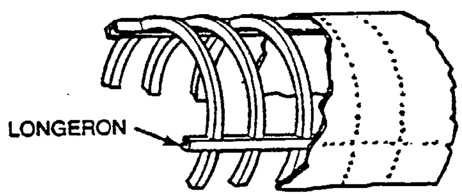


LOCK BOLT. A high-strength steel fastener with a swaged collar on the shank for retention rather than a nut. Used in tension and shear applications.



LOFT LINE. The line or lines that establish and control the shape of an object so that all intersecting cutting planes are smoothly faired.

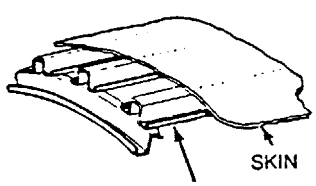
LONGERON. A principal longitudinal member of the framing of an aircraft fuselage or nacelle. Usually continuous across a number of points of support.



MACH NUMBER. A number representing the ratio of the speed of a body to the speed of sound in the surrounding atmosphere. For subsonic speed, the mach number is less than 1 and for supersonic speed it is greater than 1.

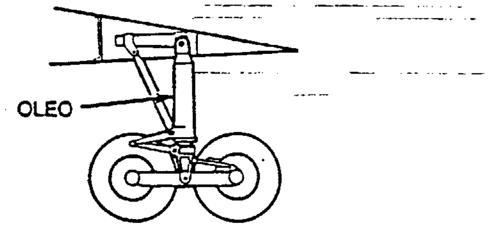
MLG. Abbreviation for main landing gear.

MONOCOQUE. A single-shell construction in which the skin carries all shear and bending stresses. In semimonocoque construction, shear and bending loads in the skin are transmitted to stringers and frames.



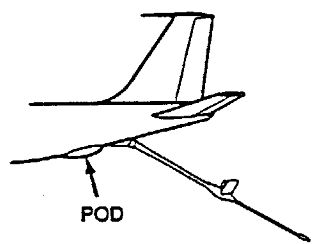
OIL CAN. A term commonly used to describe a buckling or wrinkling in the metal skin of an airplane. The skin normally should be smooth.

OLEO STRUT. A main weight-carrying strut in the landing gear that absorbs the shock of landing by the flow of oil through an orifice in the cylinder of the strut.



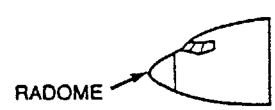
PITOT-STATIC. An airspeed indicating system that operates from ram air pressure in the pitot tube and static pressure of the atmosphere. Gives an airspeed reading that is corrected for altitude.

POD. A term sometimes used for engine nacelle. Indicates an enclosure such as the boom operator's pod on the KC-135 that encloses the boom operator and equipment in a streamlined fairing.

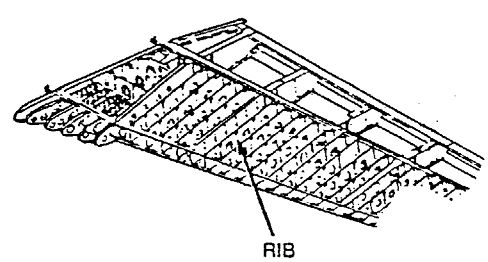


PRESSURE WEB. A web that seals an area to retain cabin pressurization.

RADOME. Coined term for radar dome. A nonmetallic streamlined fairing to cover the radar sweep.

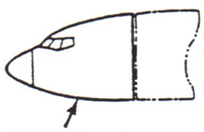


RIB. A fore and aft member of an airfoil structure (wing or aileron) of an aircraft used to give the airfoil section its form and to transmit that load from the skin to the spars.



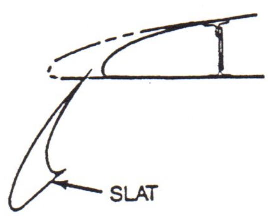
RUDDER. A hinged or movable auxiliary airfoil, attached to the vertical fin, that controls yaw.

SECTION. Any of the large subassemblies of the airplane that are built separately and then joined to form the complete airplane. The airplane is built in sections to ease production and handling problems.

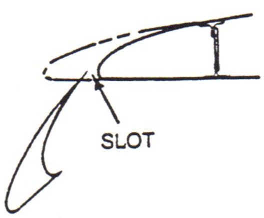


SKIN. The outside covering of an aircraft.

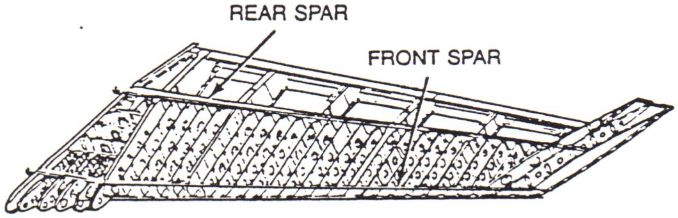
SLAT. A movable auxiliary airfoil attached to the leading edge of the wing. When closed, it forms part of the normal contour of the wing; when opened, it forms a slot and increases lift.



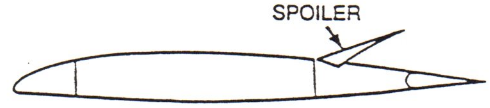
SLOT. An elongated passage through a wing whose primary function is to improve the airflow over the wing at high angles of attack.



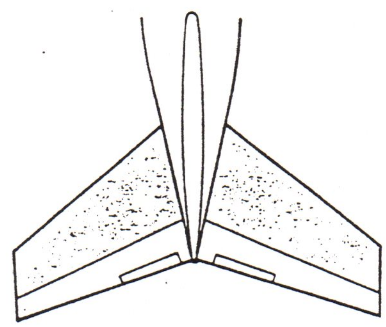
SPAR. A principal spanwise beam in the structure of a wing, stabilizer, rudder, or elevator. It is usually a primary load-carrying member in the structure.



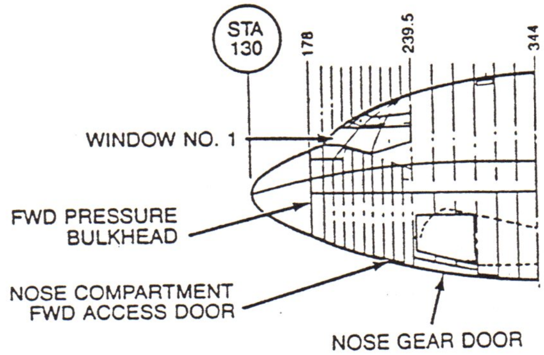
SPOILER. A hinged panel on the upper surface of a wing that "spoils" wing lift when raised. Left and right spoilers can be raised alternately for high-speed lateral control or can be raised together as speed brakes during landing.



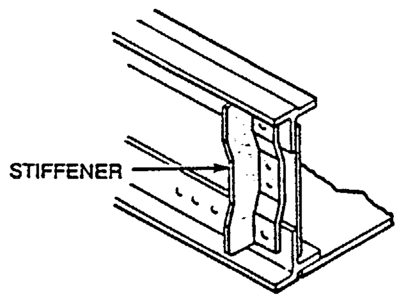
STABILIZER. A fixed horizontal tail surface that maintains stability around the lateral axis of an aircraft.



STATION LINE. All parts of an airplane are identified by a location or station number in inches from a beginning point. Station lines in the fuselage start forward of the nose; those for the wing usually start at the centerline of the fuselage. This forms a locating system that divides the aircraft cross-sectionally into a series of reference planes at right angles to the vertical centerline of the aircraft.

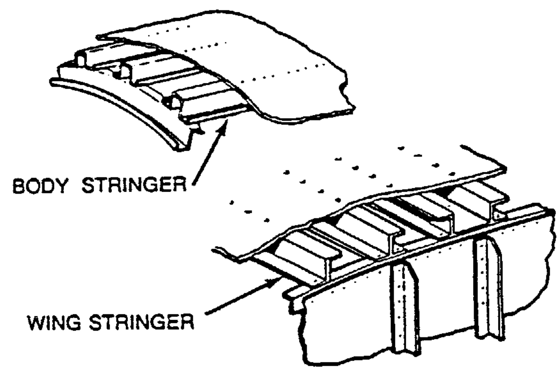


STIFFENER. A metal part, other than flat sheet, formed or extruded and used in the framing of a structure to provide rigidity.

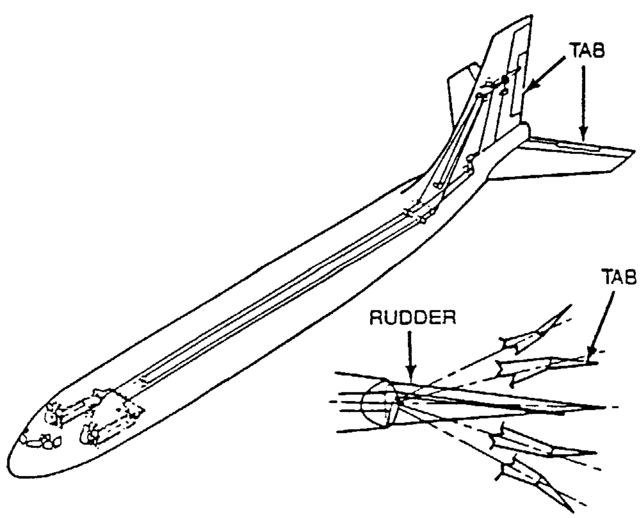


STRETCH FORM. A method used to shape skins or parts by stretching the flat sheet over a die to provide the shape.

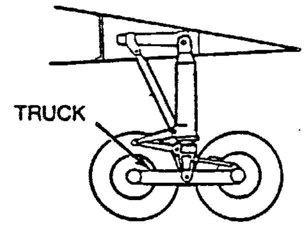
STRINGER. Longitudinal members in the fuselage or spanwise members in the wing to transmit skin loads into the body frames or wing ribs.



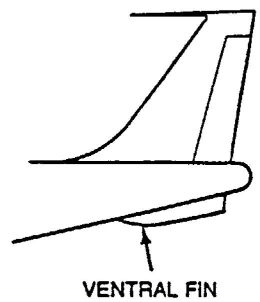
TAB. A small, hinged, auxiliary control surface attached to a primary control surface such as an aileron, rudder, or elevator. When deflected, it moves the primary surface to which it is attached. The primary surface will react in the direction opposite the control tab's deflection.



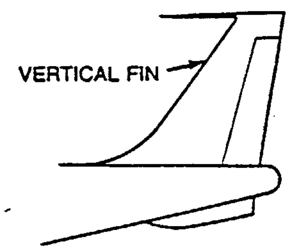
TRUCK. The portion of the main landing gear that is composed of a swiveling beam with an axle and two wheels on each end.



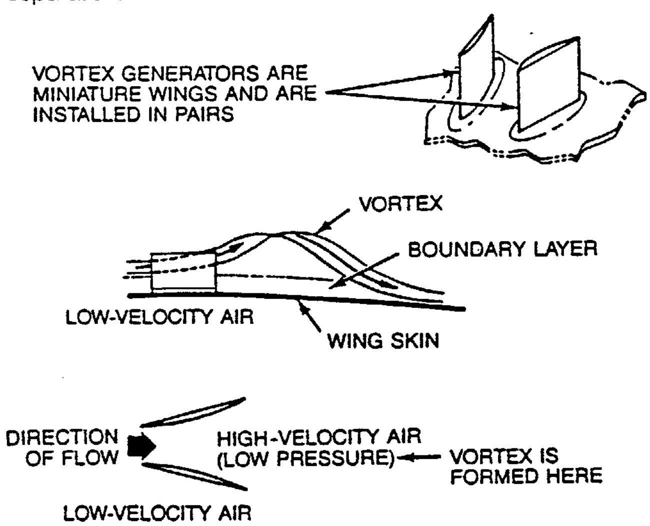
VENTRAL FIN. A stabilizing surface attached to the bottom of the fuselage near the tail.



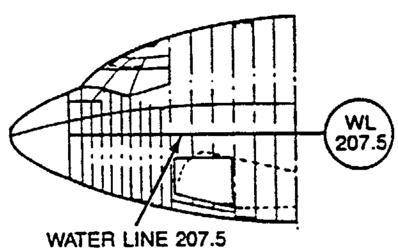
VERTICAL FIN. Sometimes referred to as vertical stabilizer. It is fixed to provide directional stability. The trailing edge is hinged to form the rudder.



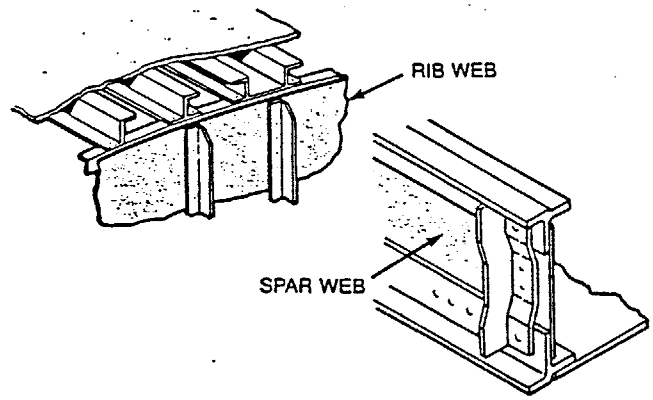
VORTEX GENERATOR. A device used on the wings and tail surfaces to decrease drag caused by the separation of the air flowing over the flight surfaces. Vortex generators appear as a row of small metal tabs set at angles to the air stream. The vortex formed by the tabs pushes the air down to the skin of the flying surface and delays drag producing separation.



WATER LINE. A reference line or horizontal plane parallel to the ground used to locate points vertically.



WEB. A thin-gage plate of sheet, when supported by stiffening angles and framing, provides great shear strength for its weight. Used in many applications throughout an aircraft because of its strength-to-weight ratio.



INTRODUCTION TO AIRCRAFT STABILITY AND CONTROL

Some Good Books

Davies, D. P.
"Handling the Big Jets (3rd Edition)"
Brabazon House, Red Hill, Surrey, England, 1971

Durand, W. F. (Chief Editor)
"Aerodynamic Theory: A General Review of Progress"
Peter Smith, Gloucester, Mass., 1976
Originally 6 volumes published 1934 to 1936

Etkin, B.
"Dynamics of Atmospheric Flight"
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Roskam Aviation and Engineering, Lawrence, Kansas, 1971

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"Stability and Control of Airplanes and Helicopters"
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Dover, New York, 1988

STABILITY AND CONTROL STAFF ROLE IN AIRCRAFT DEVELOPMENT

STABILITY & CONTROL GROUP

FUNCTION

The stability and control group is responsible for the handling characteristics of the airplane. This includes defining control requirements and the definition of the airplane's static and dynamic stability. This group's efforts ensures that the airplane flying qualities (how the pilot flies the airplane) are acceptable.

Stability and Control (S&C) engineers work closely with members of the configuration group during the preliminary design phases of an airplane. The S&C engineer must determine the proper sizing of the horizontal and vertical empennage as well as the sizing of the elevator, rudder, aileron and spoiler control surfaces. Empennage sizing determines the airplane's stability and control levels. Control surfaces are sized to ensure adequate control authority for normal maneuvers (e.g. takeoff rotation) and for system failures or unusual flight situations.

The location of the wing and landing gear is affected by S&C considerations. Wing placement is determined by airplane loadability, forward c.g. control and aft c.g. stability considerations. The placement of the landing gear is a compromise between aft c.g. ground handling and forward c.g. takeoff rotation considerations. Thus, stability and control requirements have a strong influence on the configuration of the airplane.

Extensive wind tunnel testing is conducted during the development of an airplane configuration. The S&C engineer uses the wind tunnel for a variety of developmental functions. This includes the validation of tail sizes and stability levels as well as control surface sizing and controllability levels. Control surface hinge moments must also be determined. The wind tunnel also provides the data for the development of the engineering and crew training flight simulators. The preparation of the aerodynamic data base for the simulator is primarily an S&C responsibility.

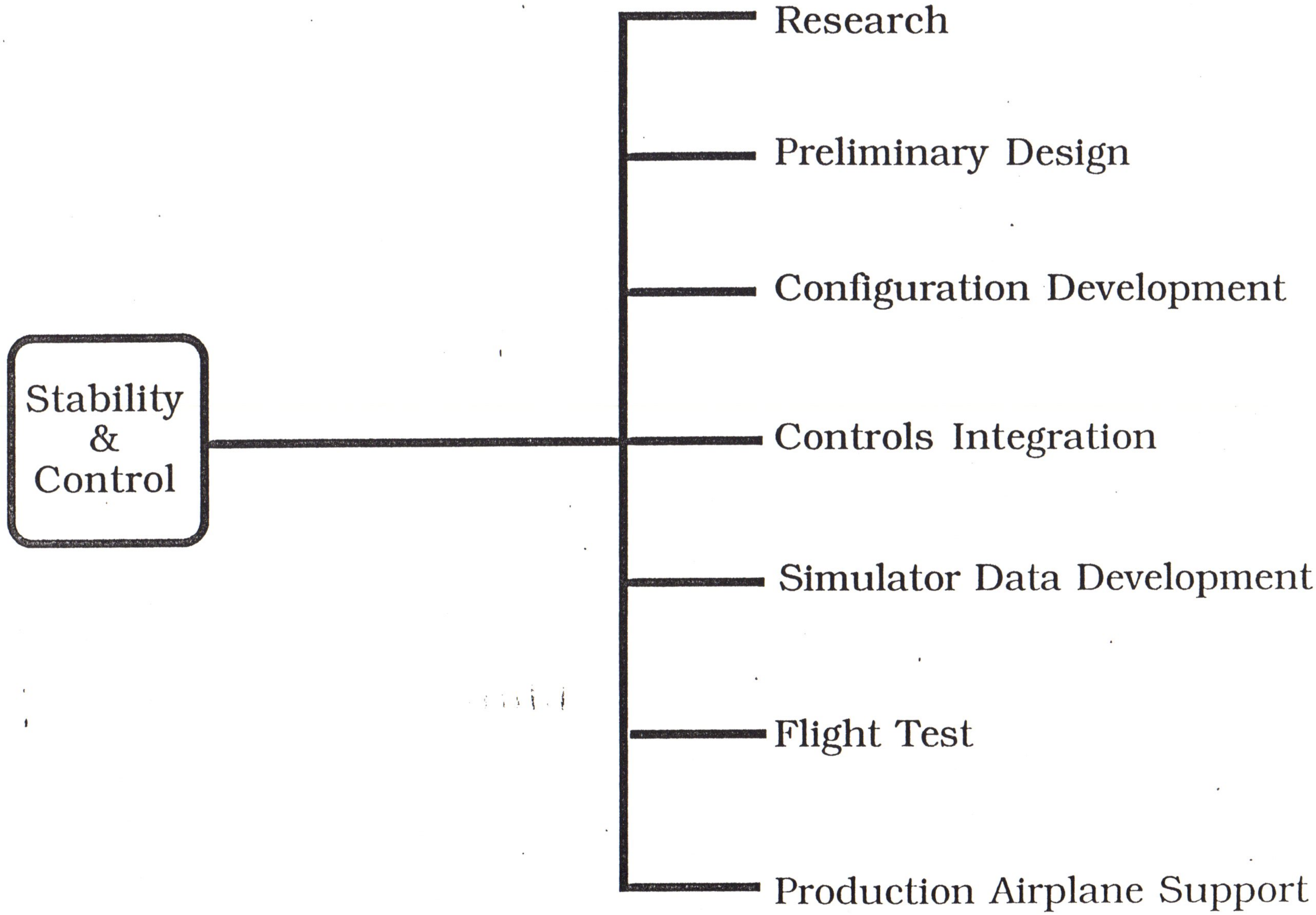
The engineering simulator has many uses. It allows the evaluation of airplane handling qualities with a pilot "in-the-loop" and the analysis of failure conditions. The development of flight systems such as autopilots and yaw dampers relies heavily on the simulator. The simulator allows initial pilot training for a new airplane program and the development of the flight test plan. Critical failures, which could not be demonstrated in flight, are certified using the simulator. Finally, the simulator may be used for the purpose of accident investigation.

In addition to participating in the development of the airplane configuration, the S&C engineer is involved in the integration of the flight control systems. Flight characteristics requirements influence the design of hydraulic and electrical power distribution and the size of the control surface actuators. These requirements are based on operation of the systems in both normal and failure modes. The control system integration effort also includes the evaluation of flight control laws and controller design.

The Stability and Control group is heavily involved the flight testing and certification of the airplane. Pre-certification testing includes verification of stability levels, controllability, handling qualities and minimum control speeds. Flight data are collected for the purpose of updating the predicted aerodynamic data base of the flight simulator. An accurate data base is required if the simulator is to be certified for crew training purposes. Certification testing of the airplane demonstrates compliance with FAR Part 25 regulations and the regulations of foreign certification agencies.

Once an airplane enters service, support is provided to the airline customers. Questions relative to the flight characteristics of the airplane are answered, support of crew training flight simulators is provided, accident/incident investigations are conducted, etc.

STABILITY & CONTROL



Stability & Control

Preliminary Design

- Work Hand-In-Hand with Aero Configuration
- Tail Sizing
  - Horizontal, Determines Stability Levels
  - Vertical, Influences Engine Location and Minimum Control Speed
- Control Sizing
  - Elevator
  - Rudder
  - Ailerons
  - Spoilers
- Wing / Gear Location
  - Affects Airplane Loadability
  - Affects Longitudinal Control / Stability Requirements

Stability & Control

Configuration Development

- Work Hand-In-Hand with Aero Configuration and Performance
- Define Wind Tunnel Test Requirements
- Perform Wind Tunnel Configuration Development and Validation Tests
- Use Wind Tunnel Data to:
    - Validate Tail Sizes, Stability Levels
    - Validate Control Sizes, Controllability Levels
    - Assist in Wing / Gear Location Balance Trade Studies
    - Define Control Surface Hinge Moments
    - Assess Static and Dynamic Handling Qualities
    - Assess Criticality of Control Failures
    - Develop Flight Simulator Data Base
- Design Requirements

Stability & Control

Controls Integration

- Hydraulic and Electrical Power Distribution
- Actuation
- Failure Effects
- Control Law Evaluation
- Controller Evaluation
- Design Requirements

Stability & Control

Flight Simulator

- Engineering Simulator
  - Handling Qualities Evaluation with Pilot-in-Loop
  - Failure Analyses
  - System Development (Augmentation On and Off)
  - Initial Pilot Training
  - Development of Flight Test Program
  - FAA Certification (Critical Failures)
  - Accident Investigation
  - Basis for Training Simulator Package

Flight Simulator

- Crew Training Simulator
  - Airline Crew Training
  - Boeing Crew Training
  - Pre-Certification Phase II Simulator for Initial Crew Training
  - Full Phase II & III Requires Matching and Validation to Flight Data

FAA Advanced Simulation Plan

Phase	Training credit	System Requirements
I	• Recency of Landings • Night Takeoff & Landings • Proficiency Check Landings	• Aero Data Must Match Airplane With Emphasis on Ground Effects • Night Visual System • 3-D Motion system
II	• Transition and Upgrade Training Pilot can receive airplane type rating without flying the airplane.	**Phase I Plus:** • Crosswind and Windshear • Ground Handling • Contaminated Runway • Control Feel Dynamics • Dusk Visual System • 6 DOF Motion • Special Cockpit Sounds
III	• All pilot training and certification can be done in the simulator.	**Phase II Plus:** • Characteristic Buffet Motion • Icing, Mach and Reverse Thrust Effects • Daylight Visual • Enhanced Cockpit Sounds

Stability & Control

Flight Test

- Pre-Certification Tests to Verify:
  - Stability Levels
  - Controllability
  - Handling Qualities
  - Minimum Control Speeds

- Certification Tests to Demonstrate:
  - Compliance with FAR Part 25 Regulations
  - Compliance with CAA, D of A, etc. Regulations

- Flight Data for Simulator Validation

Stability & Control

Production Airplane Support

- Fleet Support
  - Problem Airplane Assistance
  - Permission to Ferry
- Production Support
  - Rejection Tags
  - Support Boeing Flight Tests
- Accident / Incident Investigation

Aerodynamics / Stability & Control Organizational Interfaces

Aerodynamics
Stability & Control

Project Division

- Avionics / Flight Systems
- Mechanical / Electrical
- Structures - Design
- Test Integration
- Configurations

Outside

- FAA Certification
- NASA W.T.'s
- Vertol W.T.
- UWAL W.T.
- Japanese, NAL W.T.'s
- Airlines (Simulator Customers)
- Simulator Manufacturers

Engineering Division

Technology

- Aerodynamic Config. and Performance
- Aero S&C Research
- Aero W.T. Labs
- Propulsion
- Structures Loads & Flutter
- Weights

Other

- Airworthiness
- Air Safety

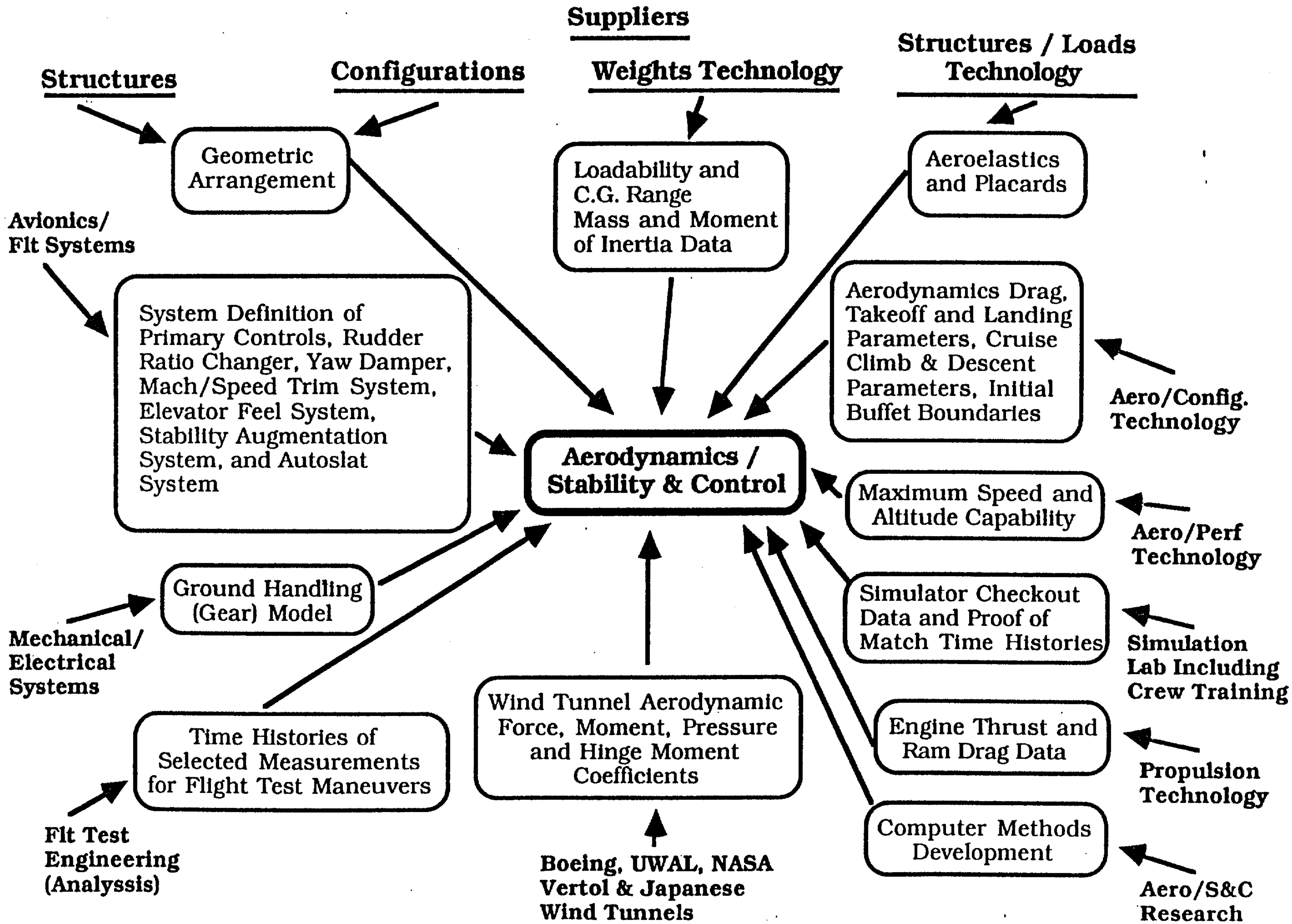
Customer Services

- Flight Test Engineering
  - Analysis
  - Operations
  - FAA Coordiantion
- Simulator Data & Engineering
- Crew Training
- Flight Operations

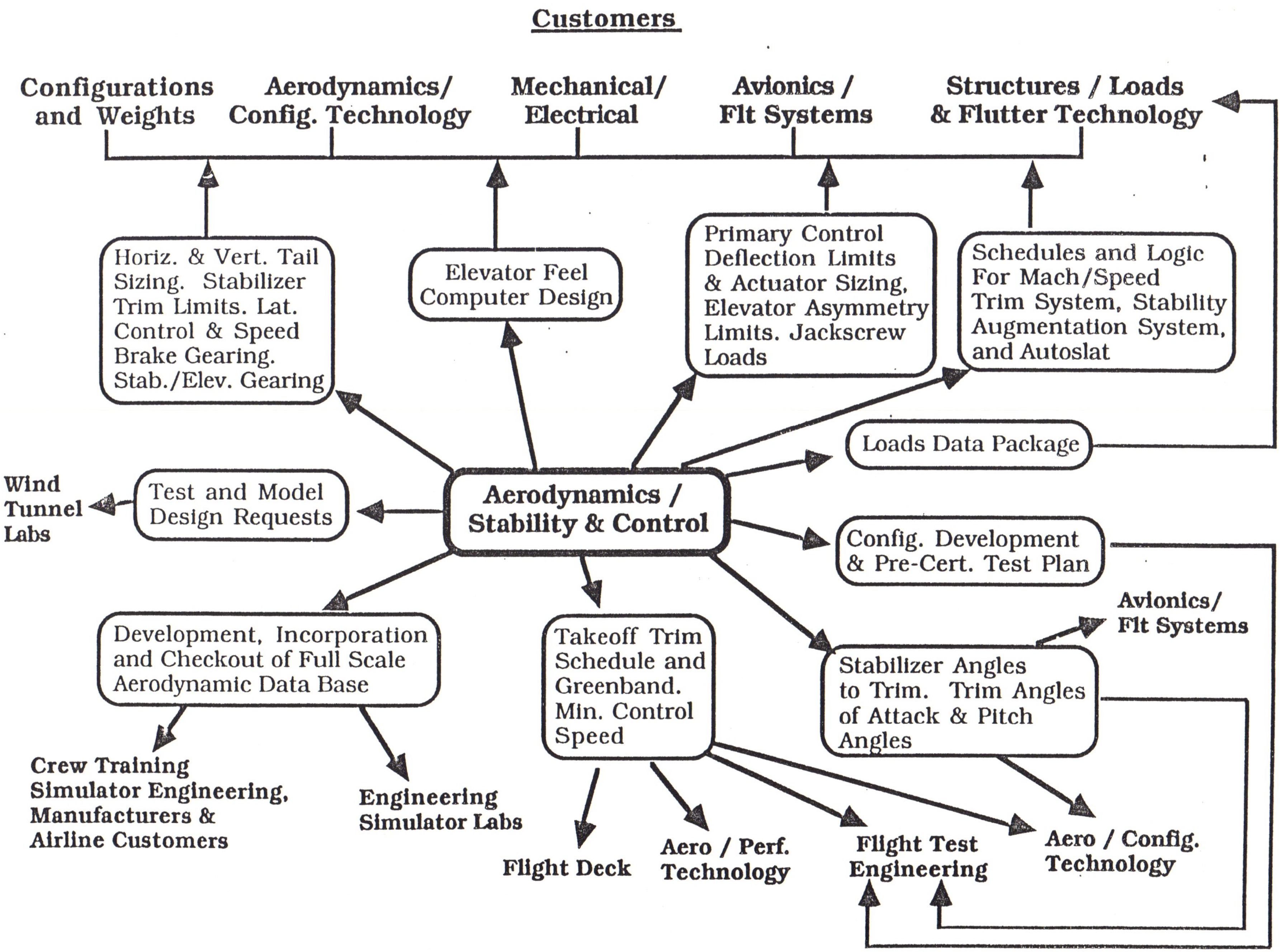
Computer Services

- Flight Simulation Labs
- Aerodynamic S&C Computing
- Crew Training Simulators
- Computing - IGDA

Major Data Flow to Aerodynamics / Stability & Control



Major Data Flow From Aerodynamics / Stability & Control



DEFINITION OF STABILITY, CONTROL, AND MANEUVERABILITY

Stability — The ability of a vehicle to return to an equilibrium state after a disturbance.

+ Return to a trim speed after disturbance by a wind gust
+ Tendancy to return to level flight after initiating a turn

Control — The ability to demand a change of vehicle state.

+ Nose over from incipient stall
+ Takeoff rotation
+ Change trim speed and climb angle

Maneuverability — The ability to control the rate of change of vehicle state.

+ Turn rate
+ Takeoff rotation rate
+ Roll rate

AIRWORTHINESS
FLYING QUALITIES
HANDLING QUALITIES

Airworthiness — Safety of flight in normal operation and under adverse conditions or with reasonable system failures.

+ Sustained flight with engine failure
+ Sustained flight with secondary structure failure
+ Capability to land with landing gear extension failure

Flying Qualities — Response of aircraft state characteristics in returning to an equilibrium state or responding to control commands

+ Effect of control friction on speed trim deadband
+ Response to step change in elevator deflection
+ Response to engine failure
+ Trim change for throttle slam or flap extension

Handling Qualities — Apparent response of aircraft with pilot control, including control forces and other feedback; the ability to carry out specific tasks under control

+ Stick force per g
+ Control applications for engine failure
+ Control applications with jammed surface

BASIC AIRCRAFT AERODYNAMICS

GAS FLOW

Gas is Fluid

Gas flow is similar to "ideal fluid flow"

"Ideal fluid flow" is mathematically precise potential flow that satisfies:

1. Continuity (Conservation of Mass)
2. Conservation of Momentum

Basic Equations:

$$\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \vec{V}) = 0 \qquad \text{Conservation of Mass}$$

$\rho \equiv$ density

$t \equiv$ time

$\vec{V} \equiv$ velocity vector

∇ operator : $\nabla = \vec{i}\frac{\partial}{\partial x} + \vec{j}\frac{\partial}{\partial y} + \vec{k}\frac{\partial}{\partial z}$

for ideal flow $\frac{\partial \rho}{\partial t} = 0 \quad \Delta \rho = 0$

$$\Rightarrow \quad \nabla \vec{V} = 0$$

$$\frac{\partial \vec{V}}{\partial t} + (\vec{V} \cdot \nabla)\vec{V} = -\frac{1}{\rho}\nabla p \qquad \text{Conservation of Momentum}$$

Euler's Equation

$p \equiv$ pressure

$$\frac{\partial \vec{\omega}}{\partial t} + (\vec{V} \cdot \nabla)\vec{\omega} = 0$$

$$\vec{\omega} = \nabla \times \vec{V} = \begin{vmatrix} \vec{i} & \vec{j} & \vec{k} \\ \frac{\partial}{\partial x} & \frac{\partial}{\partial y} & \frac{\partial}{\partial z} \\ u & v & w \end{vmatrix} \qquad \text{Vorticity}$$

$\Rightarrow$ Vorticity is constant along a streamline

Properties of Vorticity:

1. Continuity to boundaries of flow
2. Does not support pressure differences
3. Joins boundary at the perpindicular

Flow About A Body

Described by velocity potential and stream functions.

$$\vec{V} = \nabla\phi \qquad u = \frac{\partial\phi}{\partial x},\ v = \frac{\partial\phi}{\partial y},\ w = \frac{\partial\phi}{\partial z}$$

Velocity Potential

Reduced Euler Equations $\nabla\cdot\vec{V} = 0$

$\nabla\times\vec{V} = 0$

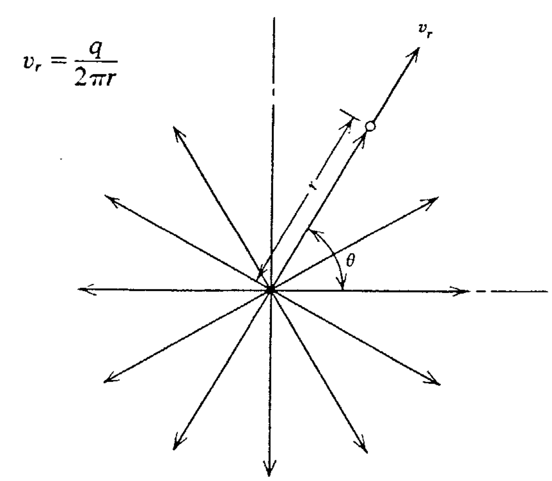
Boundary Condition $\vec{V}\cdot\vec{n} = 0$

$\vec{n} \equiv$ surface normal vector

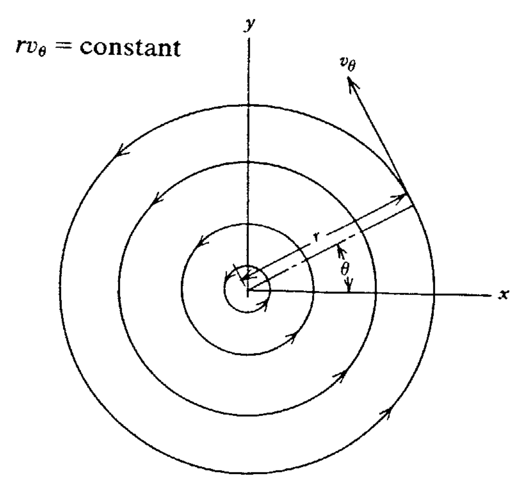
Define stream function

$$u = \frac{\partial\psi}{\partial y},\ v = -\frac{\partial\psi}{\partial x} \Rightarrow \nabla^2\psi = 0$$

These functions facilitate modeling of ideal flow by sources, sinks, and vortices

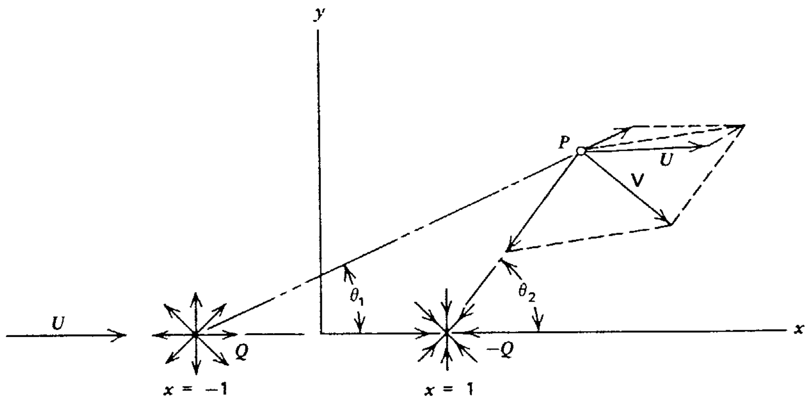


Flow from a source.



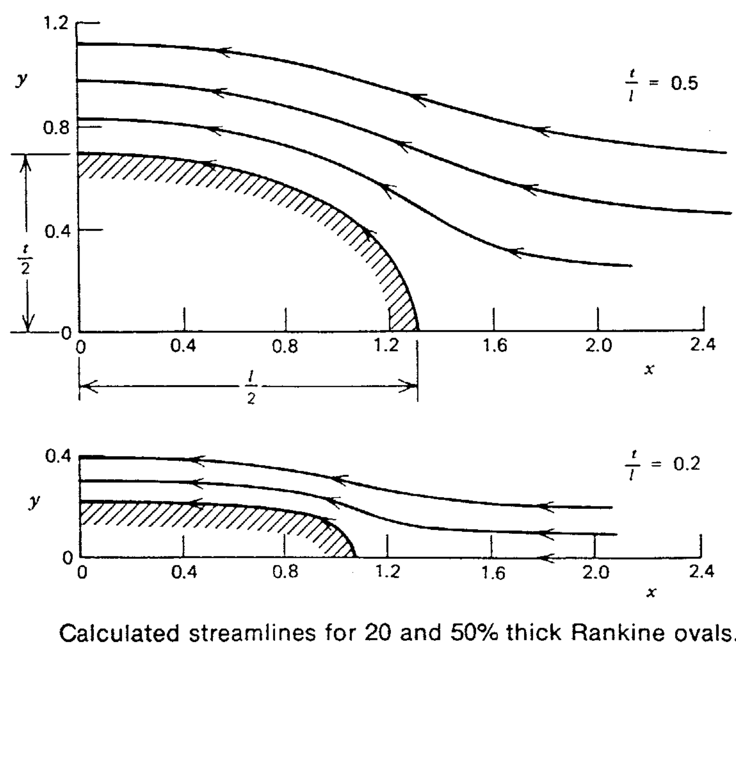
Flow field around a vortex.

A simple combination of source, sink, and uniform flow produces a Rankine Oval surface boundary. Other shapes may be modeled by increasing the number of sources and sinks and varying strengths.



Source-sink combination in a uniform flow.

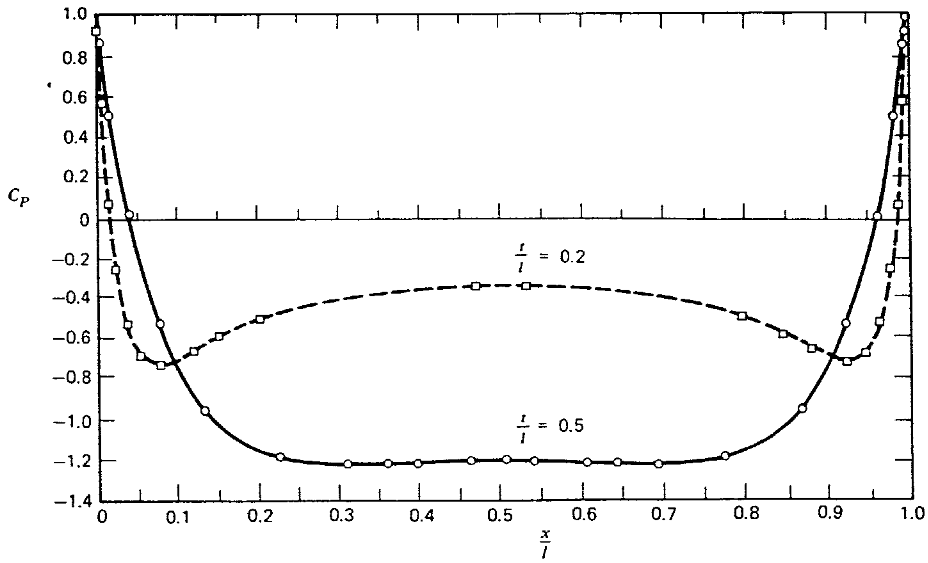
$$x = \left\{1 - y^2 - \frac{2y}{\tan \pi[1 - 2(U/q)(y - y_\infty)]}\right\}^{1/2}$$



Calculated streamlines for 20 and 50% thick Rankine ovals.

Finally, we are able to use Bernoulli's equation to compute surface pressures, normalized by:

$$C_p = \frac{p - p_0}{(1/2)\rho V_0^2}$$



Predicted pressure distributions for 20 and 50% thick Rankine ovals.

Net forces and moments are integral of pressure ~~vector~~ on surface and surface normal vector.

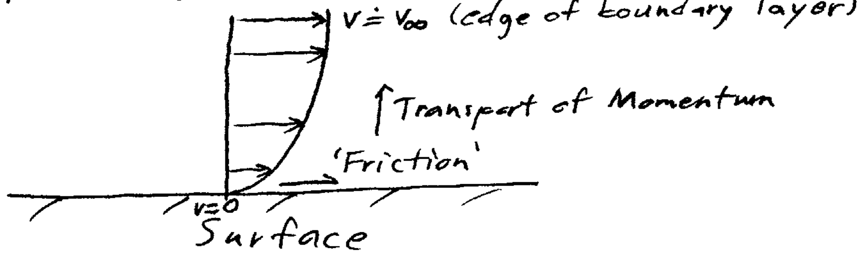
Disappointment?

- In ideal flow the net forces and moments on a body are zero.

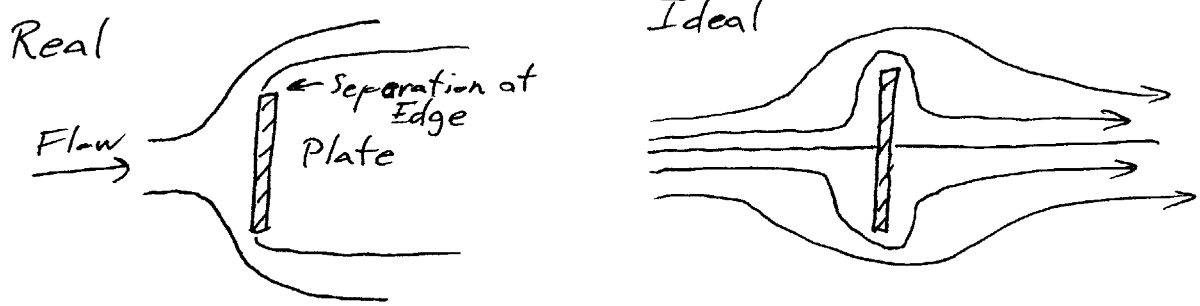
Gas is Real

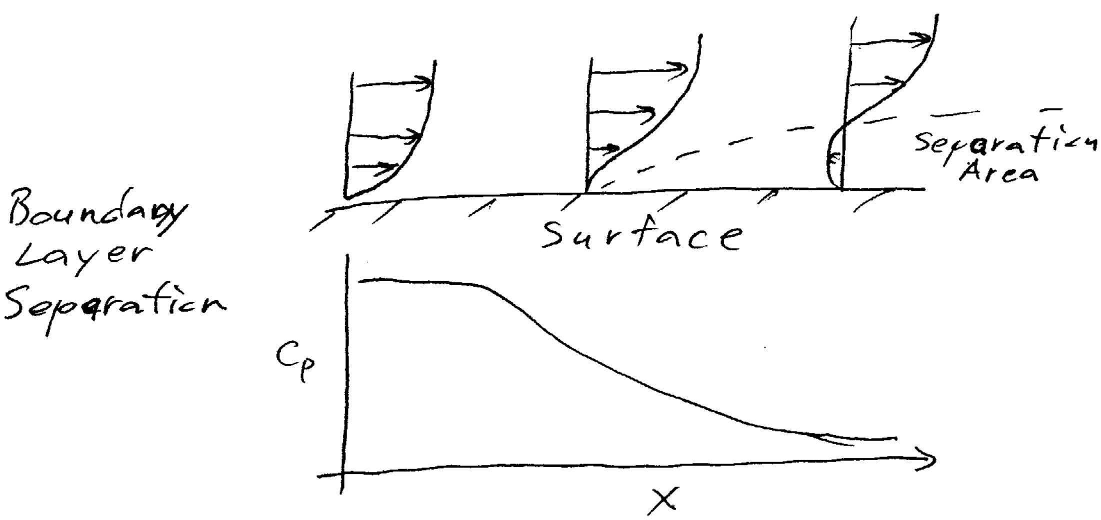
+ Gas has material properties that violate "ideal fluid" concepts

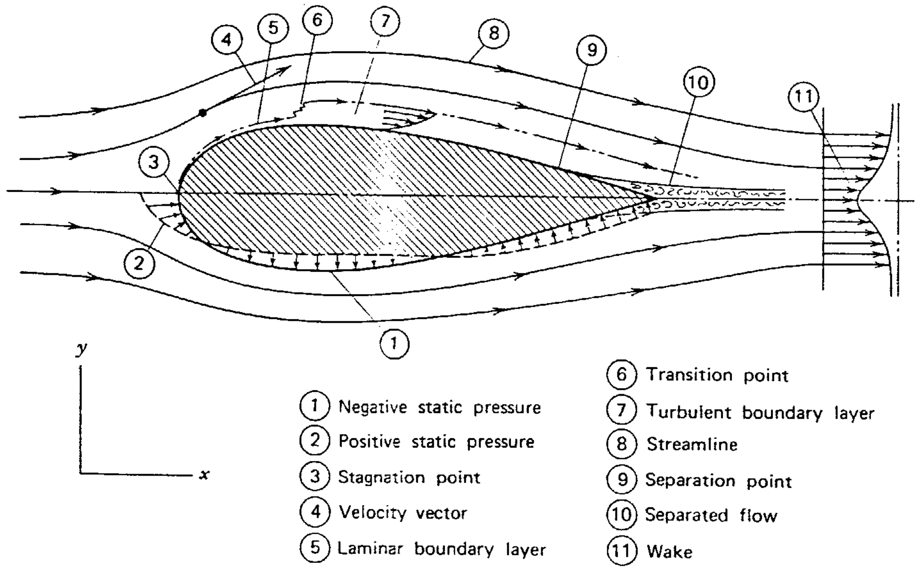
+ Gas responds to solid boundaries by 'sticking' and transporting momentum to the free field. There is a tangential force component (like friction) at the surface.



+ Boundary discontinuities imply large pressure gradients. Gas separates to an extent governed by pressure forces.







Two-dimensional flow around a 'streamlined' shape.

- Separation and wake are <u>always present</u>
- Parts of aircraft upstream produce momentum deficits affecting downstream parts
- The wake spreads due to turbulent mixing.
- The boundary layer responds to pressure, curvature, and history at the boundary. This complex flow can be normalized by Reynold's Number

$$R_e = \frac{\ell V}{\nu}$$

$\ell \equiv$ reference length
$V \equiv$ reference velocity
$\nu \equiv$ kinematic viscosity $= \frac{\mu}{\rho}$
$\mu \equiv$ viscosity

Gas is Compressible

Relationship of Pressure and Velocity

- Incompressible

$$p + \frac{1}{2}\rho V^2 = \text{constant}$$

- Compressible

$$\frac{V^2}{2} + \frac{\gamma}{\gamma - 1}\frac{p}{\rho} = \text{constant}$$

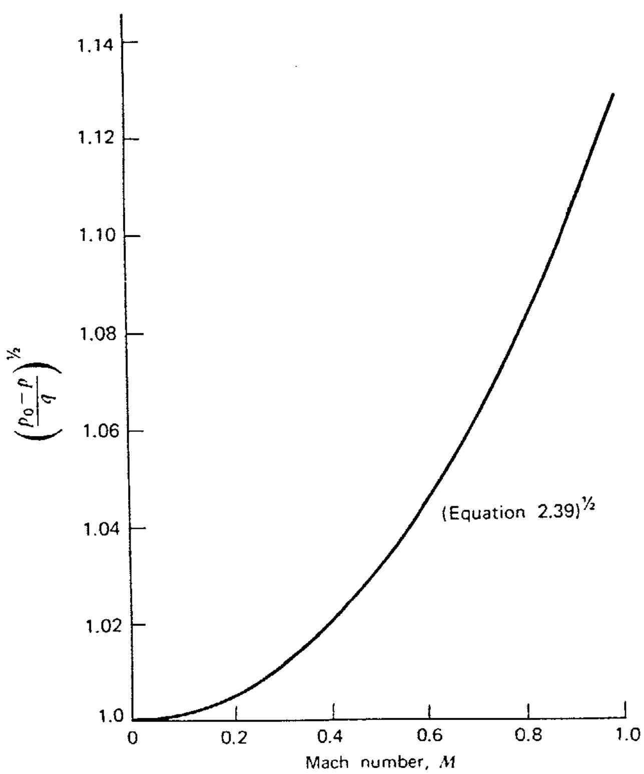
or $$\frac{V^2}{2} + \frac{a^2}{\gamma - 1} = \text{constant}$$

$\gamma \equiv$ ratio of specific heat of gas at constant pressure to specific heat at constant volume

$$\frac{p}{\rho^\gamma} = \text{constant}$$

$a \equiv$ acoustic velocity

$$a = \left(\frac{\gamma p}{\rho}\right)^{1/2} = (\gamma R T)^{1/2}$$



Relationship between reservoir pressure and dynamic pressure as a function of Mach number.

Shock Waves

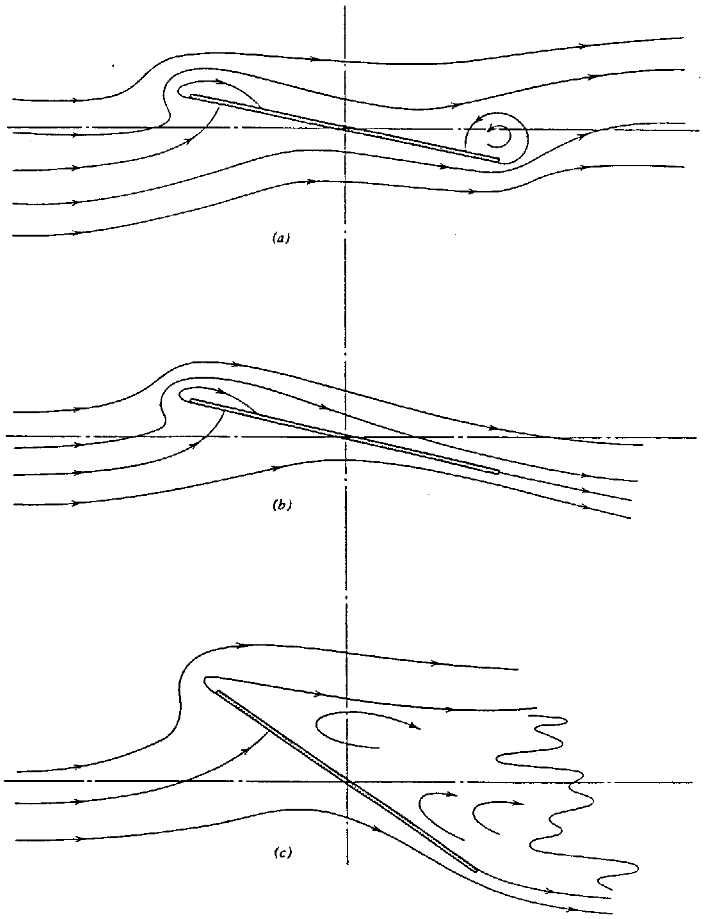
- Response to change in boundary conditions <u>in supersonic flow</u>
- Drastic change in gas state, especially increase in pressure
- Normal shock
  + Upstream flow is <u>always supersonic</u>
  + Downstream flow is <u>always subsonic</u>
- Oblique shock
  + Downstream flow is <u>usually supersonic</u>
- Shock wave location is extremely sensitive to boundary and interaction with boundary within the "sonic cone"

Compressible Characteristics are normalized by Mach Number

$$M = \frac{V}{a}$$

AIRFOILS AND WINGS

Operation of the Airfoil - 2-D



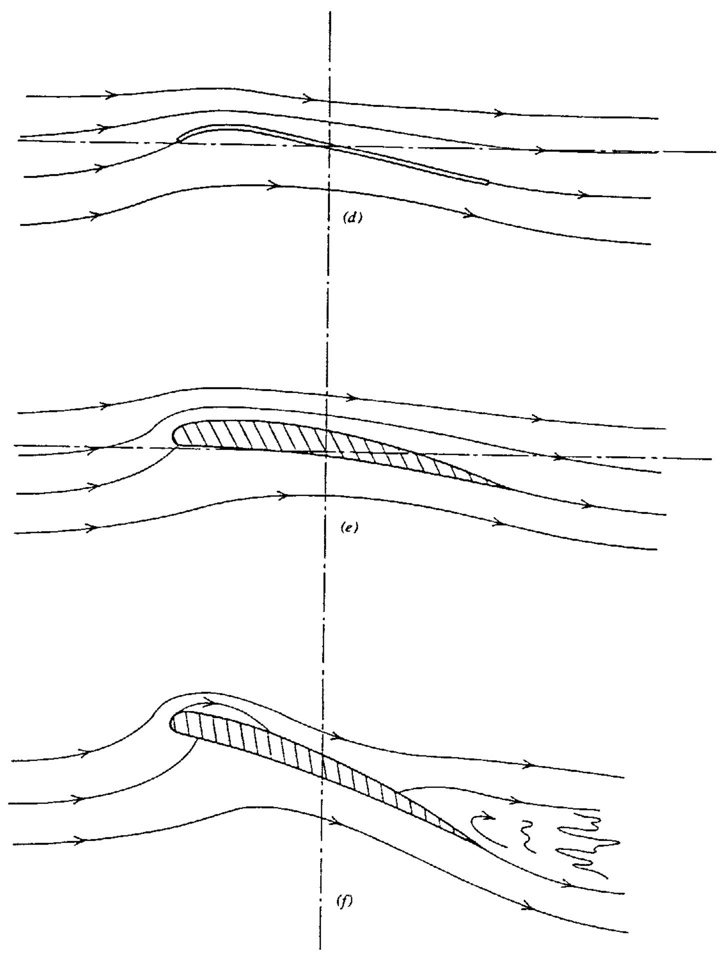


Figure 3.4 Progressive development of airfoil shapes. (*a*) Flat plate at sudden angle of attack—no lift. (*b*) Flat plate at angle of attack in steady flow and generating lift. (*c*) Flat plate experiencing leading edge separation and loss of lift (stall). (*d*) Flat plate with curved leading edge to prevent leading edge separation. (*e*) Airfoil with thickness and camber to delay stall. (*f*) Airfoil with trailing edge separation.

Airfoil (and Aircraft) forces and moments are normalized by dynamic pressure

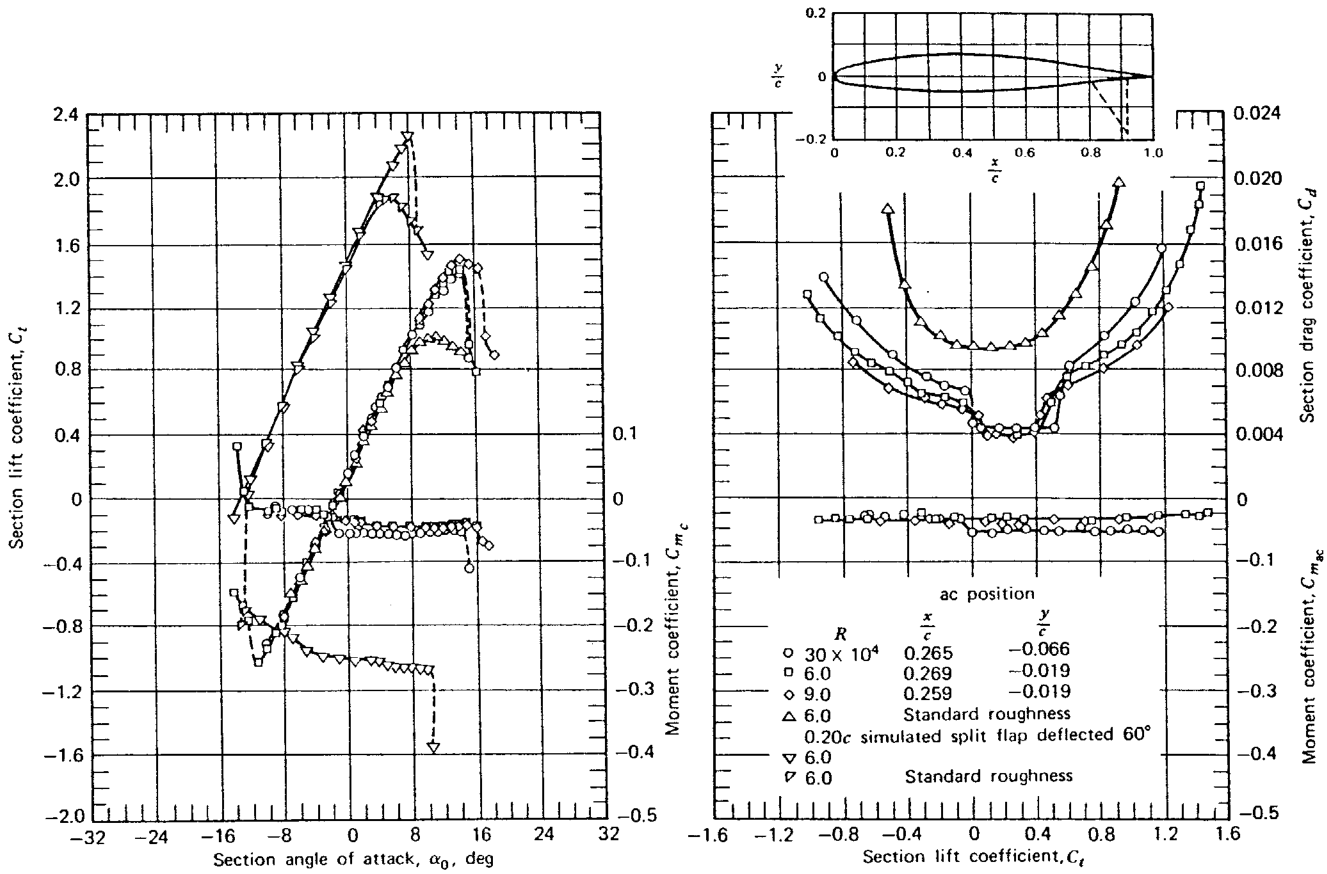
$$q = \frac{1}{2}\rho V^2 \quad C_L = \frac{Lift}{S_{REF}\, q} \quad C_D = \frac{Drag}{S_{REF}\, q} \quad C_M = \frac{Pitch\ Moment}{S_{REF}\, C_{REF}\, q}$$

S_{REF} - Reference area

C_{REF} - Reference length

- Note that C_M is associated with a moment center location

Actual data:



Aerodynamic characteristics of the NACA 65, −212, a = 0.6 airfoil.

At "ac" $\frac{dC_m}{dC_L} = 0$ (C_m may be $\neq 0$)

- Location fairly fixed with α for attached flow
- Location usually close to $x/c = 0.25$
- $\frac{dC_m}{dC_L} = 0$ location not necessarily $\frac{dC_m}{d\alpha} = 0$ location

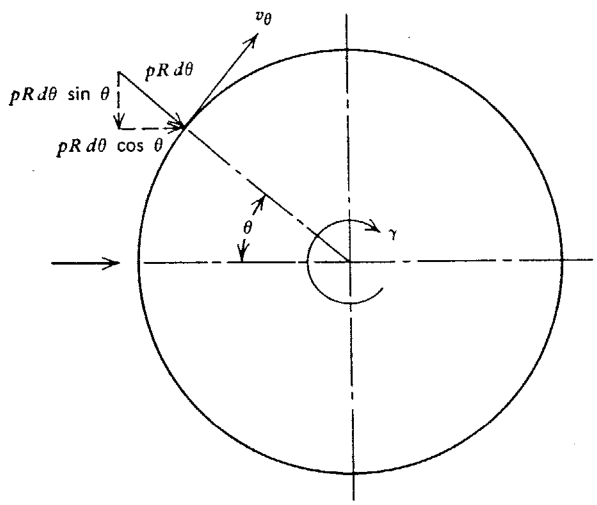
Potential Flow Lift Model for Airfoil

Lift can be modeled in 2-D by adding circulation (vortex function) to a circular cylinder streamline boundary

$$v_\theta = 2U \sin\theta + \frac{\gamma}{2\pi R}$$

$$p - p_0 = \frac{1}{2}\rho U^2 - \frac{1}{2}\rho \left|2U \sin\theta + \frac{\gamma}{2\pi R}\right|^2$$

$$L = -\int_0^\pi pR \sin\theta \, d\theta$$

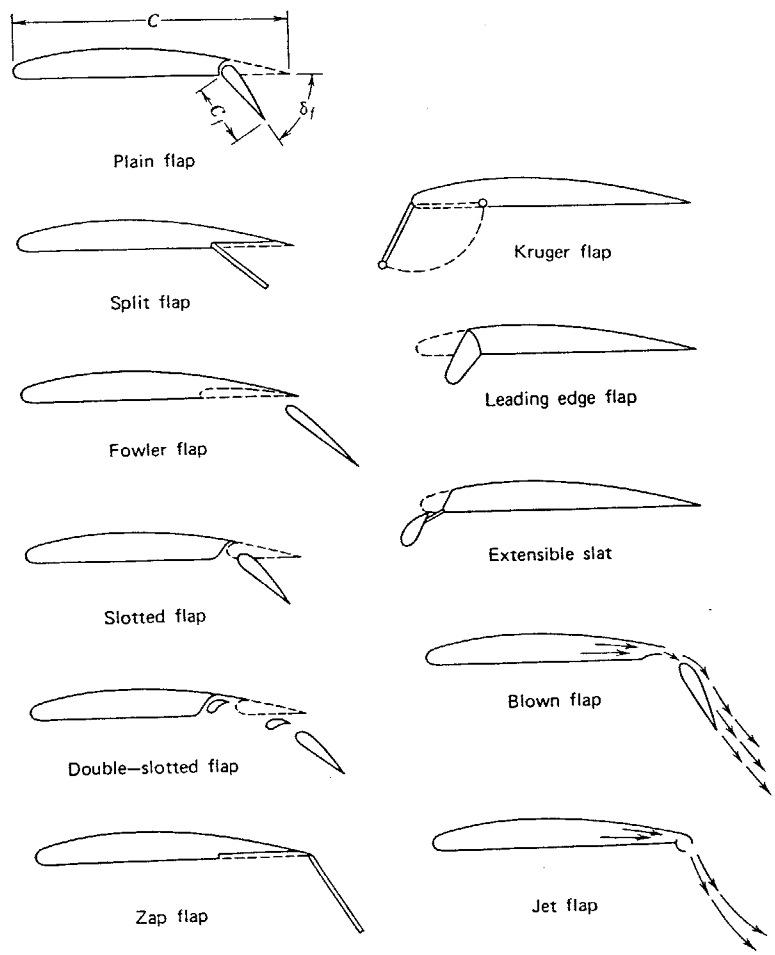


Circular cylinder with circulation.

Circulation strength is directly related to C_L. Cylinder lift is correlated to an airfoil through a geometric transformation that relates the airfoil TE to the separation streamline location on the cylinder.

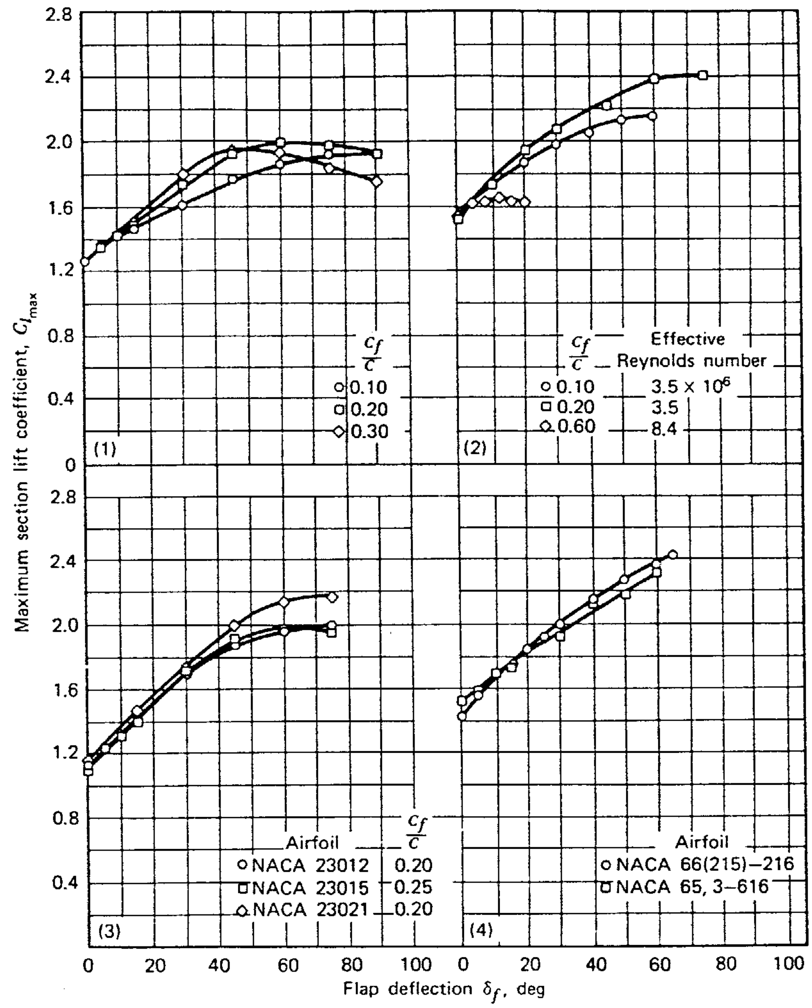
Lift Variation Through Flaps

- See previous diagram of NACA airfoil data
  - Almost constant increment of C_L with flap over α
  - C_m increment nearly constant, but not well behaved
  - Flap drastically affects drag



Flap configurations.

Plain flaps constitute the primary form of control surface; it is imperitive to thoroughly understand plain flap characteristics!



(1) Clark–Y airfoil; $R = 0.609 \times 10^6$.

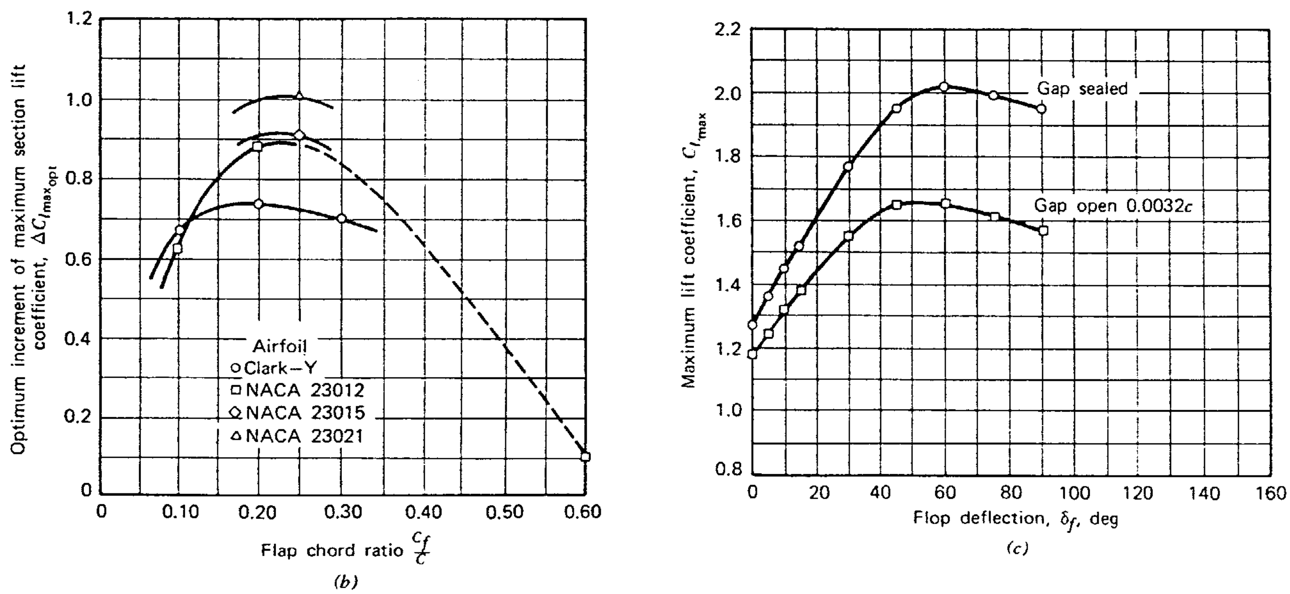
(2) NACA 230–series airfoils; $R = 0.609 \times 10^6$.

(3) NACA 23012 airfoil.

(4) NACA 6–series airfoils; $R = 6.0 \times 10^6, \frac{c_f}{c} = 0.20$.

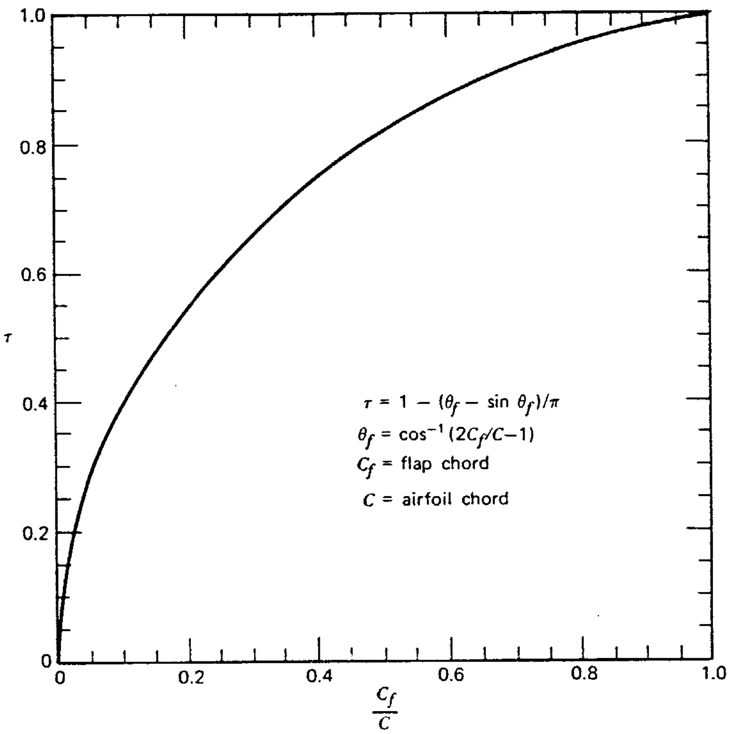
(a)

Figure 3.25 Performance of plain flaps. (*a*) Variation of maximum section lift coefficient with flap deflection for several airfoil sections equipped with plain flaps. (*b*) Variation of optimum increment of maximum section lift coefficient with flap chord ratio for several airfoil sections equipped with plain flaps. (*c*) Effect of gap seal on maximum lift coefficient of a rectangular Clark-Y wing equipped with a full-span 0.20*c* plain flap. $A = 6$, $R = 0.6 \times 10^4$.

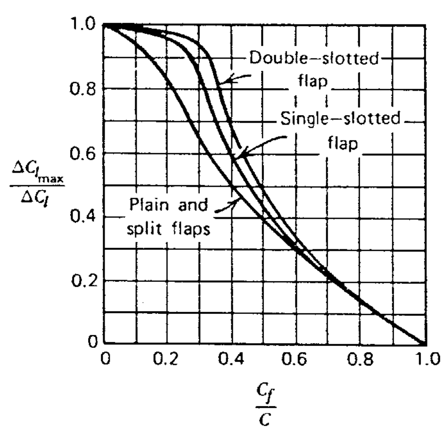


Empirical 2-D flap performance

$$c_\ell = c_{\ell_\alpha}(\alpha + \tau \eta \delta_f)$$



Flap effectiveness factor.



$C_{l_{max}}$ increment ratio as a function of flap chord ratio.

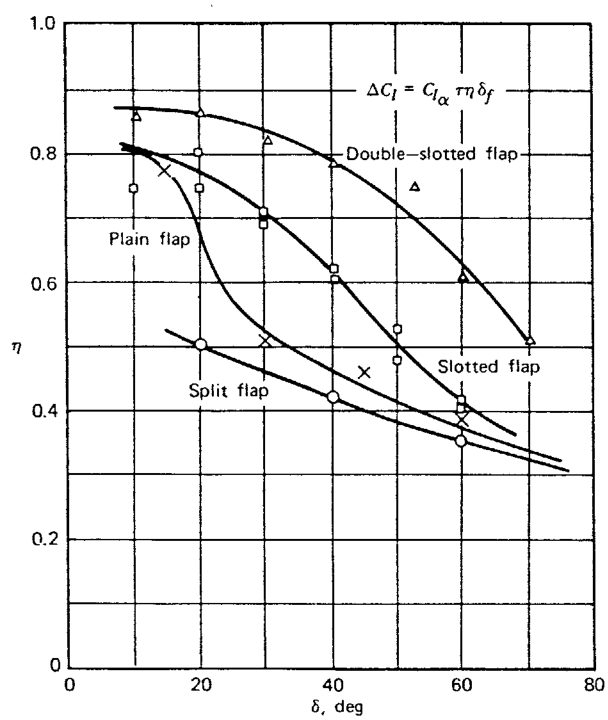


Figure 3.33 Correction factor to flap effectiveness factor τ. Note that curves apply for thickness ratios of approximately 12% and flap chord fractions of 40% or less.

The Finite Wing - 3-D

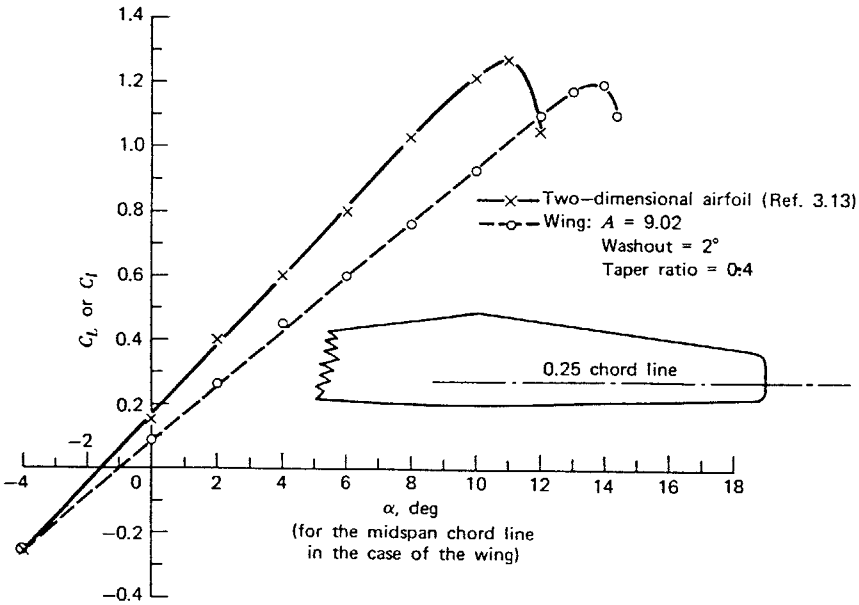
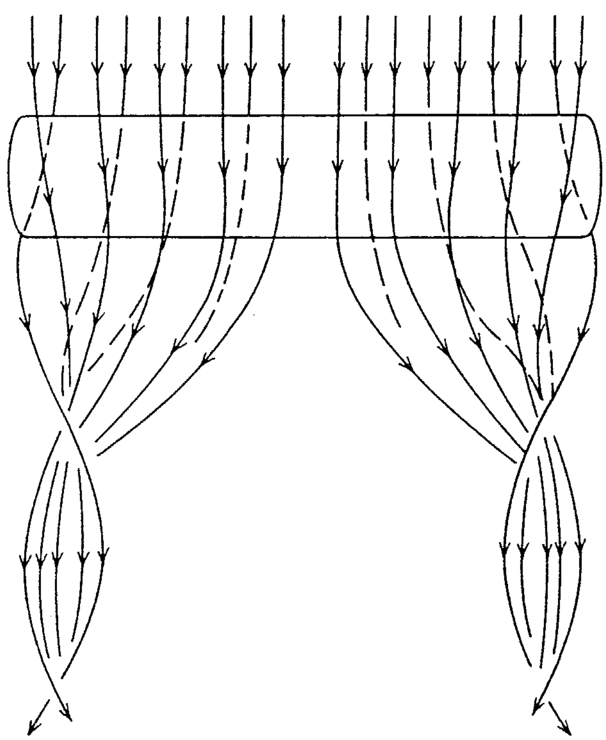
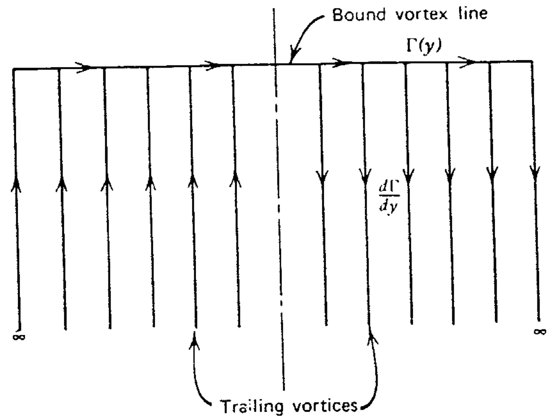


Figure 3.50 Comparison of NACA 65–210 airfoil lift curve with that of a wing using the same airfoil.



Generation of vortex system by finite aspect ratio wing.



Lifting line model of a wing and trailing vortex system.

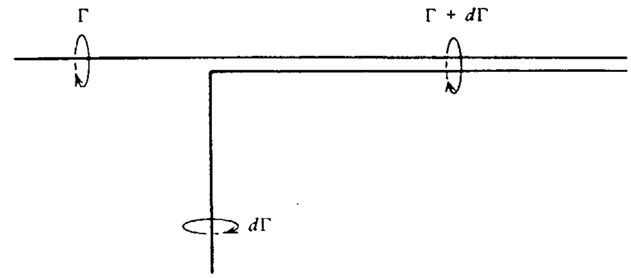
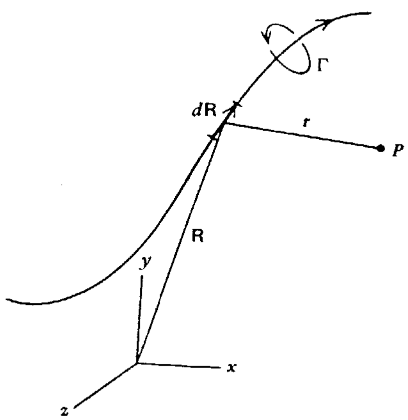
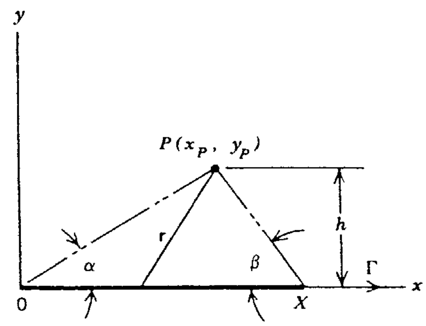


Illustration of vortex continuity.

$$d\mathbf{v}_i = \frac{\Gamma}{4\pi}\frac{\mathbf{r}\times\mathbf{dR}}{|\mathbf{r}|^3}$$



Definition of quantities used in the Biot–Savart law.



The Biot–Savart law for a straight-line vortex.

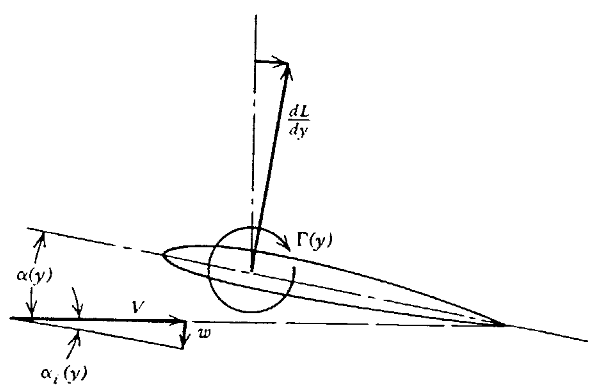
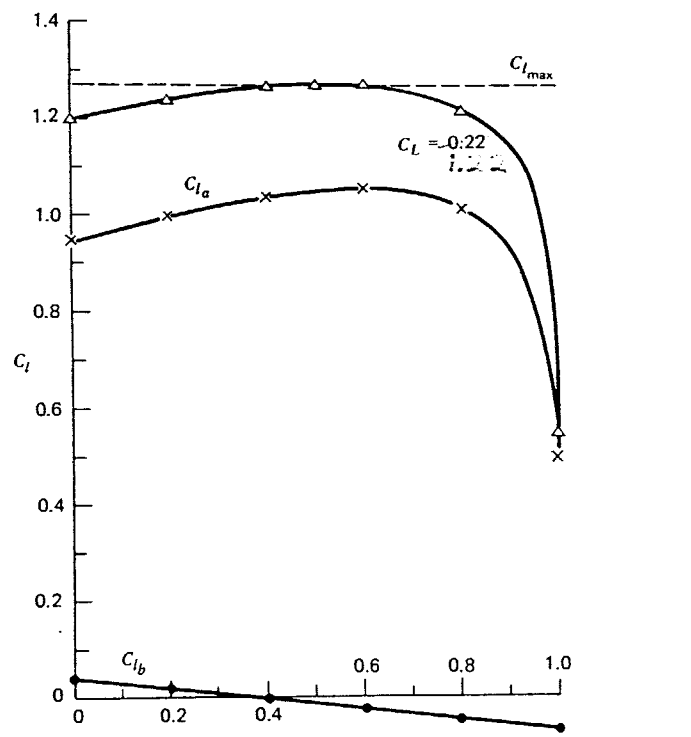


Figure 3.54 A wing section under the influence of the free-stream velocity and the downwash.

- Downwash is a very important influence on aircraft stability characteristics
- Downstream effects are very strong, dominated by trailing vortices
- Upstream effects are weak, dominated by bound vortice (non-negligible for close-coupled wings)
- See treatises of Munk and others for thorough treatment of induced flow effects



Predicted lift coefficient distributions for wing of Figure 3.50.

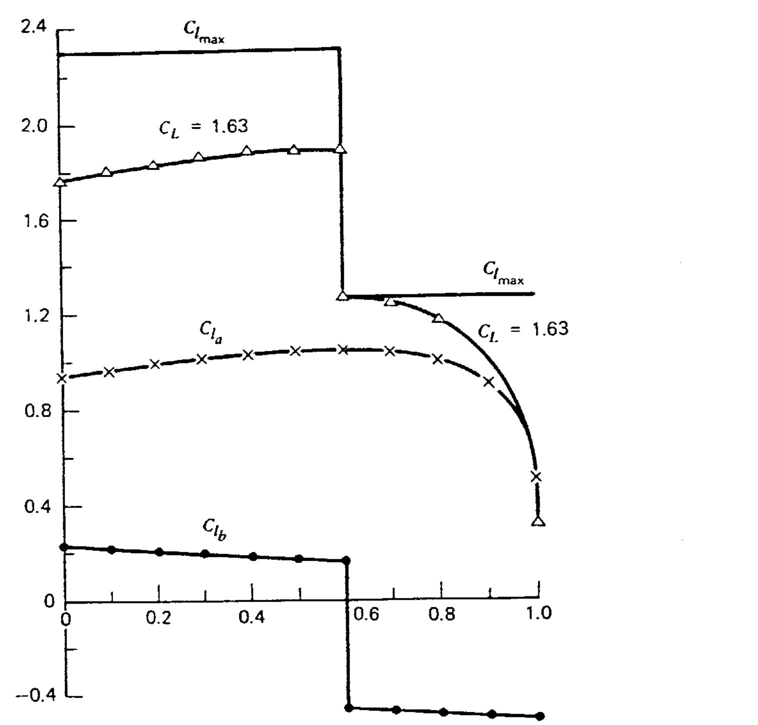
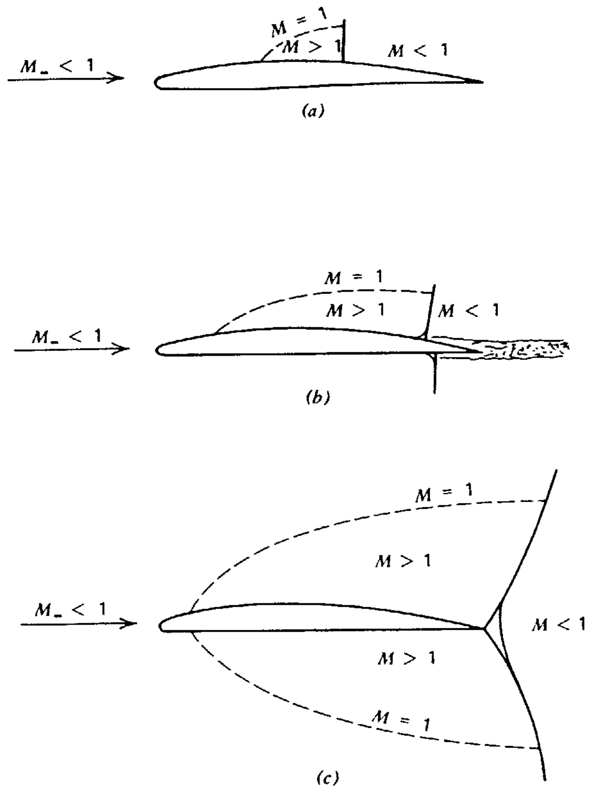
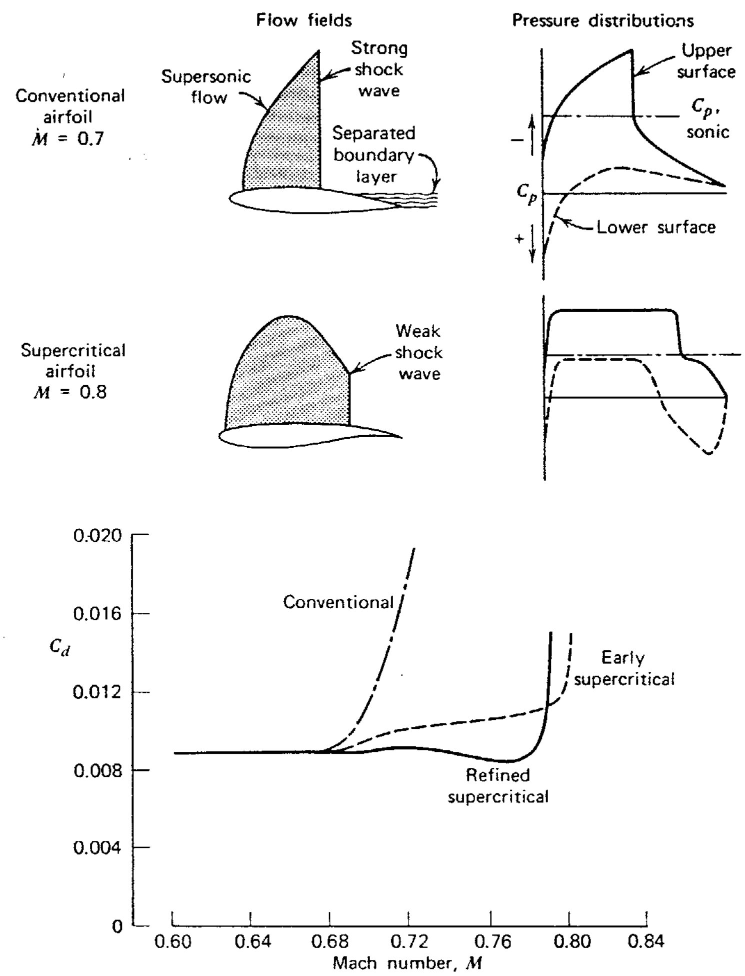


Figure 3.58 Predicted lift coefficient distributions for wing of Figure 3.50 with 60% span, 20% chord split flaps deflected 60°.

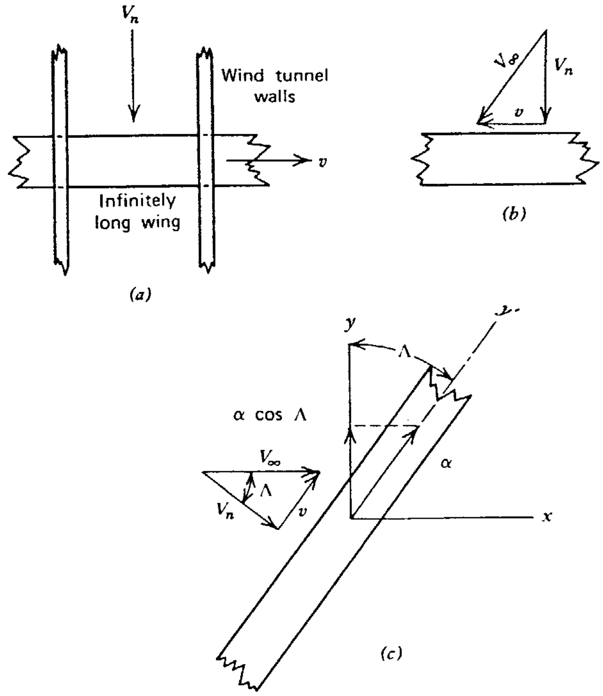
Wings and Compressibility



Airfoils in transonic flow.



Supercritical flow phenomena.



Effect of sweepback.

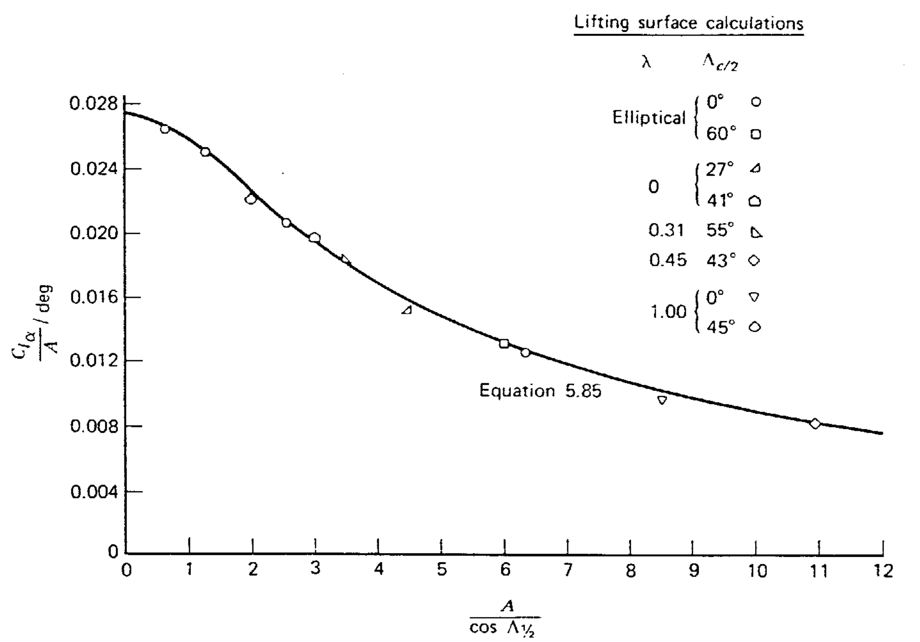
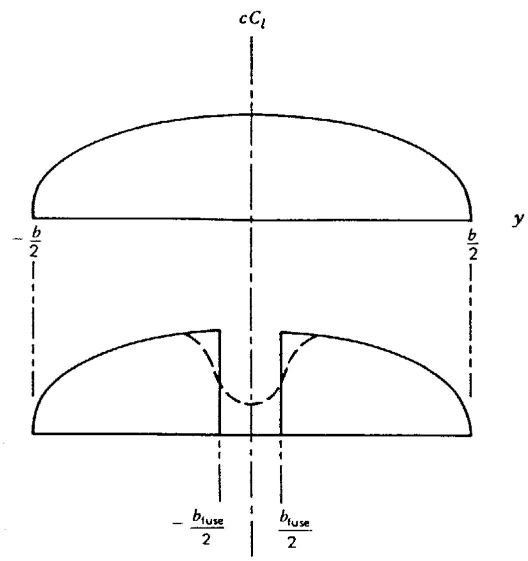


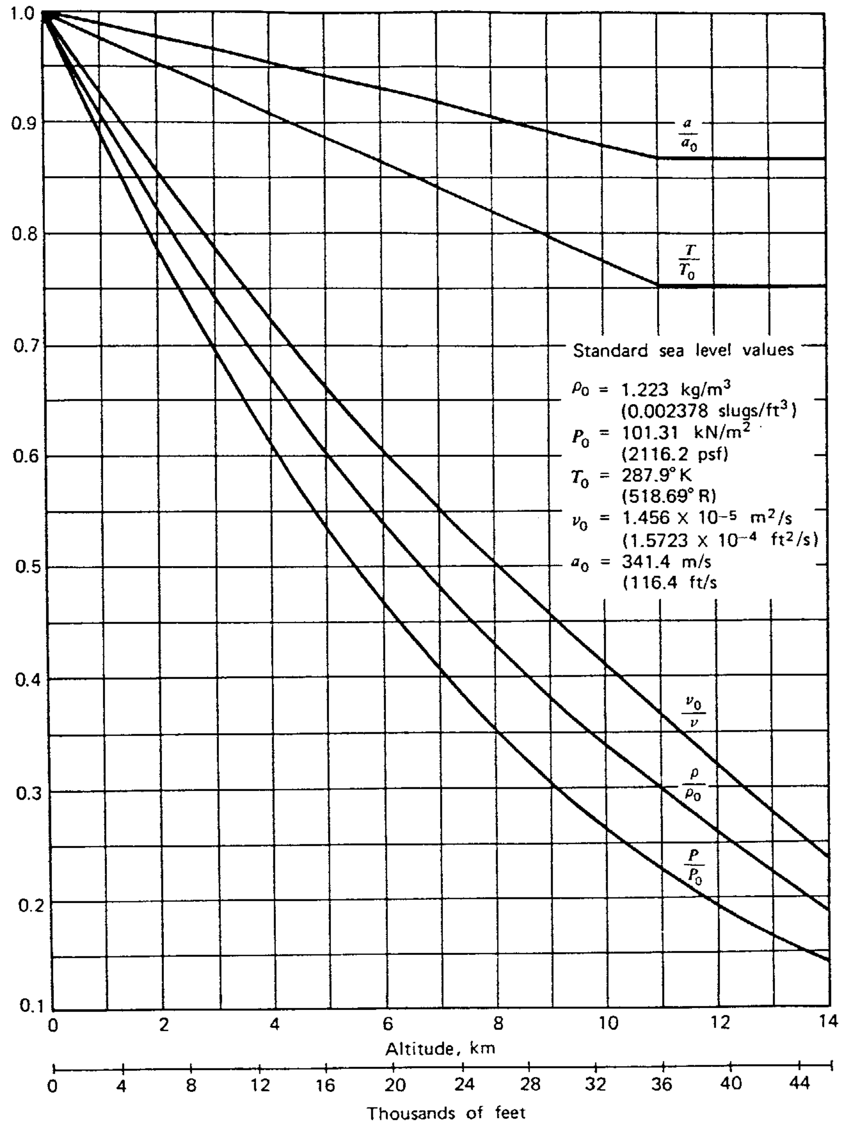
Figure 5.28 C_{L_α}/A with $A/\cos \Lambda_{1/2}$ as determined by several methods. $a_0 = 2\pi$, $M = 0$.



Effect of fuselage on spanwise lift distribution.

THE ATMOSPHERE

AIRSPEED DEFINITIONS



The standard atmosphere.

Dynamic Pressure

$$q = 1/2\ \rho V_T^2 = 1481\ \delta M^2 = \frac{V_e^2}{295} \quad \text{(PSF)}$$

(Ft/Sec) (Kts)

Impact Pressure (Pitot-Static Measure) $\quad q_c = (P_T - P_s)$

$$q_c = q F_c \qquad F_c = \left(1 + \frac{M^2}{4} + \frac{M^4}{40} + \frac{M^8}{80} \cdots\right)$$

$$q_c = \left[\left(1 + 0.2M^2\right)^{3.5} - 1\right] P_s \quad \text{(PSF)}$$

Mach No.

$$M = \sqrt{5\left[\left(\frac{P_0}{P}\left\{\left[1 + 0.2\left(\frac{V_c(\text{Kts})}{661.5}\right)^2\right]^{3.5} - 1\right\} + 1\right)^{0.286} - 1\right]}$$

Indicated Airspeed

$$V_I = V_c + \Delta V_{\text{Pitot-Static Source Error}}$$

Calibrated Airspeed

$$V_c = 1479.1\sqrt{\left[1 + \frac{P_T - P_s}{P_0}\right]^{0.28571} - 1} \quad \text{(Knots)}$$

Equivalent Airspeed

$$V_e = 32.174\sqrt{P_s\left\{\left[1 + \frac{P_T - P_s}{P_s}\right]^{0.28571} - 1\right\}} \quad \text{(Knots)}$$

True Airspeed

$$V_T = \frac{V_e}{\sqrt{\sigma}} \qquad M = \frac{V_T}{a}$$

Total Conditions

$M < 1.0 \qquad \gamma = 1.4$

$$T_T = T\left(1 + \frac{\gamma - 1}{2}\epsilon M^2\right) = T\left(1 + .2\ \epsilon M^2\right)$$

$$\rho_T = \rho\left(1 + \frac{\gamma - 1}{2}M^2\right)^{1/\gamma - 1} = \rho\left(1 + .2M^2\right)^{2.5}$$

$$P_T = P\left(1 + \frac{\gamma - 1}{2}M^2\right)^{\frac{\gamma}{\gamma - 1}} = P\left(1 + .2M^2\right)^{3.5}$$

$M = 1 \qquad \gamma = 1.4$

$T/T_T = .833$

$\rho/\rho_T = .634$

$P/P_T = .528$

Aircraft Stability and Control Short Course

Savannah Section of AIAA

January 26, 1993

Walter Lounsbery
Course Coordinator
966-4486

SESSION II - Aircraft Equations of Motion and Etc.

+ Please Register Attendence

+ Aircraft Equations of Motion

  - Assumptions
  - Axis Systems
  - Formulation
  - Perturbed Equations of Motion
  - Separation of Longitudinal and Lateral/Directional EOM

+ Kinetics of Simple Maneuvers

  - Level Flight
  - Symmetric Pull-Up
  - Turn
  - Roll

+ Break: Video of NASA kinematic experiments

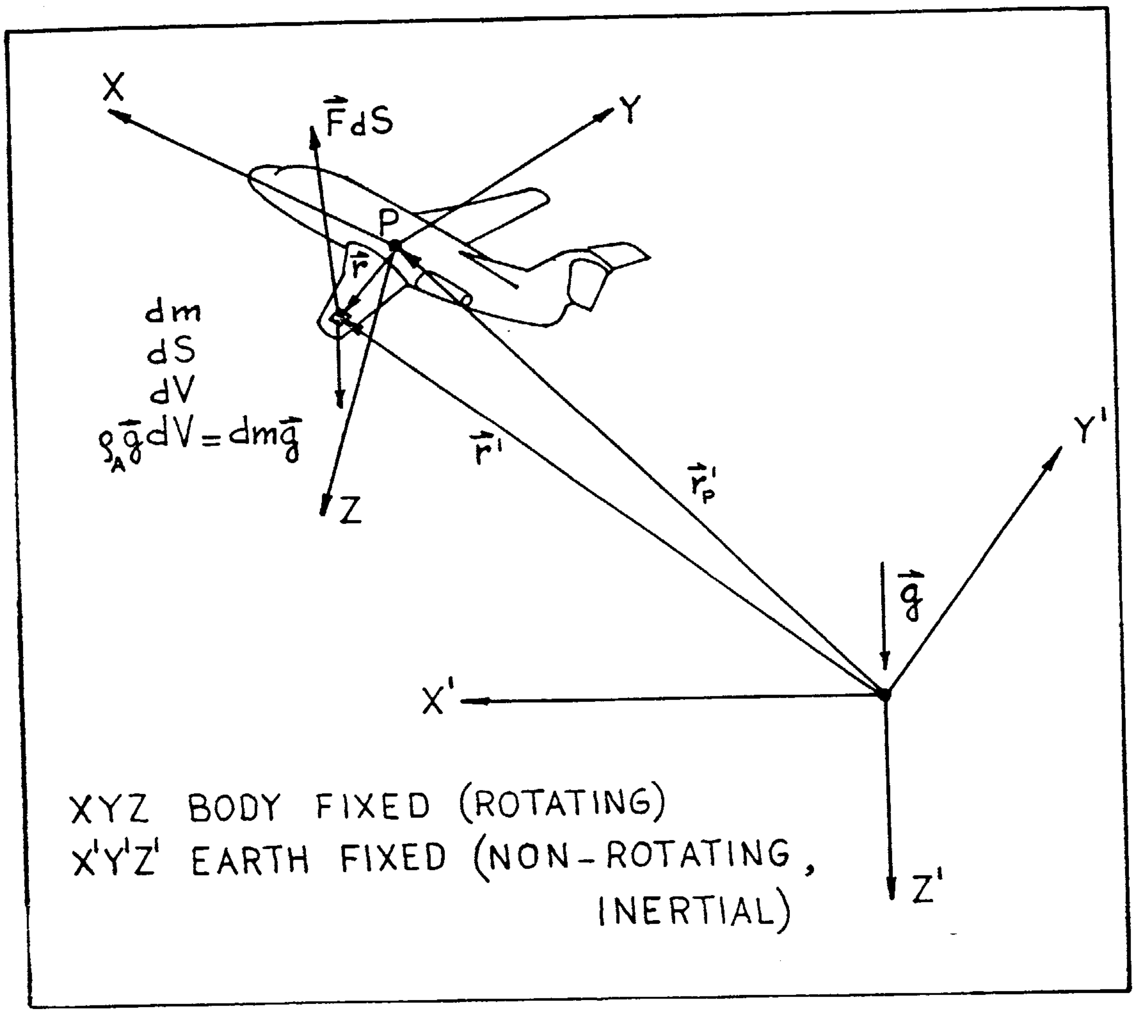
+ More Wing and Control Aerodynamics

  - Airfoil Section Properties
  - Wing Aerodynamic Center
  - Drag
  - Plain Flap Hinge Moments
  - Lift, Drag, Sideforce, and Beta

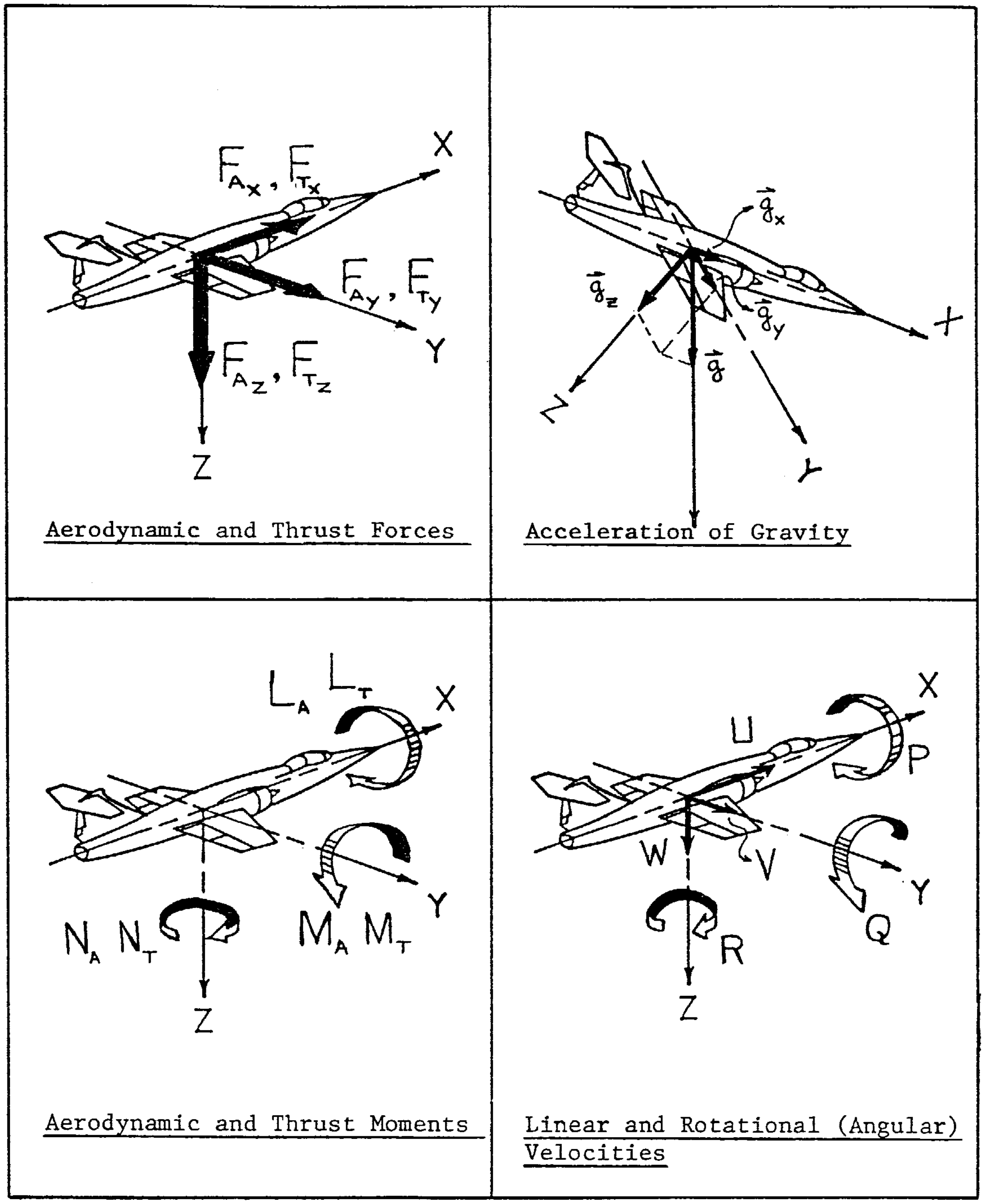
+ Longitudinal Derivatives

Assumptions

- The Earth is flat
- The aircraft is a rigid body
- Aircraft mass and inertia is essentially constant
- Aircraft have mirror symmetry about the x-z plane
- Body axis reference at center of mass



Coordinate Systems



Defintions of vector components in the equations of motion

Note: Positive sense is in the direction of the arrows

$$\frac{d}{dt}\int_V \rho_A \frac{d\vec{r}\,'}{dt}\,dV = \int_V \rho_A \vec{g}\,dV + \int_S \vec{F}\,dS$$

Linear Momentum — Applied Forces

$$\frac{d}{dt}\int_V \vec{r}\,' \times \rho_A \frac{d\vec{r}\,'}{dt}\,dV = \int_V \vec{r}\,' \times \rho_A \vec{g}\,dV + \int_S \vec{r}\,' \times \vec{F}\,dS$$

Angular Momentum — Applied Moments

provides:

$$m(\dot{U} - VR + WQ) = mg_x + F_{A_x} + F_{T_x}$$

$$m(\dot{V} + UR - WP) = mg_y + F_{A_y} + F_{T_y}$$

$$m(\dot{W} - UQ + VP) = mg_z + F_{A_z} + F_{T_z}$$

$$I_{xx}\dot{P} - I_{xz}\dot{R} - I_{xz}PQ + (I_{zz} - I_{yy})RQ = L_A + L_T$$

$$I_{yy}\dot{Q} + (I_{xx} - I_{zz})PR + I_{xz}(P^2 - R^2) = M_A + M_T$$

$$I_{zz}\dot{R} - I_{xz}\dot{P} + (I_{yy} - I_{xx})PQ + I_{xz}QR = N_A + N_T$$

$$g_x = -g\sin\theta$$

$$g_y = g\sin\phi\cos\theta$$

$$g_z = g\cos\phi\cos\theta$$

$$\dot{\phi} = P + Q\sin\phi\tan\theta + R\cos\phi\tan\theta = P + \dot{\psi}\sin\theta$$

$$\dot{\theta} = Q\cos\phi - R\sin\phi$$

$$\dot{\psi} = (Q\sin\phi + R\cos\phi)\sec\theta$$

or more directly

$$\dot{U} = VR - WQ - g\sin\theta + F_x/m$$

$$\dot{V} = WP - UR + g\sin\phi\cos\theta + F_y/m$$

$$\dot{W} = UQ - VP + g\cos\phi\cos\theta + F_z/m$$

Let $I_1 = \dfrac{I_{yy} - I_{xx}}{I_{zz}} - 1$ $\qquad I_2 = \dfrac{I_{zz} - I_{yy}}{I_{xz}} + \dfrac{I_{xz}}{I_{zz}}$

$I_3 = \dfrac{I_{xz}}{I_{zz}} - \dfrac{I_{xx}}{I_{xz}}$ $\qquad I_4 = I_{zz} - I_{xx}$

$I_5 = \dfrac{I_{yy} - I_{xx}}{I_{xz}} - \dfrac{I_{xz}}{I_{xx}}$ $\qquad I_6 = 1 + \dfrac{I_{zz} - I_{yy}}{I_{xx}}$

$I_7 = \dfrac{I_{xz}}{I_{xx}} - \dfrac{I_{zz}}{I_{xz}}$

Then

$$\dot{P} = \frac{1}{I_3}\left(I_1 PQ + I_2 QR - \frac{L}{I_{xz}} - \frac{N}{I_{zz}}\right)$$

$$\dot{Q} = \frac{1}{I_{yy}}\left(I_4 PR + I_{xz}(R^2 - P^2) + M\right)$$

$$\dot{R} = \frac{1}{I_7}\left(I_5 PQ + I_6 QR - \frac{L}{I_{xx}} - \frac{N}{I_{xz}}\right)$$

$$\dot{x}' = U\cos\theta\cos\psi + V(\sin\phi\sin\theta\cos\psi - \cos\phi\sin\psi) + W(\sin\phi\sin\psi + \cos\phi\sin\theta\cos\psi)$$

$$\dot{y}' = U\cos\theta\sin\psi + V(\cos\phi\cos\psi + \sin\phi\sin\theta\sin\psi) + W(\cos\phi\sin\theta\sin\psi - \sin\phi\cos\psi)$$

$$\dot{z}' = U(-\sin\theta) + V(\sin\phi\cos\theta) + W(\cos\theta\cos\phi)$$

Perturbed Equations of Motion

Objective is to take six simultaneous nonlinear differential equations (first order in U, V, W and second order in ψ, θ, ϕ), and produce linearized relations suitable for stability studies.

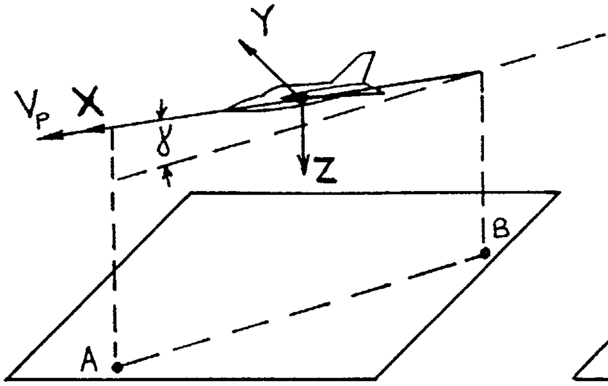
* Steady state flight is defined as a flight condition for which all motion variables remain constant with time relative to a body fixed coordinate system.

* Perturbed state flight is defined as a flight condition for which all motion variables are defined relative to a steady state flight condition.

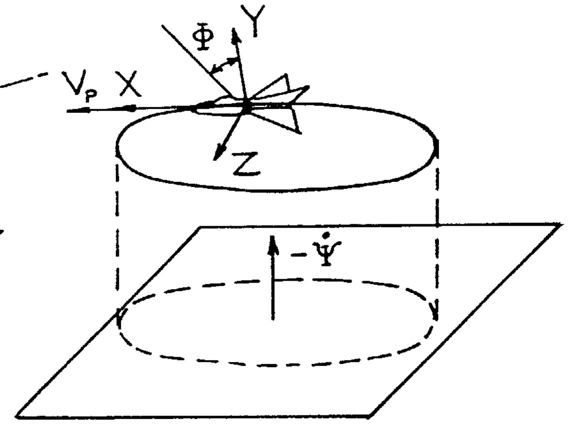
Let there be stability axes:

$$U = U_o + u \qquad W = w \qquad \alpha = \tan^{-1}\frac{W}{U_o + u} \doteq \frac{W}{U_o}$$

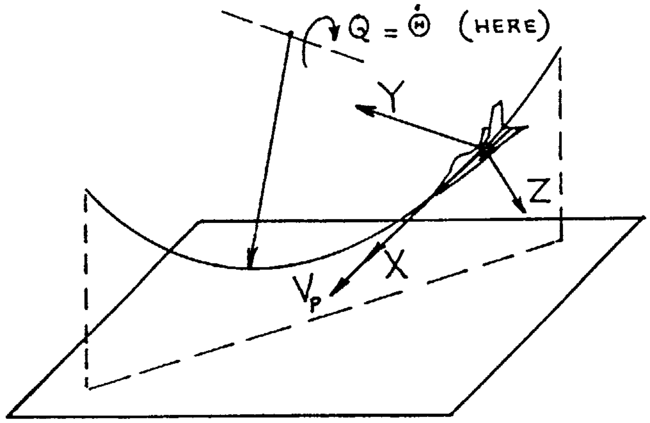
$$\frac{u}{U_o} \ll 1 \quad \frac{W}{U_o} \ll 1$$



1. Rectilinear Flight

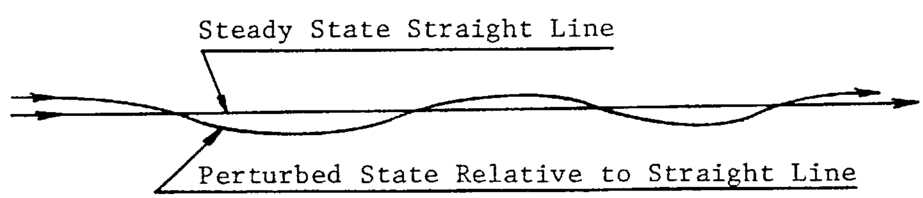


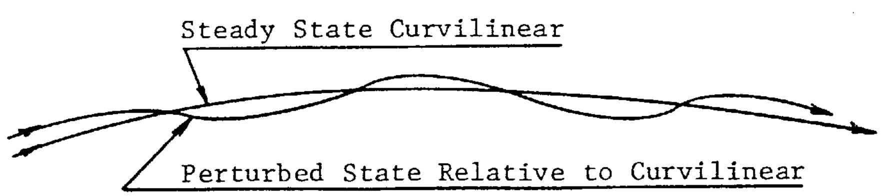
2. Steady Level Turn



3. Steady Symmetrical Pull-up

Examples of Steady State Flight Paths





Examples of Perturbed State Flight Paths subscript '1' denotes steady-state

$$m(\dot{u} - V_1 r - R_1 v + W_1 q + Q_1 w) = -mg\theta\cos\Theta_1 + f_{A_x} + f_{T_x}$$

$$m(\dot{v} + U_1 r + R_1 u - W_1 p - P_1 w) = -mg\theta\sin\Phi_1\sin\Theta_1 + mg\phi\cos\Phi_1\cos\Theta_1 + f_{A_y} + f_{T_y}$$

$$m(\dot{w} - U_1 q - Q_1 u + V_1 p + P_1 v) = -mg\theta\cos\Phi_1\sin\Theta_1 - mg\phi\sin\Phi_1\cos\Theta_1 + f_{A_z} + f_{T_z}$$

$$I_{xx}\dot{p} - I_{xz}\dot{r} - I_{xz}(P_1 q + Q_1 p) + (I_{zz} - I_{yy})(R_1 q + Q_1 r) = l_A + l_T$$

$$I_{yy}\dot{q} + (I_{xx} - I_{zz})(P_1 r + R_1 p) + I_{xz}(2P_1 p - 2R_1 r) = m_A + m_T$$

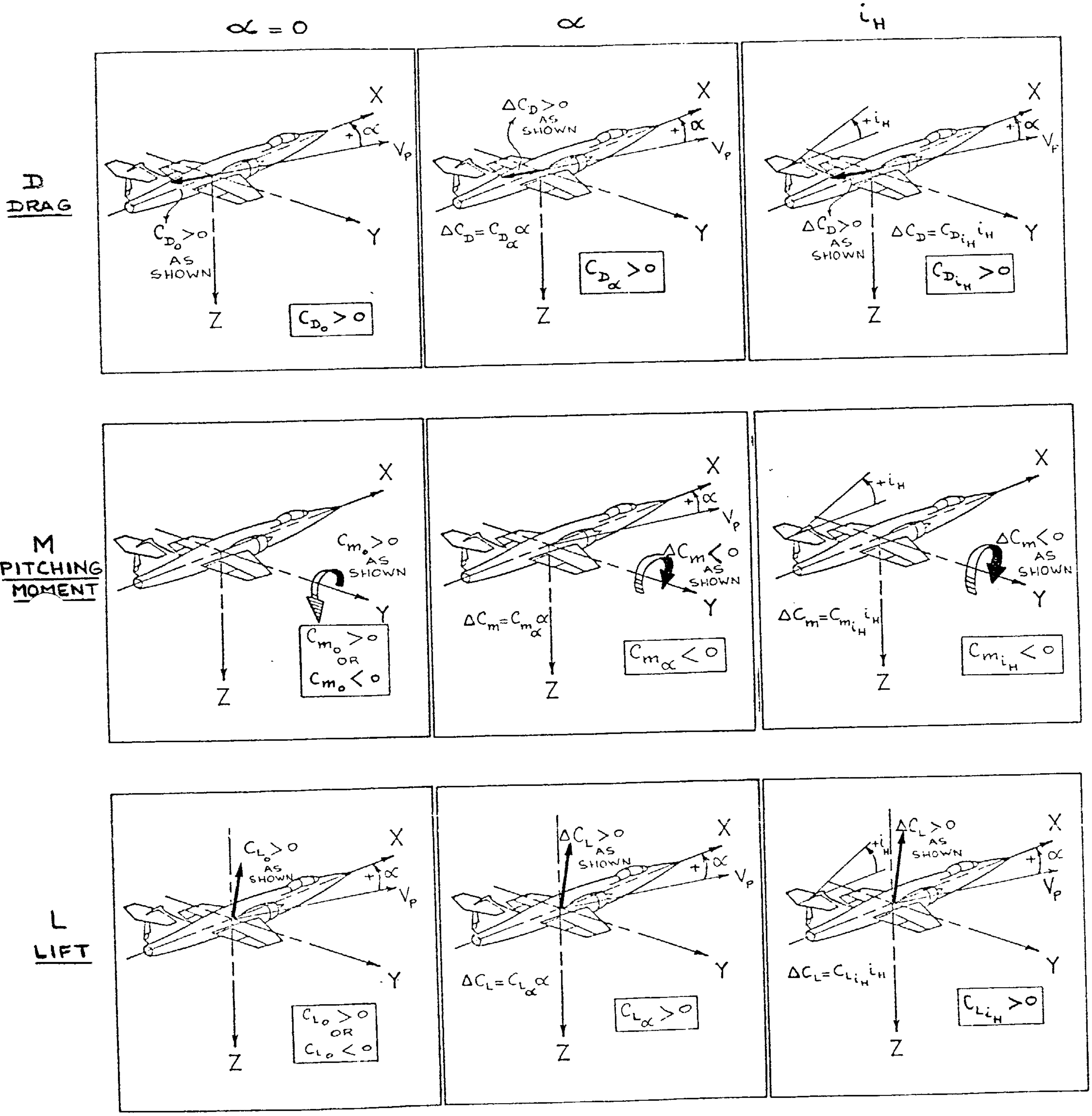
$$I_{zz}\dot{r} - I_{xz}\dot{p} + (I_{yy} - I_{xx})(P_1 q + Q_1 p) + I_{xz}(Q_1 r + R_1 q) = n_A + n_T$$

$$p = \dot{\phi} - \dot{\Psi}_1\theta\cos\Theta_1 - \dot{\psi}\sin\Theta_1$$

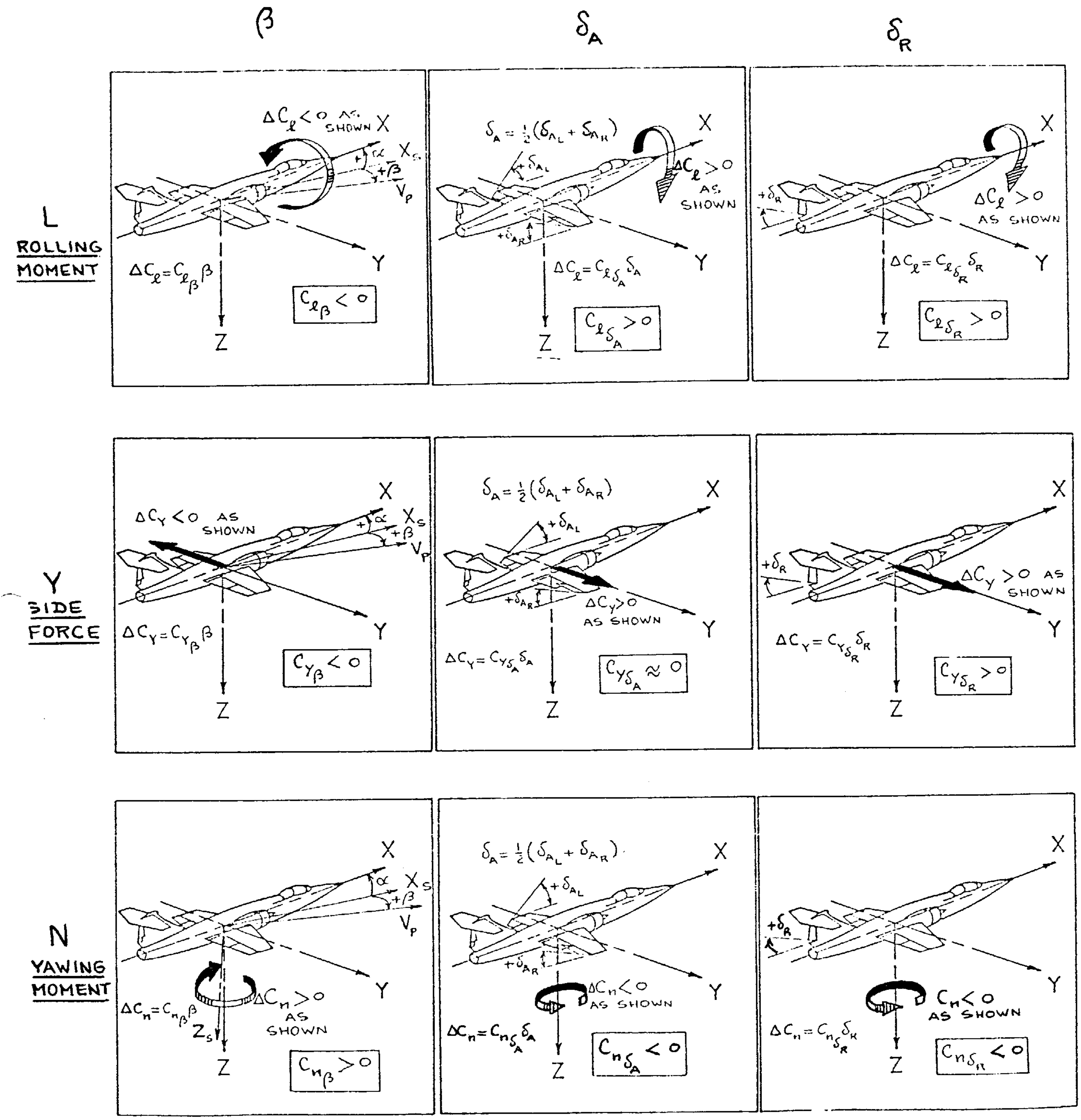
$$q = -\dot{\Theta}_1\phi\sin\Phi_1 + \dot{\theta}\cos\Phi_1 + \dot{\Psi}_1\phi\cos\Theta_1\cos\Phi_1 - \dot{\Psi}_1\theta\sin\Theta_1\sin\Phi_1 + \dot{\psi}\cos\Theta_1\sin\Phi_1$$

$$r = -\dot{\Psi}_1\phi\cos\Theta_1\sin\Phi_1 - \dot{\Psi}_1\theta\sin\Theta_1\cos\Phi_1 + \dot{\psi}\cos\Theta_1\cos\Phi_1 - \dot{\Theta}_1\phi\cos\Phi_1 - \dot{\theta}\sin\Phi_1$$

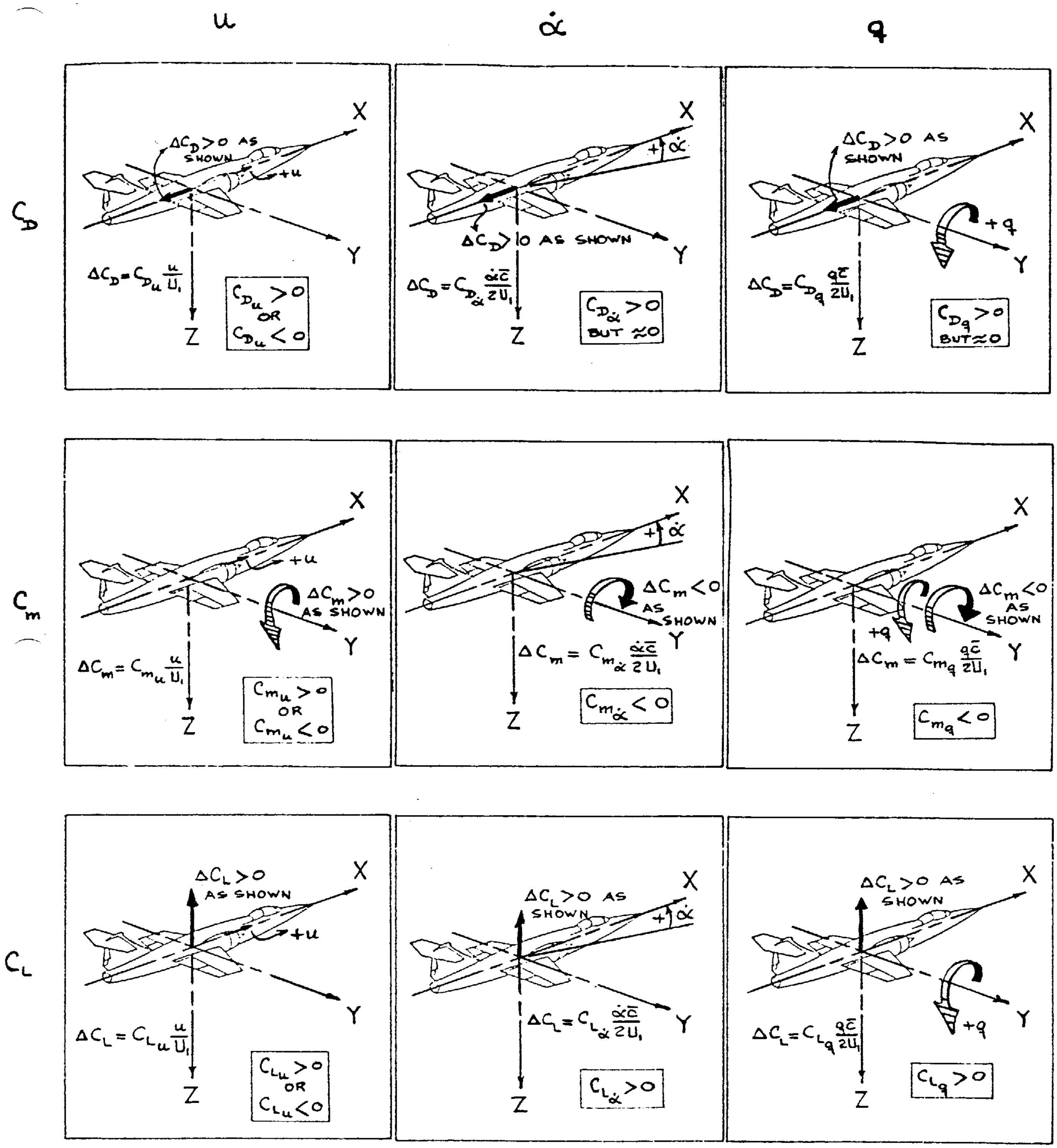
Perturbed Equations of Motion



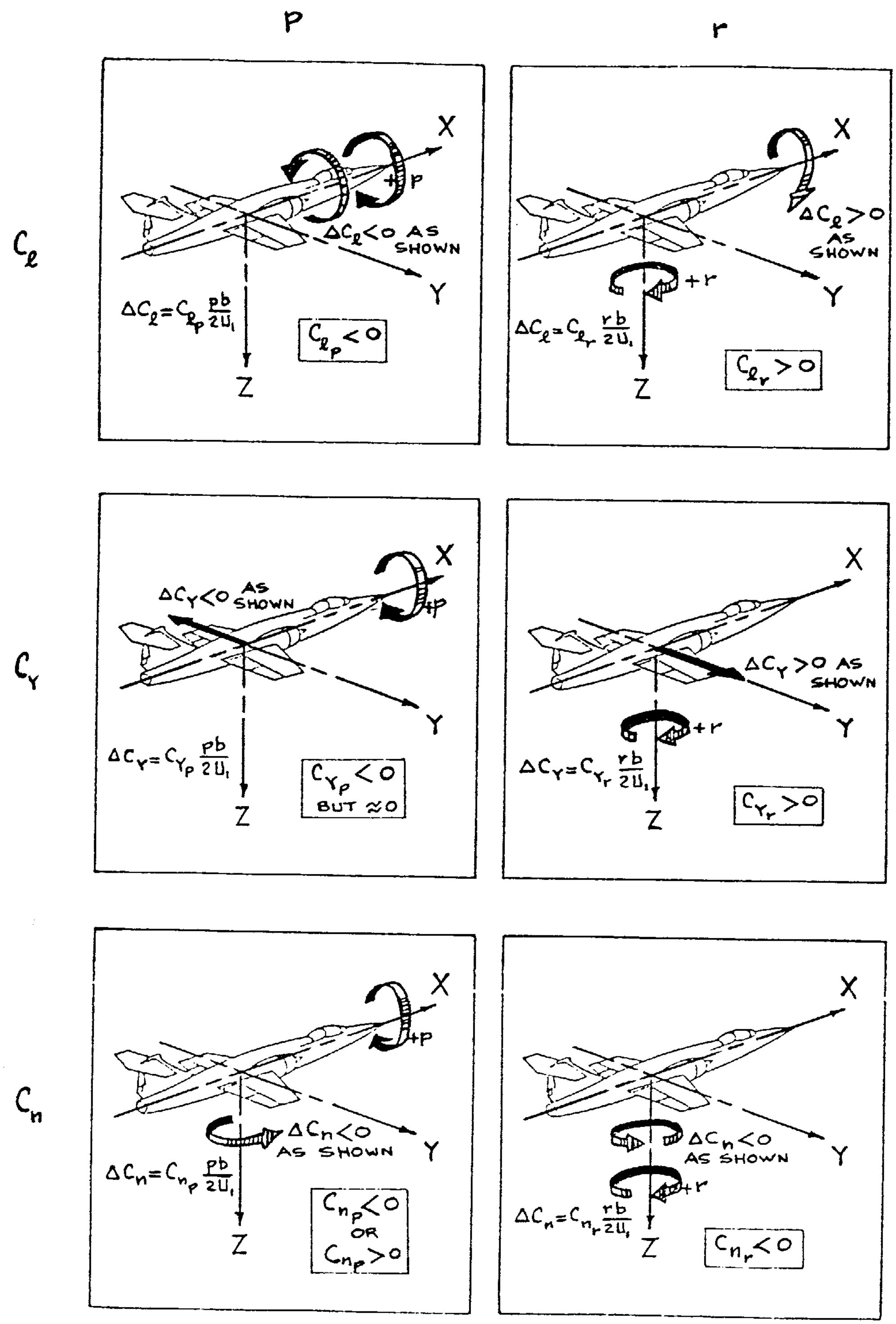
Summary of Longitudinal Steady State Force and Moment Derivatives



Summary of Lateral-Directional Steady State Force and Moment Derivatives



Summary of Longitudinal Speed, Angle-of-Attack Rate and Pitch Rate Derivatives



Summary of Lateral-Directional Roll and Yaw Rate Derivatives

Longitudinal Small Perturbation Equations

$$m\dot{u} = -mg\theta\cos\theta_1 + \bar{q}_1 S\{-(C_{D_u}+2C_{D_1})\frac{u}{U_1} + (C_{T_{x_u}}+2C_{T_{x_1}})\frac{u}{U_1}$$
$$-(C_{D_\alpha}-C_{L_1})\alpha - C_{D_{\delta_E}}\delta_E\}$$

$$m(\dot{w}-U_1 q) = -mg\theta\sin\theta_1 + \bar{q}_1 S\{-(C_{L_u}+2C_{L_1})\frac{u}{U_1} - (C_{L_\alpha}+C_{D_1})\alpha$$
$$-C_{L_{\dot{\alpha}}}\frac{\dot{\alpha}\bar{c}}{2U_1} - C_{L_q}\frac{q\bar{c}}{2U_1} - C_{L_{\delta_E}}\delta_E\}$$

$$I_{YY}\dot{q} = \bar{q}_1 S\bar{c}\{(C_{m_u}+2C_{m_1})\frac{u}{U_1} + (C_{m_{T_u}}+2C_{m_{T_1}})\frac{u}{U_1}$$
$$+C_{m_\alpha}\alpha + C_{m_{T_\alpha}}\alpha + C_{m_{\dot{\alpha}}}\frac{\dot{\alpha}\bar{c}}{2U_1} + C_{m_q}\frac{q\bar{c}}{2U_1} + C_{m_{\delta_E}}\delta_E\}$$

Lateral-Directional Small Perturbation Equations

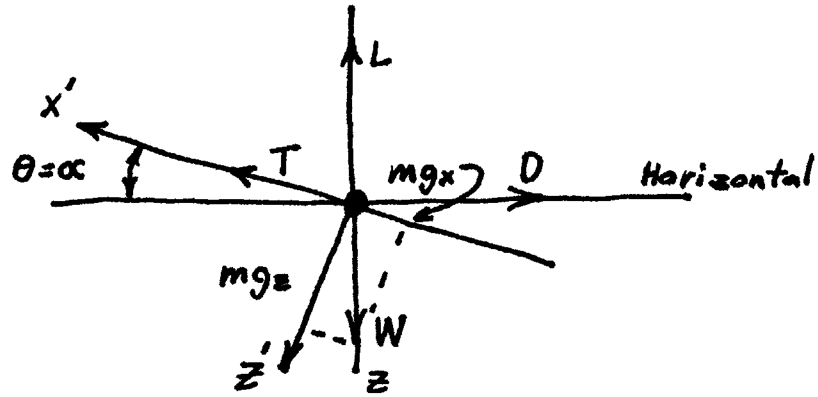
$$m(\dot{v}+U_1 r) = mg\phi\cos\theta_1 + \bar{q}_1 S(C_{Y_\beta}\beta + C_{Y_p}\frac{pb}{2U_1}$$
$$+C_{Y_r}\frac{rb}{2U_1} + C_{Y_{\delta_A}}\delta_A + C_{Y_{\delta_R}}\delta_R)$$

$$I_{XX}\dot{p} - I_{XZ}\dot{r} = \bar{q}_1 Sb(C_{\ell_\beta}\beta + C_{\ell_p}\frac{pb}{2U_1} + C_{\ell_r}\frac{rb}{2U_1}$$
$$+C_{\ell_{\delta_A}}\delta_A + C_{\ell_{\delta_R}}\delta_R)$$

$$I_{ZZ}\dot{r} - I_{XZ}\dot{p} = \bar{q}_1 Sb(C_{n_\beta}\beta + C_{n_{T_\beta}}\beta + C_{n_p}\frac{pb}{2U_1}$$
$$+C_{n_r}\frac{rb}{2U_1} + C_{n_{\delta_A}}\delta_A + C_{n_{\delta_R}}\delta_R)$$

Simple Maneuvers

Steady Level Flight



$$\dot\phi = \phi = 0 \quad \dot\psi = 0 \quad \psi = \text{constant} \quad \dot\theta = 0 \quad \theta = \text{constant}$$

$$\dot P = P = \dot Q = Q = \dot R = R = 0 \qquad \dot U = \dot V = \dot W = V = 0$$

$$m(\cancelto{0}{\dot U} - \cancelto{0}{VR} + \cancelto{0}{WQ}) = mg_x + F_{A_x} + F_{T_x} \qquad 0 = mg_x + F_{A_x} + F_{T_x}$$

$$m(\cancelto{0}{\dot V} + \cancelto{0}{UR} - \cancelto{0}{WP}) = \cancelto{0}{mg_y} + F_{A_y} + F_{T_y} \qquad 0 = F_{A_y} + F_{T_y}$$

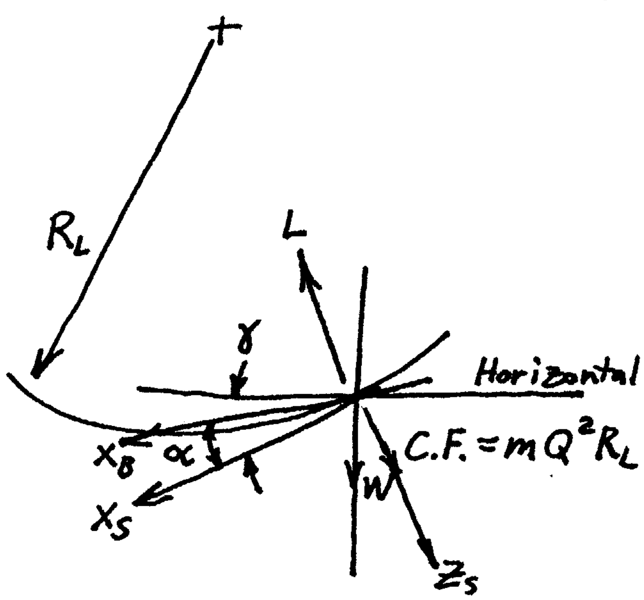
$$m(\cancelto{0}{\dot W} - \cancelto{0}{UQ} + \cancelto{0}{VP}) = mg_z + F_{A_z} + F_{T_z} \qquad 0 = mg_z + F_{A_z} + F_{T_z}$$

$$I_{yy}\cancelto{0}{\dot Q} + (I_{xx} - I_{zz})\cancelto{0}{PR} + I_{xz}(\cancelto{0}{P^2} - \cancelto{0}{R^2}) = M_A + M_T$$

$$0 = M_A + M_T$$

Symmetric Steady Pull Up

$$\dot\theta = Q \quad \dot\phi = \phi = 0 \quad \dot\psi = 0 \quad \psi = \text{constant}$$



$$0 = mg_x + F_{A_x} + F_{T_x}$$

$$0 = mg_y + F_{A_y} + F_{T_y}$$

$$mUQ = mg_z + F_{A_z} + F_{T_z}$$

$$I_{yy}\cancelto{0}{\dot Q} + (I_{xx} - I_{zz})\cancelto{0}{PR}$$

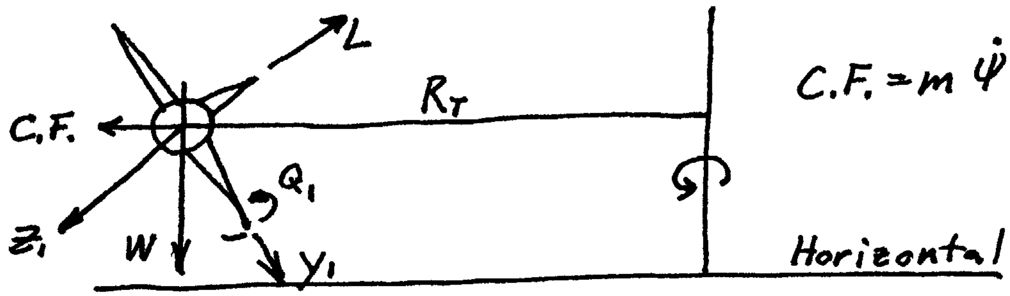
$$+ I_{xz}(\cancelto{0}{P^2} - \cancelto{0}{R^2}) = M_A + M_T$$

$$0 = M_A + M_T$$

Steady Level Turn

$$P = \dot{\phi} - \dot{\psi}\sin\theta \qquad Q = \dot{\theta}\cos\phi + \dot{\psi}\sin\phi\cos\theta$$

$$R = -\dot{\theta}\sin\phi + \dot{\psi}\cos\phi\cos\theta$$



$$C.F. = m\dot{\psi}^2 R_T$$

$\dot{\phi} = 0 \quad \phi = \text{constant} \quad \dot{\psi} = \text{constant} \quad \dot{\theta} = 0 \quad \theta = \text{constant}$

$$P = -\dot{\psi}\sin\theta \qquad Q = \dot{\psi}\sin\phi\cos\theta \qquad R = \dot{\psi}\cos\phi\cos\theta$$

$$m(\cancelto{0}{\dot{U}} - VR + WQ) = mg_x + F_{A_x} + F_{T_x}$$

$$m(\cancelto{0}{\dot{V}} + UR - WP) = mg_y + F_{A_y} + F_{T_y}$$

$$m(\cancelto{0}{\dot{W}} - UQ + VP) = mg_z + F_{A_z} + F_{T_z}$$

$$(I_{zz} - I_{yy})RQ = L_A + L_T$$

$$-I_{xz}R^2 = M_A + M_T$$

$$I_{xz}QR = N_A + N_T$$

Steady Roll (Body Axis aligned with roll axis, $\theta = 0$)

This is a level roll axis...

$$\dot{\psi} = \psi = 0 \qquad \dot{\phi} = \text{constant} \qquad \dot{\theta} = \theta = 0$$

$$\dot{P} = 0 \quad P = \dot{\phi} \quad \dot{Q} = Q = 0 \quad \dot{R} = R = 0$$

$$\alpha = \text{constant} \Rightarrow \frac{W}{V} = \text{constant}$$

$$m(\dot{U} - V\cancelto{0}{R} + W\cancelto{0}{Q}) = mg_x + F_{A_x} + F_{T_x}$$

$$m\dot{U} = mg_x + F_{A_x} + F_{T_x}$$

$$m(\dot{V} + U\cancelto{0}{R} - WP) = mg_y + F_{A_y} + F_{T_y}$$

$$m(\dot{V} - WP) = mg_y + F_{A_y} + F_{T_y}$$

$$m(\dot{W} - U\cancelto{0}{Q} + VP) = mg_z + F_{A_z} + F_{T_z}$$

$$m(\dot{W} + VP) = mg_z + F_{A_z} + F_{T_z}$$

$$I_{xx}\cancelto{0}{\dot{P}} - I_{xz}\cancelto{0}{\dot{R}} - I_{xz}P\cancelto{0}{Q} + (I_{zz} - I_{yy})R\cancelto{0}{Q} = L_A + L_T$$

$$0 = L_A + L_T$$

$$I_{yy}\cancelto{0}{\dot{Q}} + (I_{xx} - I_{zz})P\cancelto{0}{R} + I_{xz}(P^2 - \cancelto{0}{R^2}) = M_A + M_T$$

$$I_{xz}P^2 = M_A + M_T$$

$$I_{zz}\cancelto{0}{\dot{R}} - I_{xz}\cancelto{0}{\dot{P}} + (I_{yy} - I_{xx})P\cancelto{0}{Q} + I_{xz}\cancelto{0}{Q}R = N_A + N_T$$

$$0 = N_A + N_T$$

But how do you accelerate? At $\dot{P} \neq 0$ which means

$$I_{xx}\dot{P} = L_A + L_T$$

$$-I_{xz}\dot{P} = N_A + N_T$$

Aircraft Stability and Control Short Course

Savannah Section of AIAA

February 2, 1993

Walter Lounsbery
Course Coordinator
966-4486

SESSION III - Longitudinal Aero Derivatives and Longitudinal Static Stability

+ Please Register Attendence

+ No session February 16

+ More Wing and Control Aerodynamics
  - Airfoil Section Properties
  - Wing Aerodynamic Center
  - Drag
  - Control Gearing and Hinge Moments
  - Lift, Drag, Sideforce, and Beta

+ Break: Video of NASA kinematic experiments

+ Longitudinal Derivatives
  - Drag
  - Lift
  - Pitch Moment

+ Longitudinal Static Stability
  - Lift and Speed
  - Angle of Attack
  - Influence of Zero-Lift Pitch Moment
  - Some General Conclusions

More Wing and Control Surface Aerodynamics

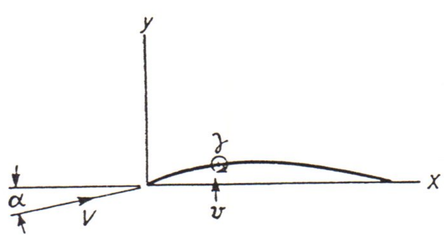
Airfoil Section Properties – Thin Airfoil Theory

see Abbot, Ira H. & Von Doenhoff, Albert E., "Theory of Wing Sections", Dover, New York, 1959

Thin airfoil theory is a key to understanding airfoil section behavior.

The airfoil can be reduced to a zero thickness mean line that can be theoretically analyzed for:

1. The chordwise load distribution
2. The angle of zero lift
3. The pitching moment coefficient
4. Slope of the lift curve
5. Approximate aerodynamic center



Configuration for analysis of mean lines.

- The thin airfoil is a stream line

Total circulation $\Gamma = \int_0^c \gamma \, dx$

$$V_n(x_1) = \int_0^c \frac{\gamma \, dx}{2\pi(x - x_1)}$$

$$\cancel{\alpha_0 =} \; \alpha_0 + \frac{V_n}{V} = \frac{dy}{dx}$$

Let $x = \frac{c}{2}(1-\cos\theta) \qquad dx = \frac{c}{2}\sin\theta \, d\theta$

$$\gamma = 2V\left(A_0 \cot\frac{\theta}{2} + \sum_{n=1}^{\infty} A_n \sin n\theta\right)$$

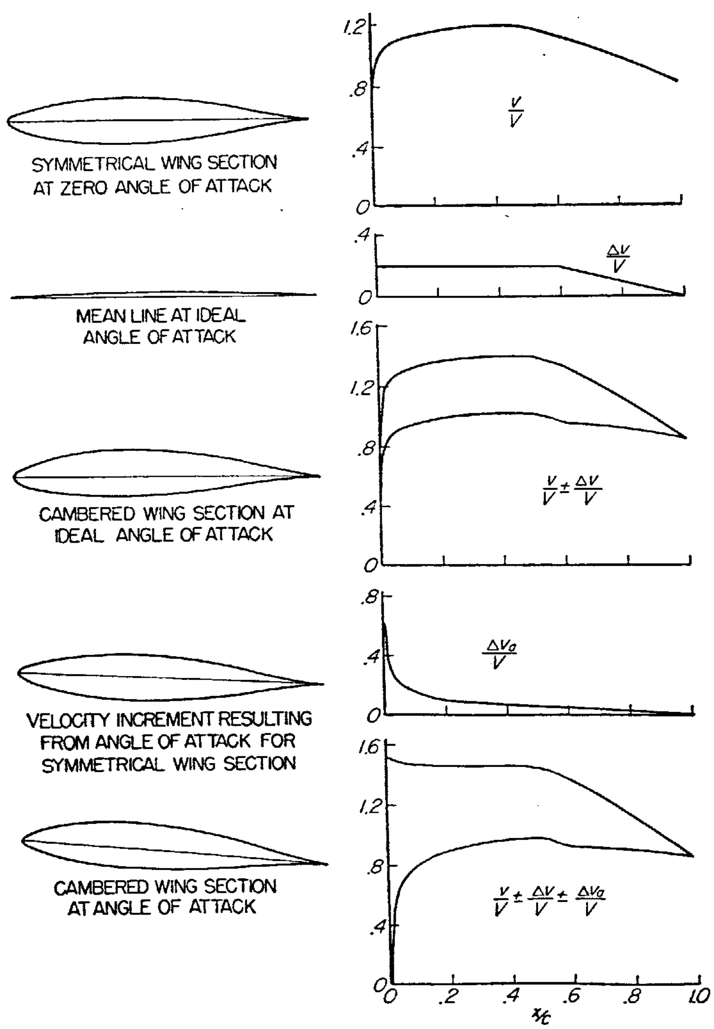
$$c_l = 2\pi\left(A_0 + \tfrac{1}{2}A_1\right) \qquad C_{m_{LE}} = \frac{\pi}{4}(A_2 - A_1) - \frac{1}{4}c_l$$

defining $\beta_0 = \frac{2}{\pi}\int_c^{\pi} \frac{y}{c}\frac{d\theta}{1+\cos\theta} \qquad \mu_0 = \int_c^{\pi} \frac{y}{c}\cos\theta \, d\theta$

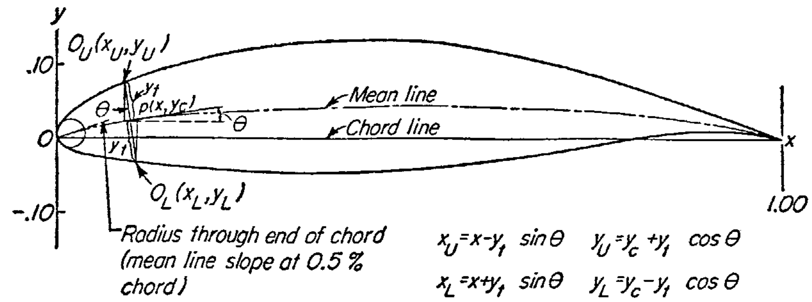
then $\beta_0 = A_0 - \alpha_0 + \frac{1}{2}A_1 \qquad \mu_0 = -\frac{\pi}{4}\left(\alpha_0 - A_0 - \frac{1}{2}A_2\right)$

$$c_l = 2\pi(\alpha_0 + \beta_0)$$

$$C_{m_{LE}} = \left(2\mu_0 - \frac{\pi}{2}\beta_0\right) - \frac{1}{4}c_l \qquad C_{m_{c/4}} = 2\mu_0 - \frac{\pi}{2}\beta_0$$



Synthesis of pressure distribution.



Sample calculations for derivation of the NACA 65,3-818 airfoil ($a = 1.0$)											
x	y_t*	y_c†	$\tan\theta$	$\sin\theta$	$\cos\theta$	$y_t \sin\theta$	$y_t \cos\theta$	x_U	y_U	x_L	y_L
0	0	0	....	....	......	0	0	0	0	0	0
0.005	0.01324	0.00200	0.33696‡	0.31932	0.94765	0.00423	0.01255	0.00077	0.01455	0.00923	−0.01055
0.05	0.03831	0.01264	0.18744	0.18422	0.98288	0.00706	0.03765	0.04294	0.05029	0.05706	−0.02501
0.25	0.08093	0.03580	0.06996	0.06979	0.99756	0.00565	0.08073	0.24435	0.11653	0.25565	−0.04493
0.50	0.08593	0.04412	0	0	1.00000	0	0.08593	0.50000	0.13005	0.50000	−0.04181
0.75	0.04456	0.03580	−0.06996	−0.06979	0.99756	−0.00311	0.04445	0.75311	0.08025	0.74689	−0.00865
1.00	0	0	....	....	......	0	0	1.00000	0	1.00000	0

* Thickness distribution obtained from ordinates of the NACA 65,3-018 airfoil.
† Ordinates of the mean line, 0.8 of the ordinate for $c_{l_i} = 1.0$.
‡ Slope of radius through end of chord.

Method of combining mean lines and basic-thickness forms.

NACA 63_2-215

(Stations and ordinates given in per cent of airfoil chord)

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
0.399	1.250	0.601	− 1.150
0.637	1.528	0.863	− 1.388
1.120	1.980	1.380	− 1.766
2.348	2.792	2.652	− 2.420
4.829	3.960	5.171	− 3.328
7.323	4.847	7.677	− 3.999
9.823	5.569	10.177	− 4.535
14.834	6.682	15.166	− 5.336
19.852	7.487	20.148	− 5.895
24.875	8.049	25.125	− 6.259
29.900	8.392	30.100	− 6.448
34.926	8.530	35.074	− 6.470
39.952	8.457	40.048	− 6.315
44.977	8.194	45.023	− 6.004
50.000	7.768	50.000	− 5.562
55.019	7.203	54.981	− 5.013
60.035	6.524	59.965	− 4.382
65.047	5.751	64.953	− 3.691
70.053	4.906	69.947	− 2.962
75.055	4.014	74.945	− 2.224
80.051	3.105	79.949	− 1.513
85.043	2.213	84.957	− 0.867
90.030	1.368	89.970	− 0.334
95.014	0.616	94.986	0.016
100.000	0	100.000	0

L.E. radius: 1.594
Slope of radius through L.E.: 0.0842

NACA 63_2-415

(Stations and ordinates given in per cent of airfoil chord)

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
0.300	1.287	0.700	− 1.087
0.525	1.585	0.975	− 1.305
0.991	2.074	1.509	− 1.646
2.198	2.964	2.802	− 2.220
4.660	4.264	5.340	− 3.000
7.147	5.261	7.853	− 3.565
9.647	6.077	10.353	− 4.009
14.669	7.348	15.331	− 4.656
19.705	8.279	20.295	− 5.095
24.750	8.941	25.250	− 5.361
29.800	9.362	30.200	− 5.474
34.852	9.559	35.148	− 5.439
39.905	9.527	40.095	− 5.243
44.955	9.289	45.045	− 4.909
50.000	8.871	50.000	− 4.459
55.039	8.298	54.961	− 3.918
60.070	7.595	59.930	− 3.311
65.093	6.780	64.907	− 2.660
70.106	5.877	69.894	− 1.989
75.109	4.907	74.891	− 1.327
80.102	3.900	79.898	− 0.716
85.085	2.885	84.915	− 0.193
90.059	1.884	89.941	0.184
95.028	0.931	94.972	0.333
100.000	0	100.000	0

L.E. radius: 1.594
Slope of radius through L.E.: 0.1685

NACA 63_2-615

(Stations and ordinates given in per cent of airfoil chord)

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
0.205	1.317	0.795	− 1.017
0.418	1.634	1.082	− 1.214
0.866	2.159	1.634	− 1.517
2.050	3.129	2.950	− 2.013
4.492	4.560	5.508	− 2.664
6.973	5.667	8.027	− 3.123
9.473	6.578	10.527	− 3.476
14.504	8.010	15.496	− 3.972
19.558	9.066	20.442	− 4.290
24.625	9.830	25.375	− 4.460
29.700	10.331	30.300	− 4.499
34.778	10.587	35.222	− 4.407
39.857	10.598	40.143	− 4.172
44.932	10.384	45.068	− 3.814
50.000	9.974	50.000	− 3.356
55.058	9.393	54.942	− 2.823
60.105	8.665	59.895	− 2.239
65.139	7.809	64.861	− 1.629
70.159	6.847	69.841	− 1.015
75.163	5.800	74.837	− 0.430
80.153	4.693	79.847	0.083
85.127	3.555	84.873	0.483
90.089	2.398	89.911	0.704
95.042	1.245	94.958	0.651
100.000	0	100.000	0

L.E. radius: 1.594
Slope of radius through L.E.: 0.2527

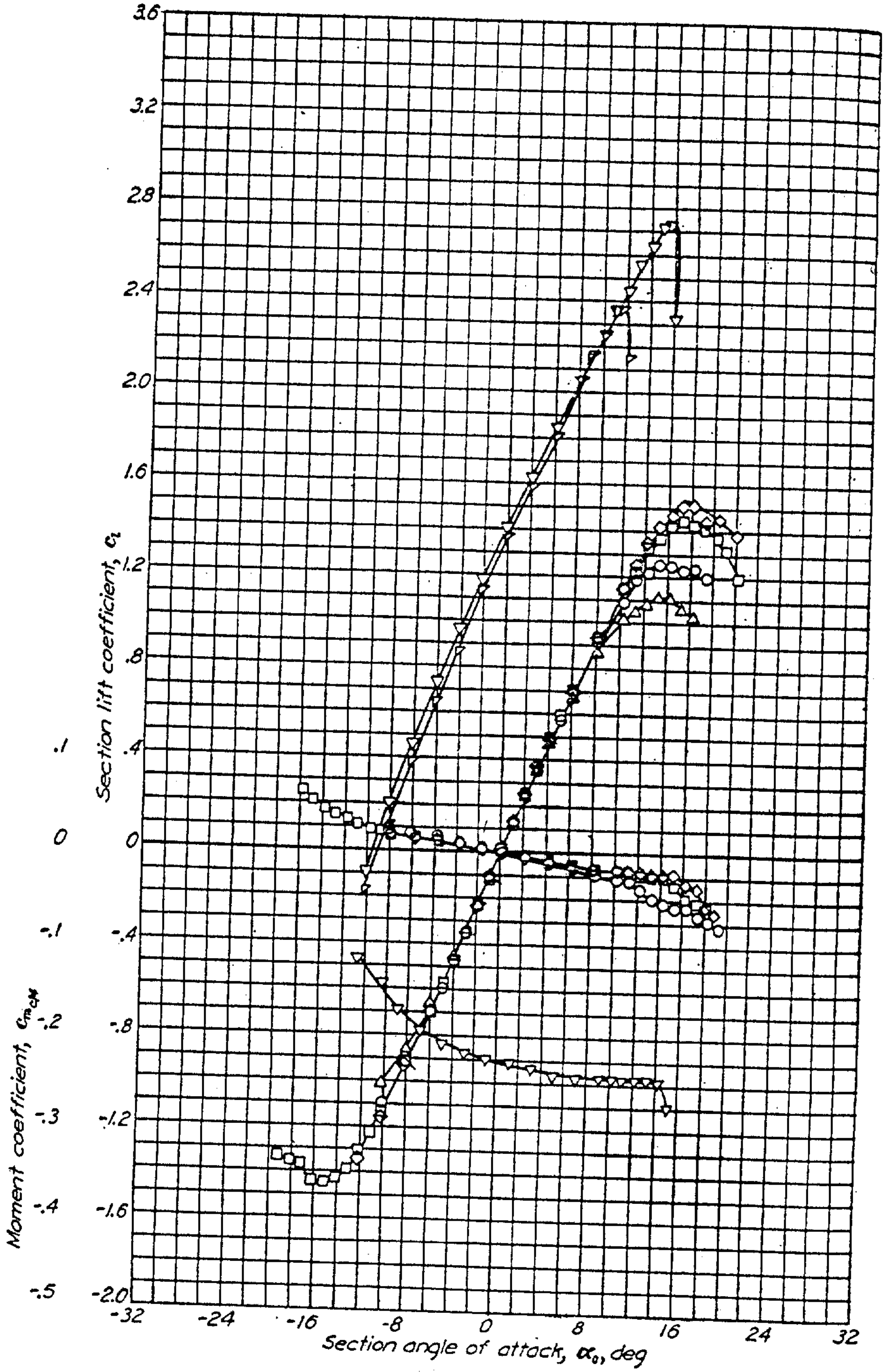
NACA 63_3-218

(Stations and ordinates given in per cent of airfoil chord)

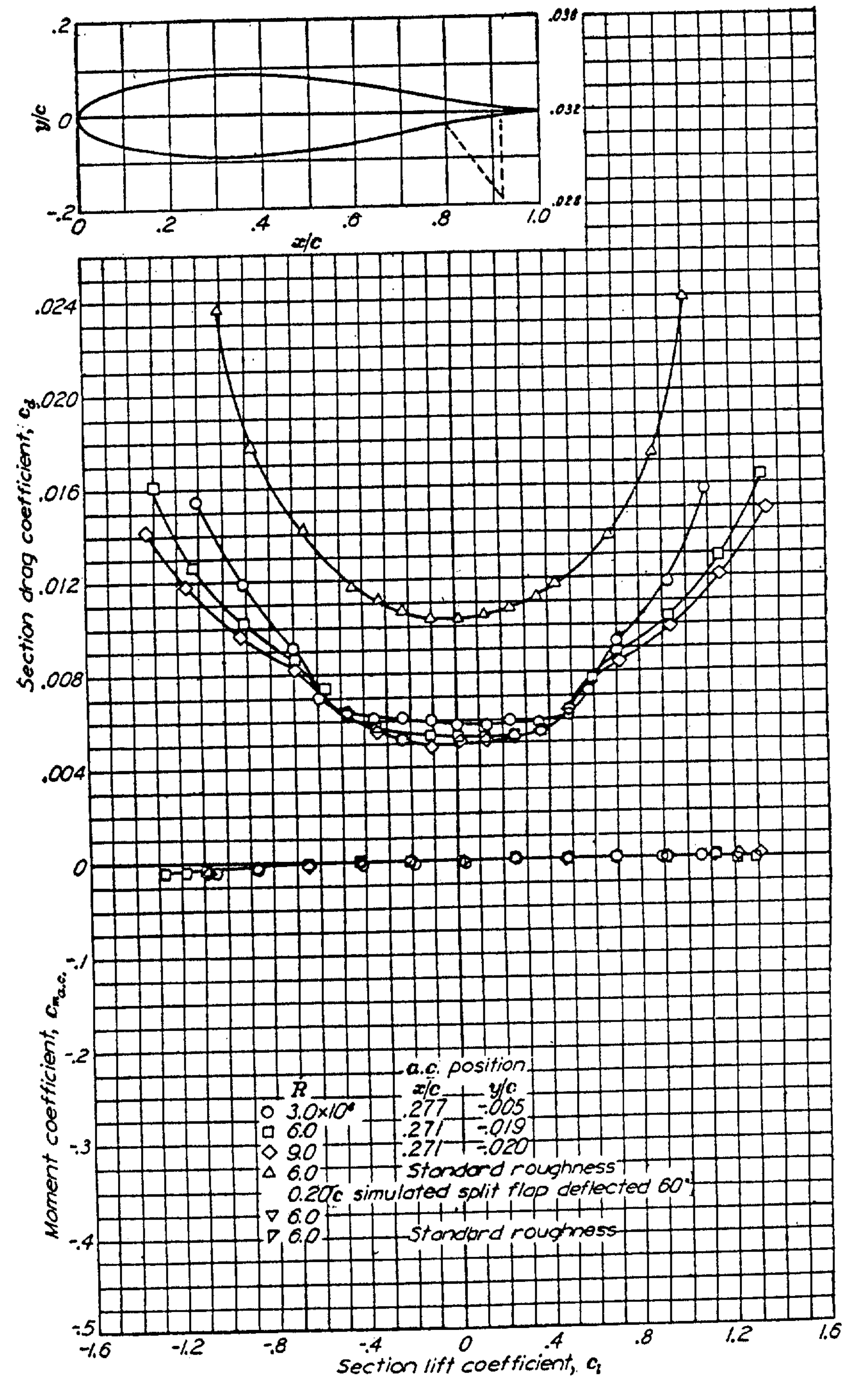
Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
0.382	1.449	0.618	− 1.349
0.617	1.778	0.883	− 1.638
1.096	2.319	1.404	− 2.105
2.319	3.285	2.681	− 2.913
4.796	4.673	5.204	− 4.041
7.288	5.728	7.712	− 4.880
9.788	6.581	10.212	− 5.547
14.801	7.895	15.199	− 6.549
19.822	8.842	20.178	− 7.250
24.850	9.494	25.150	− 7.704
29.880	9.884	30.120	− 7.940
34.911	10.030	35.089	− 7.970
39.943	9.916	40.057	− 7.774
44.973	9.577	45.027	− 7.387
50.000	9.045	50.000	− 6.839
55.023	8.351	54.977	− 6.161
60.042	7.526	59.958	− 5.384
65.055	6.597	64.945	− 4.537
70.062	5.594	69.938	− 3.650
75.064	4.544	74.936	− 2.754
80.059	3.486	79.941	− 1.894
85.049	2.459	84.951	− 1.113
90.034	1.501	89.966	− 0.467
95.016	0.664	94.984	− 0.032
100.000	0	100.000	0

L.E. radius: 2.120
Slope of radius through L.E.: 0.0842

4

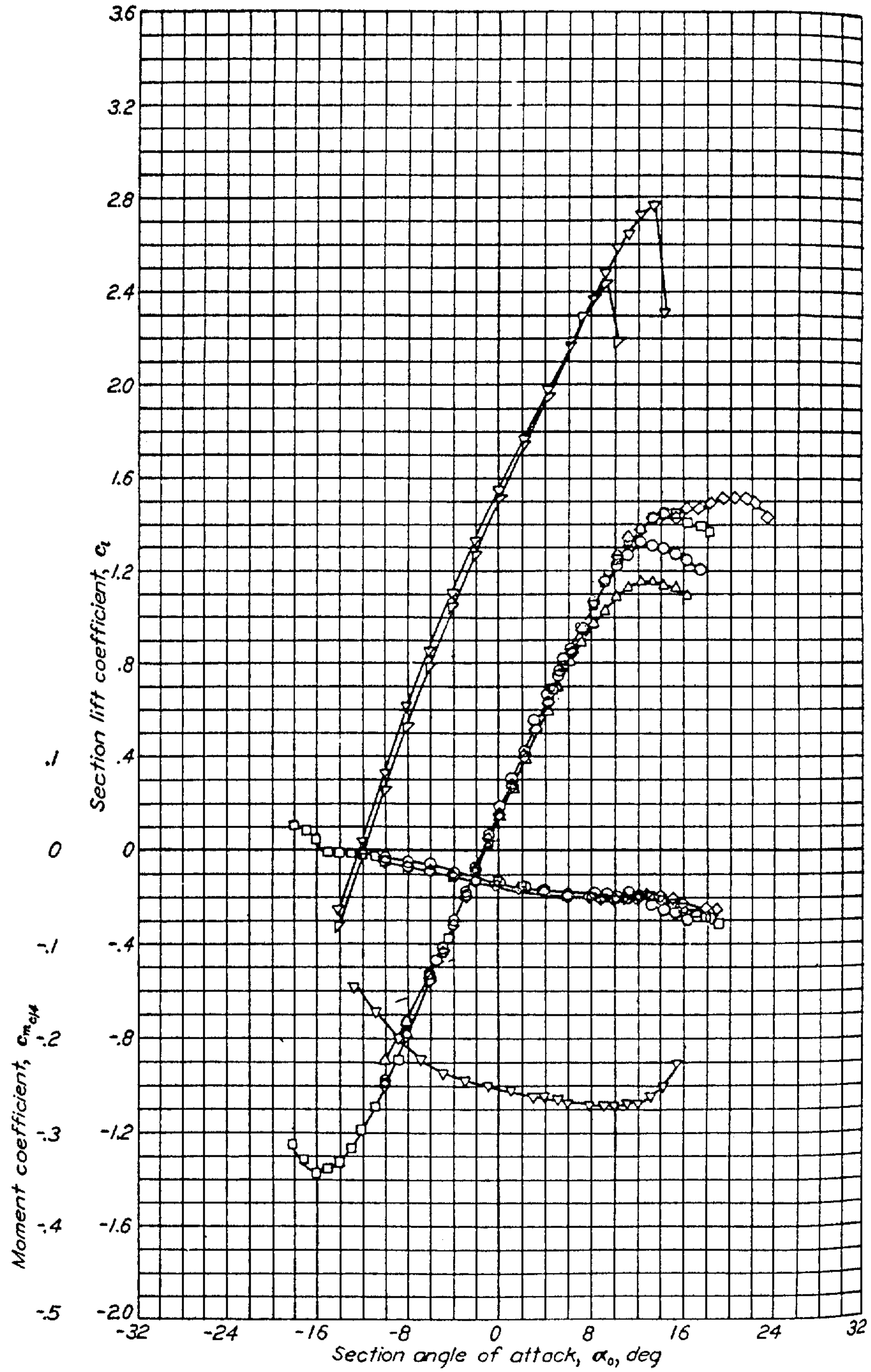


NACA 63_3-018 Wing Section

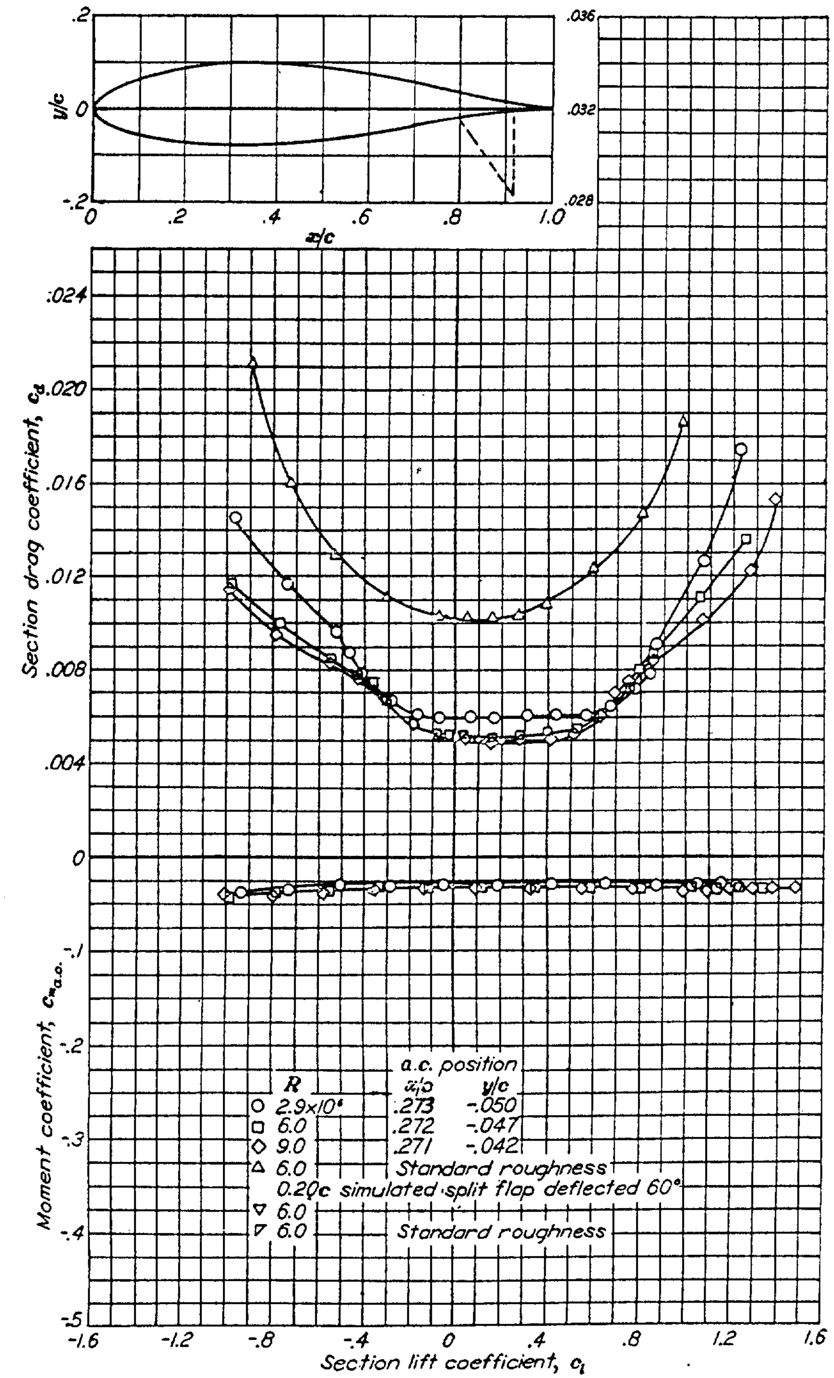


NACA 63_3-018 Wing Section (*Continued*)

5

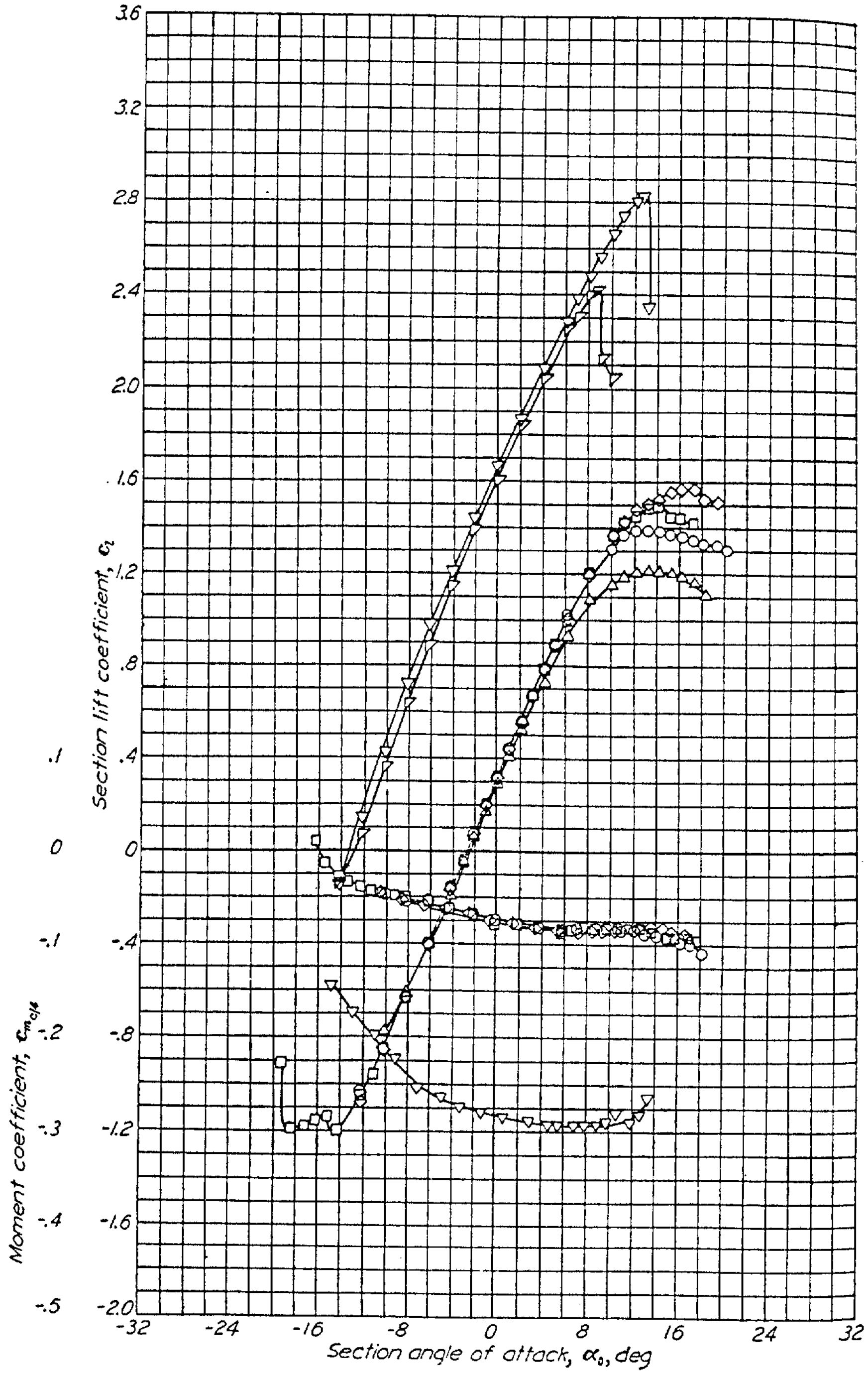


NACA 63_3-218 Wing Section

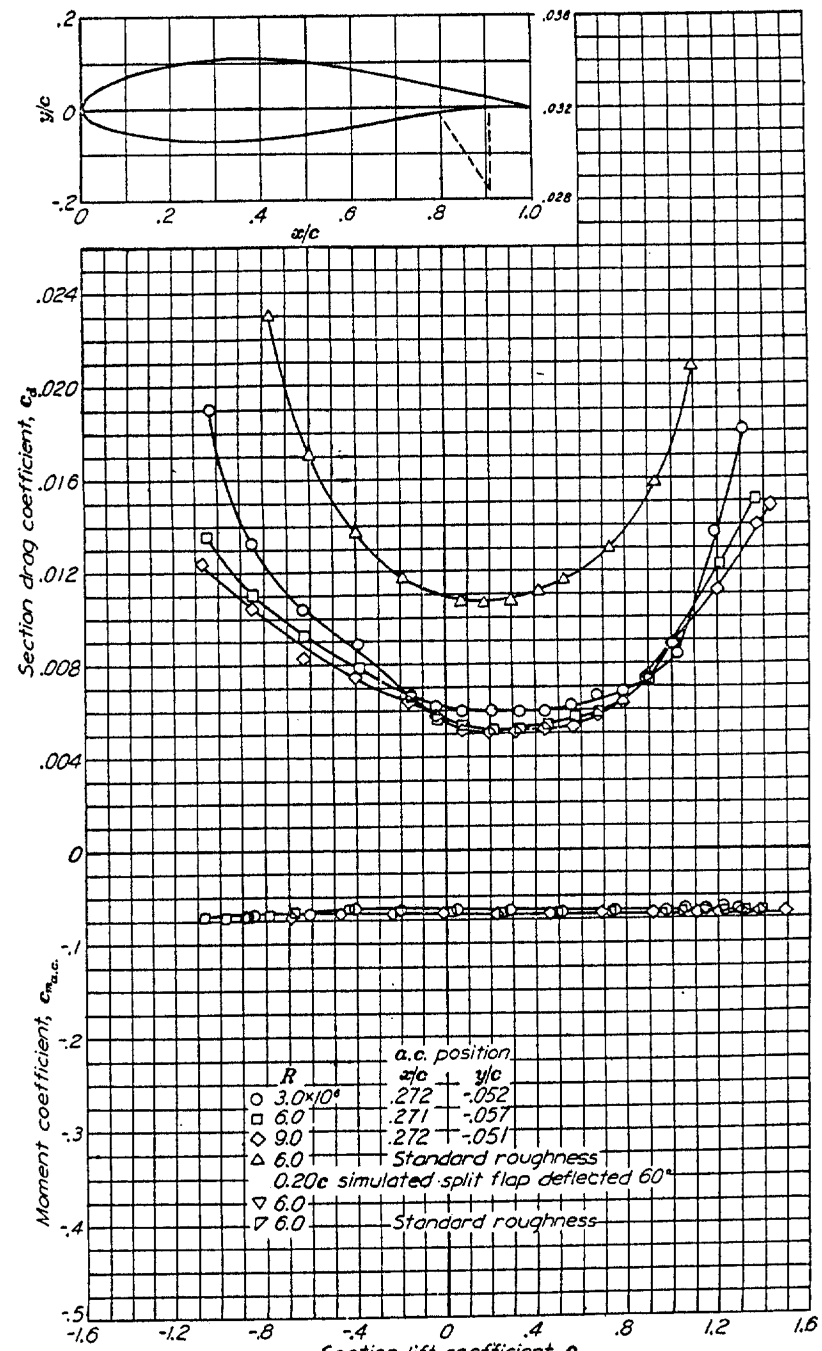


NACA 63_3-218 Wing Section (*Continued*)

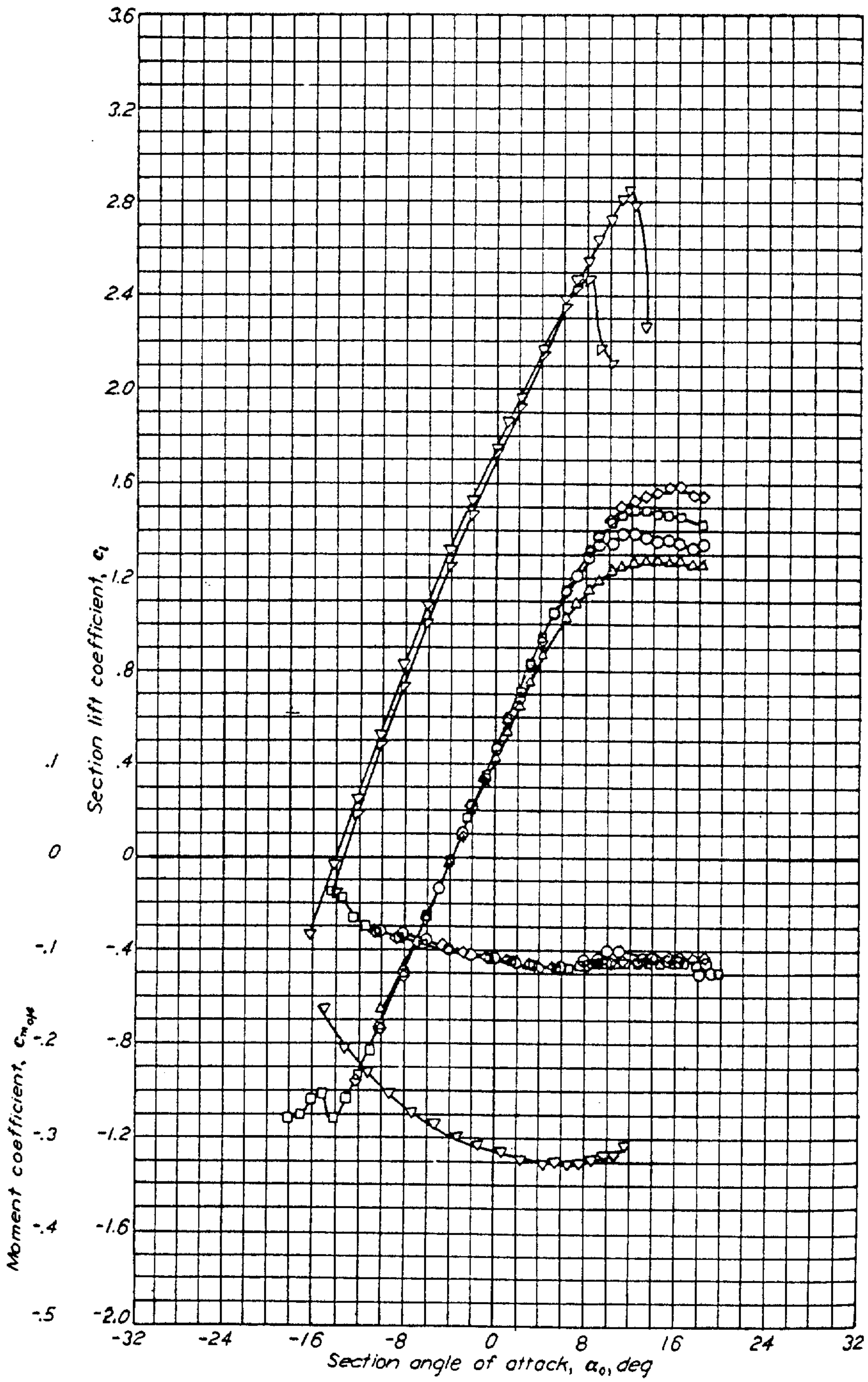
6



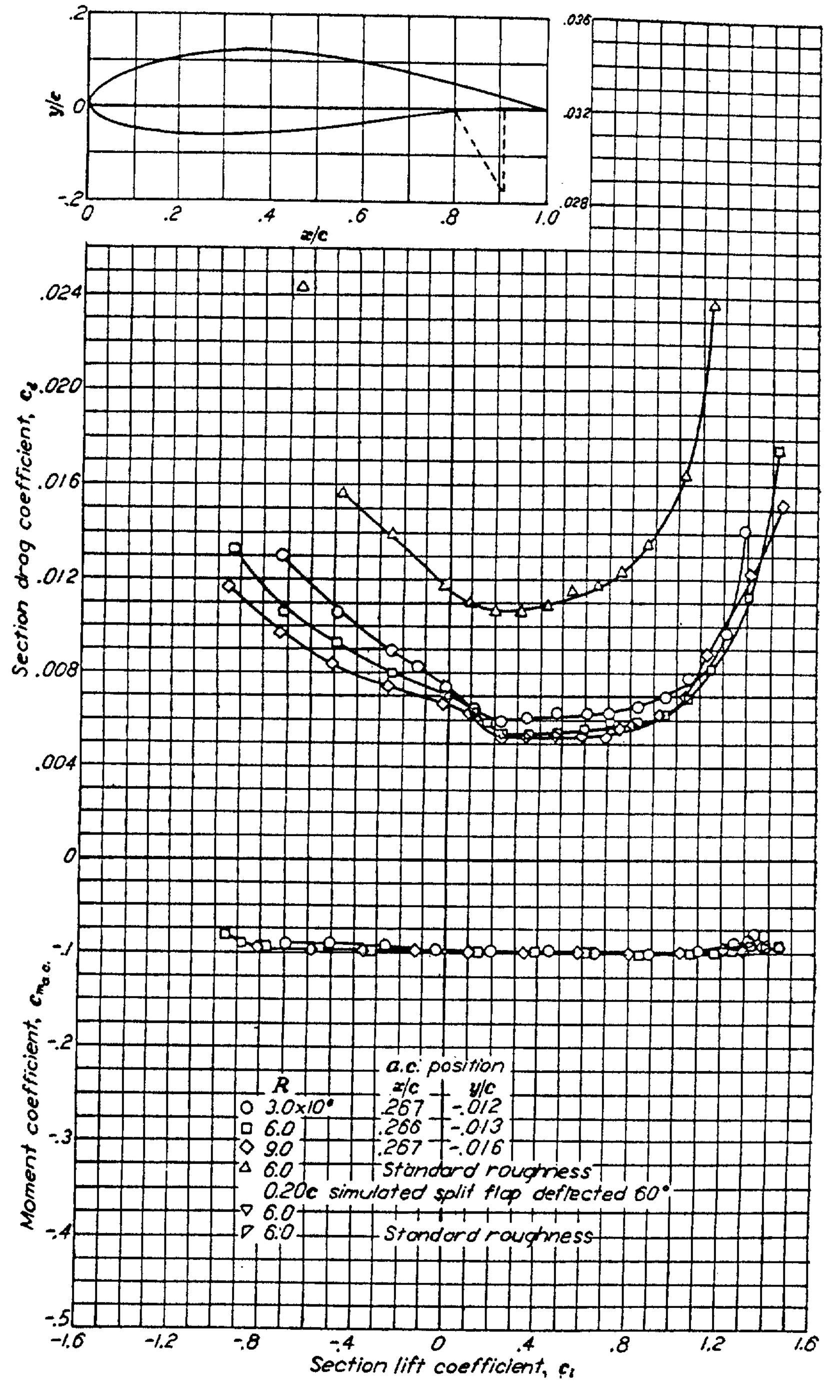
NACA 63_3-418 Wing Section



NACA 63_3-418 Wing Section (*Continued*)



NACA 63_3-618 Wing Section



NACA 63_3-618 Wing Section (*Continued*)

8

```
REM Airfoil CMAC program
REM
REM Walt Lounsbery, 1-31-93
REM
REM Data statements in data file are in this order
REM "Airfoil Name", Coord. % Flag (1=% of chord, 0=ratio to chord)
REM Upper surface coord. pairs as
REM chordline distance, surface distance above reference line
REM lower surface coord. pairs as for upper surface
REM enter coordinates from LE to TE
REM
DIM U(2,60),L(2,60)
CLS: PRINT "AIRFOIL CMAC PROGRAM": PRINT: INPUT "AIRFOIL FILE NAME";N$
PRINT: INPUT "ZERO LIFT ANGLE";A
OPEN N$ FOR INPUT AS #1
INPUT #1,N$,FC
FOR J=1 TO 60: INPUT #1,U(1,J): INPUT #1,U(2,J)
IF FC=0 THEN
  IF U(1,J)=1 THEN GOTO QUITUPPER
ELSE
  IF U(1,J)=100 THEN GOTO QUITUPPER
END IF
NEXT J
QUITUPPER:
UN=J
FOR J=1 TO 60: INPUT #1,L(1,J): INPUT #1,L(2,J)
IF FC=0 THEN
  IF U(1,J)=1 THEN GOTO QUITLOWER
ELSE
  IF U(1,J)=100 THEN GOTO QUITLOWER
END IF
NEXT J
QUITLOWER:
LN=J: CLOSE 1
LPRINT: LPRINT ">>>>> ZERO LIFT MOMENT CALCULATION <<<<<"
LPRINT: LPRINT: LPRINT "AIRFOIL: ";N$
LPRINT: LPRINT"*** UPPER ***";SPACE$(12);"*** LOWER ***"
LPRINT "X/C";SPACE$(11);"Z/C";SPACE$(11);"X/C";SPACE$(11);"Z/C %": LPRINT
D=UN: IF LN>UN THEN D=LN
FOR J=1 TO D: LPRINT " ";
IF J>UN THEN GOTO SKIPUPPER
U1$=STR$(U(1,J)):U2$=STR$(U(2,J)): D1=14-LEN(U1$): D2=14-LEN(U2$)
LPRINT U1$;SPACE$(D1);U2$;SPACE$(D2);
GOTO LOWPRINT
SKIPUPPER:
LPRINT SPACE$(28);
LOWPRINT:
L1$=STR$(L(1,J)): L2$=STR$(L(2,J)): D1=14-LEN(L1$): D2=14-LEN(L2$)
LPRINT L1$;SPACE$(D1);L2$;SPACE$(D2);
LPRINT " ": NEXT J
A=A*3.14159/180
IF FC=1 THEN
  FOR J=1 TO UN: FOR I=1 TO 2: U(I,J)=U(I,J)/100: NEXT I: NEXT J
  FOR J=1 TO LN: FOR I=1 TO 2: L(I,J)=L(I,J)/100: NEXT I: NEXT J
```

```
END IF
IN=0: XC=.01: GOSUB MEANLINE
IA=(-1)/((1-XC)*SQR(XC-XC*XC))*ZM
IN=ZM*(1-2*XC)/SQR(XC-XC*XC): XC=.99: GOSUB MEANLINE
Q1=ZM*(1-2*XC)/SQR(XC-XC*XC): Q2=-1/((1-XC)*SQR(XC-XC*XC))
IN=(IN+Q1)*.005: IA=(IA+ZM*Q2)*.005
FOR XC=.01 TO .99 STEP .01
  GOSUB MEANLINE
  Q1=ZM*(1-2*XC)/SQR(XC-XC*XC): Q2=-1/((1-XC)*SQR(XC-XC*XC))
  IN=IN+Q1*.01: IA=IA+Q2*ZM*.01
NEXT XC
C1=2*IN+IA/2: C2=2*IN-3.14159*A/2
LPRINT: LPRINT "ZERO LIFT MOMENT COEFFICIENT:"
LPRINT: LPRINT "     VIA COMPUTED ZERO LIFT ANGLE =";C1
LPRINT: LPRINT "     VIA ACTUAL ZERO LIFT ANGLE =";C2
LPRINT: LPRINT "COMPUTED ZERO LIFT ANGLE IS";IA*180/(3.14159^2);" DEGREES"
LPRINT: LPRINT "ACTUAL ZERO LIFT ANGLE IS";-A*180/3.14159;" DEGREES"
END
MEANLINE:
FOR J=2 TO UN
IF XC<=U(1,J) THEN GOTO UFOUND
NEXT J
UFOUND:
D=(XC-U(1,J-1))/(U(1,J)-U(1,J-1))
UC=D*(U(2,J)-U(2,J-1))+U(2,J-1)
FOR J=2 TO LN
IF XC<=L(1,J) THEN GOTO LFOUND
NEXT J
LFOUND:
D=(XC-L(1,J-1))/(L(1,J)-L(1,J-1))
LC=D*(L(2,J)-L(2,J-1))+L(2,J-1)
ZM=(LC+UC)/2
RETURN
```

>>>>> ZERO LIFT MOMENT CALCULATION <<<<<

AIRFOIL: NACA 63(3)-218

*** UPPER ***		*** LOWER ***	
X/C	Z/C	X/C	Z/C %
0	0	0	0
.382	1.449	.618	-1.349
.617	1.778	.883	-1.638
1.096	2.319	1.404	-2.105
2.319	3.285	2.681	-2.913
4.796	4.673	5.204	-4.041
7.288	5.728	7.712	-4.88
9.788	6.581	10.212	-5.547
14.801	7.895	15.199	-6.549
19.822	8.842	20.178	-7.25
24.85	9.494	25.15	-7.704
29.88	9.884	30.12	-7.94
34.911	10.03	35.089	-7.97
39.943	9.916	40.057	-7.774
44.973	9.577	45.027	-7.387
50	9.045	50	-6.839
55.023	8.351	54.977	-6.161
60.042	7.526	59.958	-5.384
65.055	6.597	64.945	-4.537
70.062	5.594	69.938	-3.65
75.064	4.544	74.936	-2.754
80.059	3.486	79.941	-1.894
85.049	2.459	84.951	-1.113
90.034	1.501	89.966	-.467
95.016	.664	94.984	-.032
100	0	100	0

ZERO LIFT MOMENT COEFFICIENT:

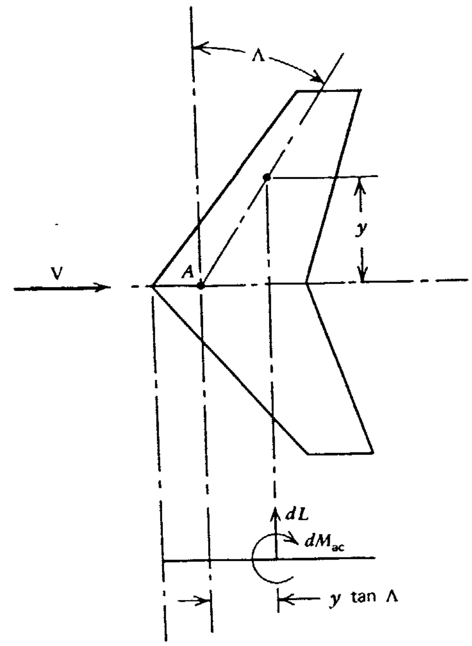
VIA COMPUTED ZERO LIFT ANGLE =-3.895856E-02

VIA ACTUAL ZERO LIFT ANGLE = 1.335671E-03

COMPUTED ZERO LIFT ANGLE IS-1.46976 DEGREES

ACTUAL ZERO LIFT ANGLE IS 0 DEGREES

Wing Aerodynamic Center - Trapezoidal Wing



Calculation of wing aerodynamic center.

$$M_A = q\int_{-b/2}^{b/2} c^2 C_{mac}\, dy - q\int_{-b/2}^{b/2} c\, C_l\, y \tan\Lambda\, dy$$

$$M_{ac} = M_A + L\, X_A$$

$$C_{M_{ac}} = C_{M_A} + C_L \frac{X_A}{\bar{c}}$$

$$0 = \frac{d C_{MA}}{d\alpha} + \frac{X_A}{\bar{c}} C_{L\alpha} \qquad \frac{dC_{Mac}}{d\alpha} = 0$$

$$X_A = \frac{1}{C_{L\alpha} S}\int_{-b/2}^{b/2} c\, C_{l\alpha}\, y \tan\Lambda\, dy$$

if $C_{l\alpha} = \text{constant}$ $\quad X_A = \tan\Lambda \left[\frac{\int_0^{b/2} c\, y\, dy}{S/2}\right] = \bar{y}\tan\Lambda$

if $\lambda = \frac{C_{TIP}}{C_{ROOT}}$ $\quad X_A = \left(\frac{1+2\lambda}{1+\lambda}\right)\frac{1}{3}\left(\frac{b}{2}\tan\Lambda\right)$

$$\bar{c} = \frac{1}{S}\int_{-b/2}^{b/2} c^2\, dy = \frac{2\, c_R}{3}\,\frac{1+\lambda+\lambda^2}{1+\lambda}$$

Drag

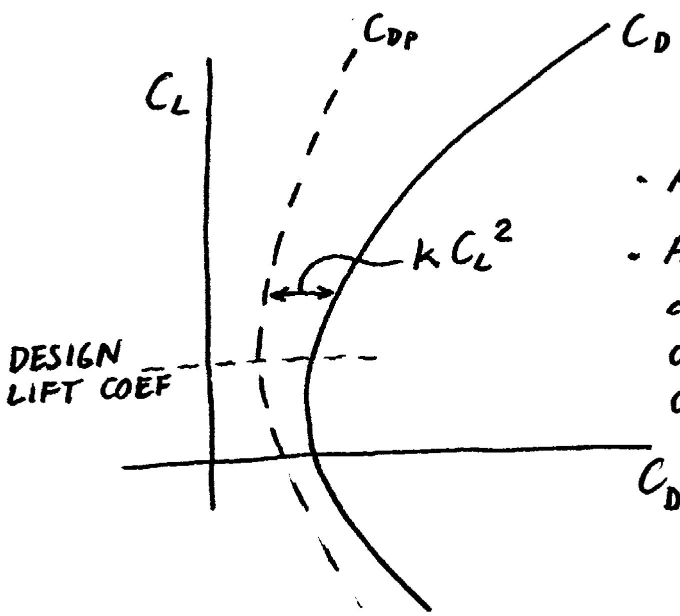
Drag is split into 2 or 3 components

- Pressure drag } Parasite drag
- Viscous drag }
- Induced drag

Drag is important in aircraft stability and control because:

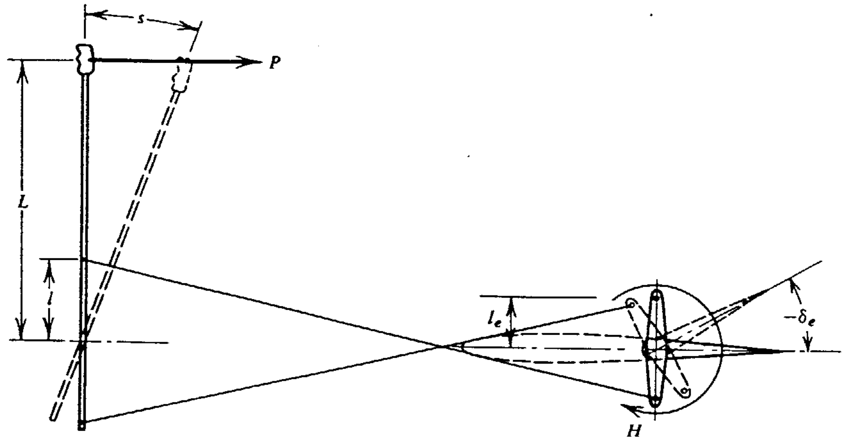
- Aircraft state requires force and momentum balance
- Speed stability depends on drag characteristics
- Control surfaces can produce drag
- Propulsion settings depend mainly on drag and couple to all axes.

Approximately $C_D = C_{D_0} + k\,C_L^2 \qquad k = \frac{e}{\pi AR}$



- Applies to wing
- Applies to entire aircraft at a given configuration and with control settings constant

Control Gearing and Hinge Moments



Schematic of a longitudinal control system.

$$P = G H \qquad \delta_e = -\frac{s}{L}\frac{l}{l_e}$$

$$PL = \left(\frac{H}{l_e}\right) l \qquad P = \left(-\frac{\delta_e}{s}\right) H \qquad G = -\frac{\delta_e}{s}$$

Note: Gearing ratio has units of $\frac{\text{deg}}{\text{inch}}$, typically

With simple gearing, the control force is due to:

- Aerodynamic hinge moment
- Linkage variations in gearing ratio
- Mass imbalance, imposed accelerations, and gravity
- Inertia, friction, aerodynamic damping

The pilot expects consistent control forces that only vary with:

- Control deflection
- Speed
- Load factor

Structural designers expect control deflections that do not produce excessive loading.

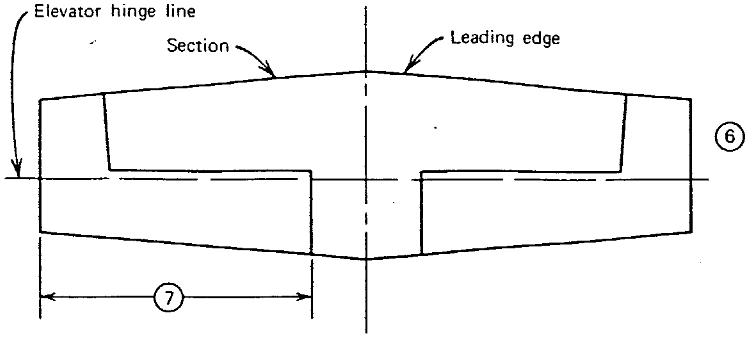
⇒ Provided by control deflection limits and pilot control force limitations

Aerodynamic hinge moments are important. Hinge moment is highly dependent on fine details of airfoil and control surface contour, gap seals or venting, balance panels and overhang, as well as special treatments affecting hinge moment.

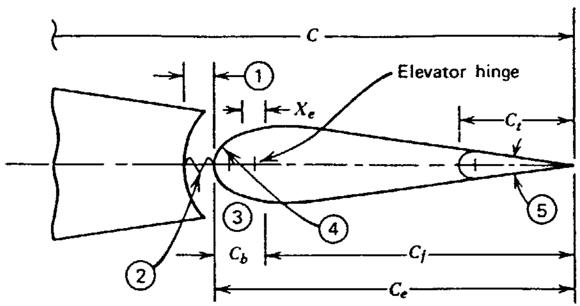
The following charts describe variations from a typical elevator of:

- 10% thick, symmetric airfoil
- $c_e/c = 0.3$
- No aerodynamic balance
- Round nose on elevator

  $b_1 = -0.55 \qquad b_2 = -0.89$



(a)

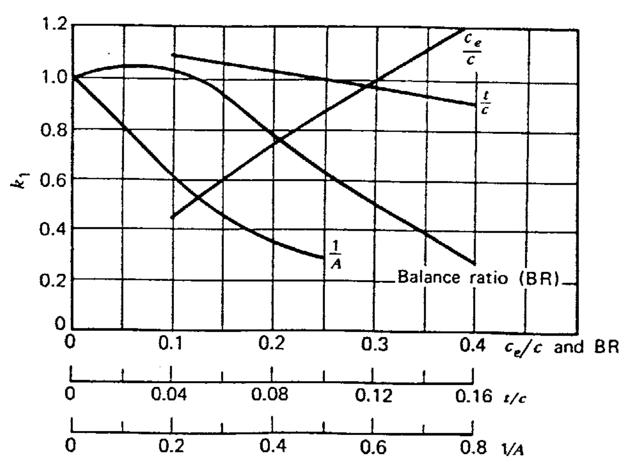


Factors affecting tail performance

1. Gap
2. Seal (or lack of)
3. Aerodynamic balance (ratio of x_e to c_f)
4. Nose shape
5. Trailing edge bevel
6. Control horn balance
7. Spanwise extent of elevator
8. Total tail aspect ratio and taper ratio
9. Ratio of c_e to c

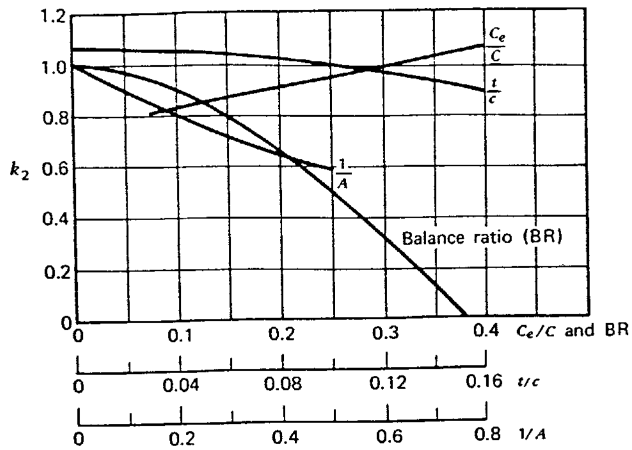
(b)

Horizontal tail geometry. (*a*) Planform view, (*b*) Section view.



$$b_1 = -0.55\ k_1(c_e/c),\ k_1(t/c)\ k_1(\mathrm{BR})\ k_1(1/A)$$

Figure 8.11 Factors determining the rate of change of elevator hinge moment with angle of attack.



$b_2 = 0.89\, k_2(c/_e c)\, k_2(\mathrm{BR})\, k_2(1/A)$

Figure 8.12 Factors for determining the rate of change of elevator hinge moment with elevator angle.

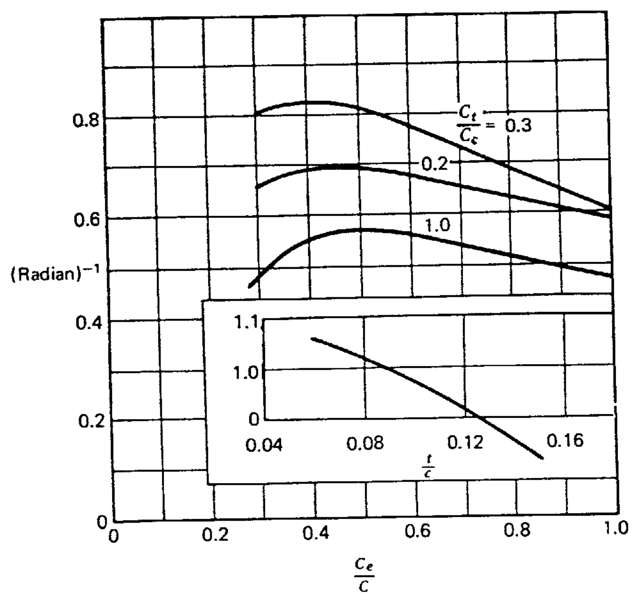


Figure 8.13 Rate of change of elevator hinge moment coefficient with trim tab angle.

Suppose $A=3.6$ $\frac{c_e}{c}=0.37$ $\frac{t}{c}=0.1$ $\frac{b_e}{b}=1.0$

$\frac{c_t}{c_e}=0.30$ $\frac{b_t}{b}=0.25$ $BR=0$

From Fig. 8.11 $k_1(\frac{c_e}{c})=1.16$ $k_1(\frac{t}{c})=1.0$ $k_1(BR)=1.0$ $k_1(\frac{1}{A})=0.49$

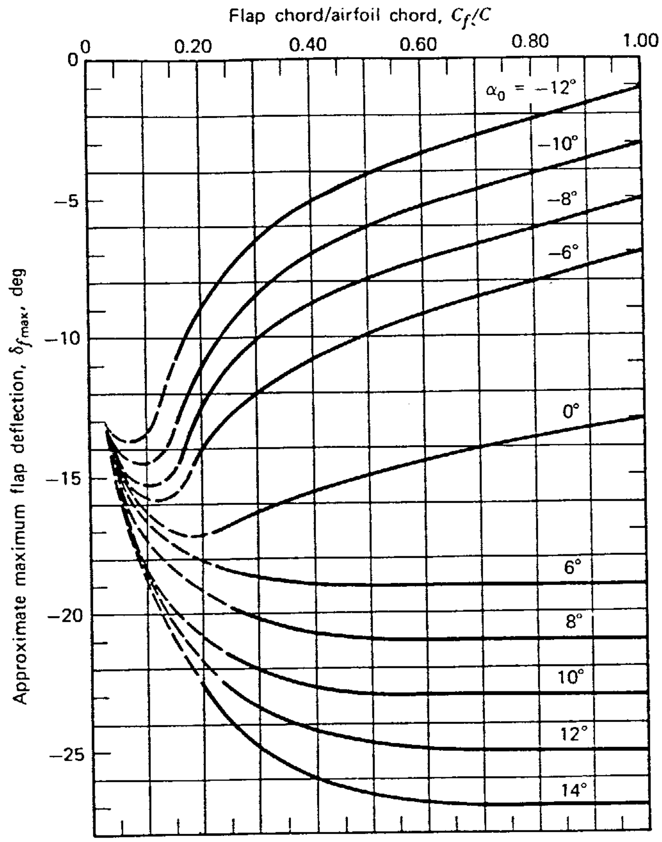
From Fig. 8.12 $k_2(\frac{c_e}{c})=1.05$ $k_2(\frac{t}{c})=1.0$ $k_2(BR)=1.0$ $k_2(\frac{1}{A})=0.73$

Then $b_1=-0.55(1.16)(0.49)=-0.31$ $b_2=-0.89(1.05)(0.73)=-0.68$

From Fig. 8.13 $\frac{b_3}{K}=-0.83$ $K=0.97 \Rightarrow b_3=-0.81$ (2-D)

for 25% span $b_3=-0.20 \Rightarrow C_H=-0.31\alpha-0.68\delta_e-0.20\delta_t$

$\alpha, \delta_e, \delta_t$ in radians



Approximate maximum allowable flap deflections for linear limits of airfoil characteristics at various angles of attack.

Tab linear range is ± 15° to ± 20°

Longitudinal Aerodynamic Derivatives

Steady State Derivatives

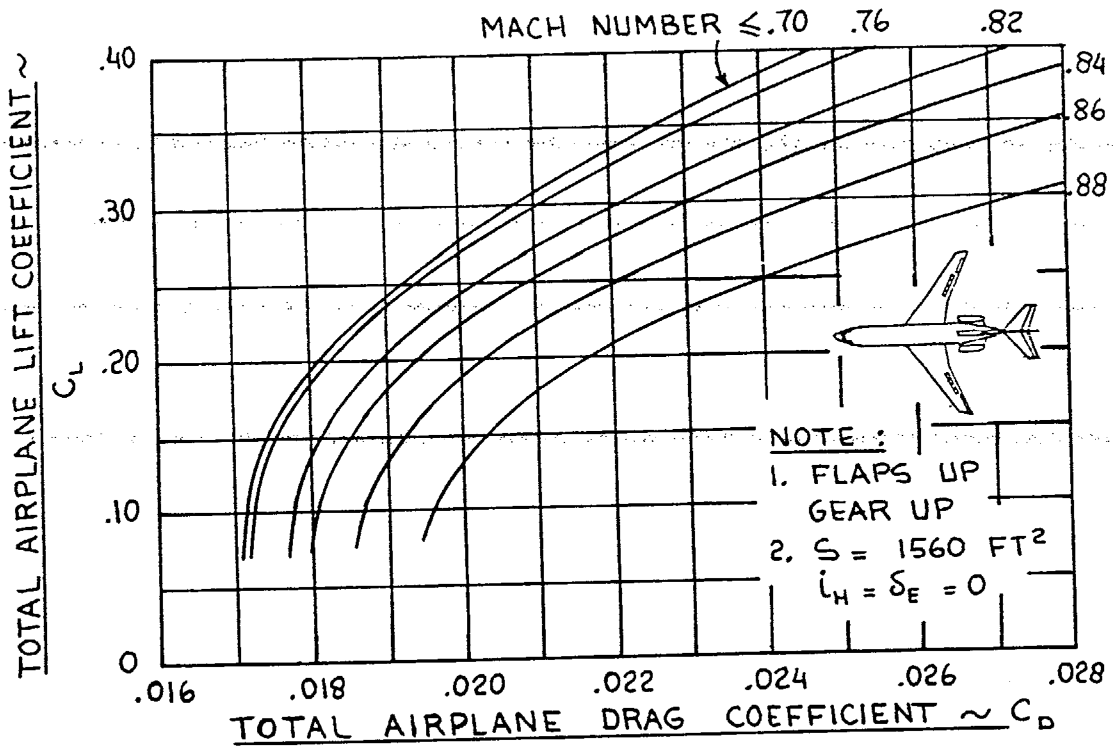
- Derivatives relate to Lift, Drag, and Pitch Moment
- Variables are:
  - α
  - δ_e, i_H
  - $\bar{q}$
  - Mach and Reynolds Number

Drag

$$D = C_D q S$$

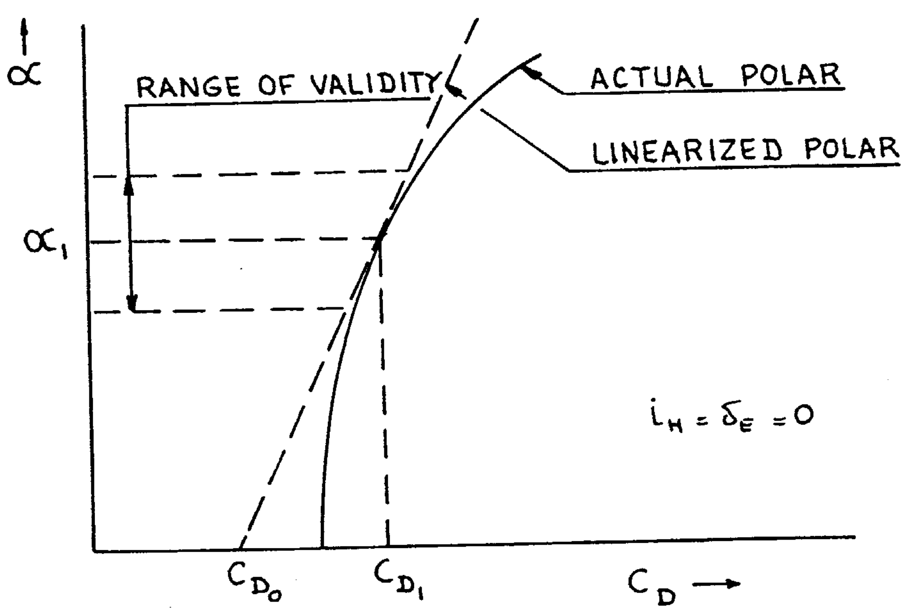
$$C_D = C_{D_0} + C_{D_\alpha} \alpha + C_{D_{i_H}} i_H + C_{D_{\delta_E}} \delta_E$$

- C_{D_0} and C_{D_α} for given α (and C_L) $i_H = 0$ $\delta_e = 0$ (C_{D_0} not C_D at $\alpha = 0$!)
- $C_{D_{i_H}}$ and $C_{D_{\delta_e}}$ at $\alpha = 0$
- Permutations based on real properties are common
- No explicit induced drag term! Linearized + Lousy

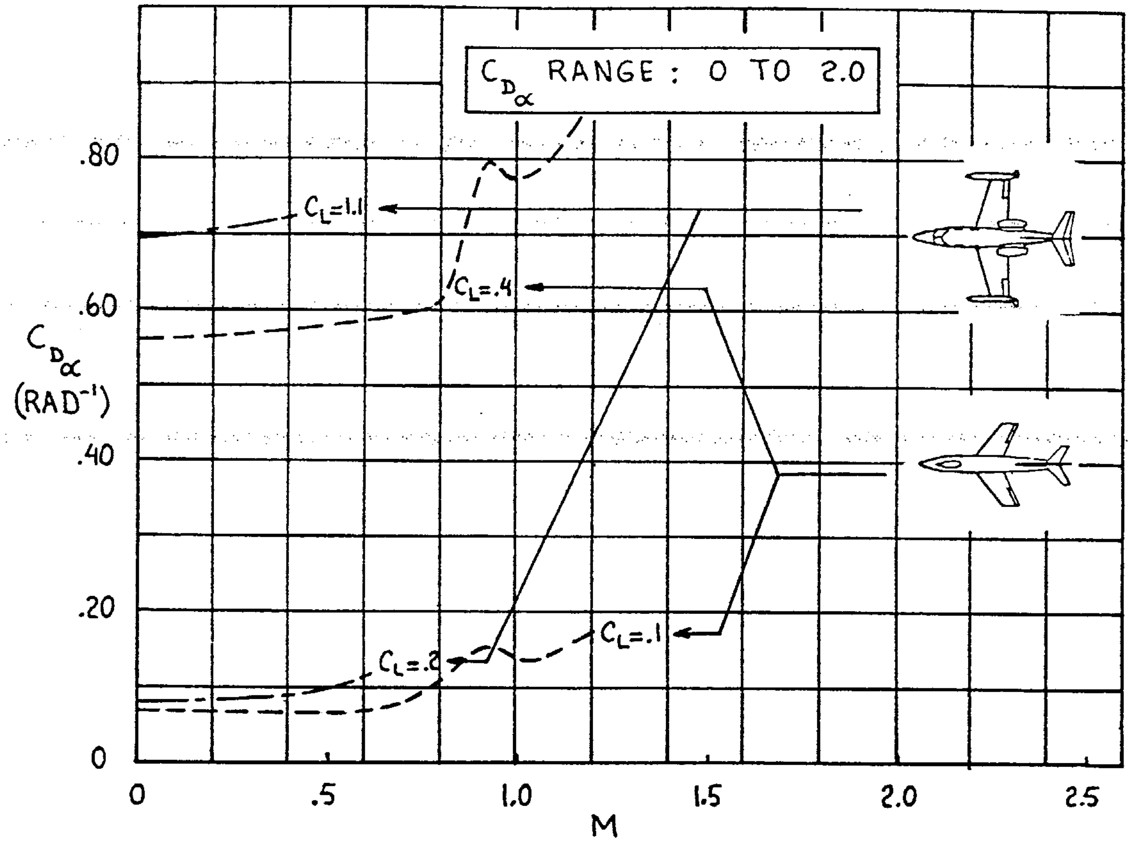


NOTE: 1) At Point A, which is a typical cruise point, L/D = 14.6.
2) $L/D|_{max}$ at M = .70 is ∿ 17.5!

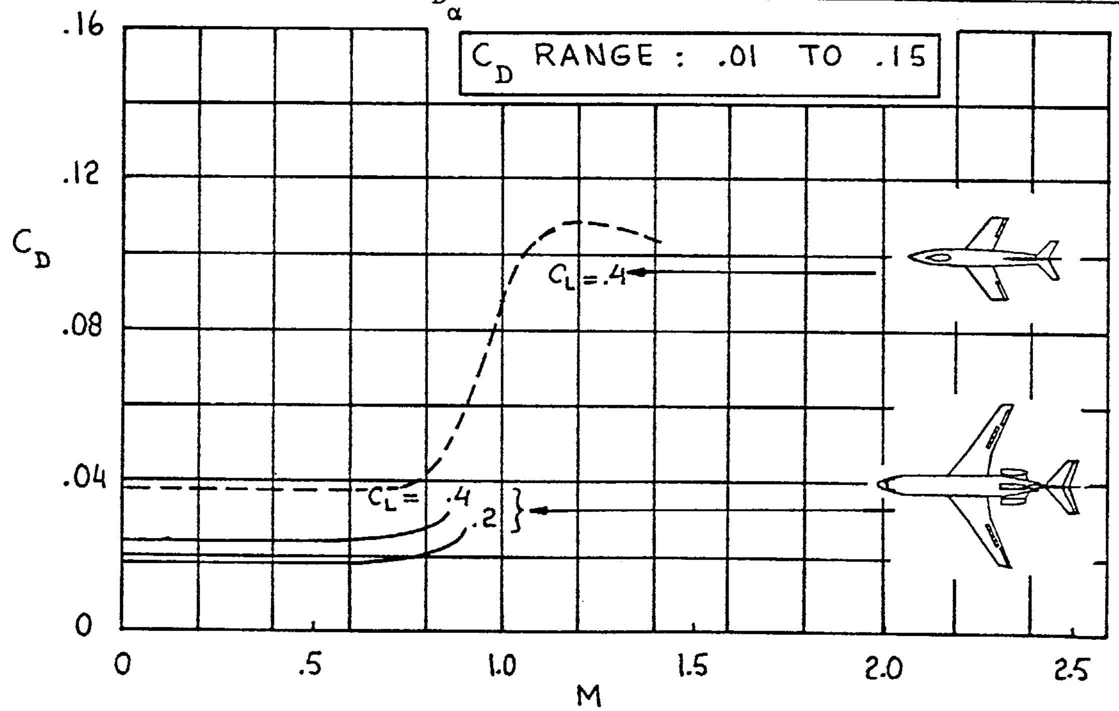
Example Drag Polars for a Subsonic Jet Transport



Validity of the Linear Drag Polar Representation



Variation of C_{D_α} with Mach Number for Typical Jet Aircraft



Variation of C_D with Mach Number for Typical Jet Aircraft

Lift

$$L = C_L q S$$

$$C_L = C_{L_0} + C_{L_\alpha}\alpha + C_{L_{i_H}} i_H + C_{L_{\delta_e}}\delta_e$$

- Basic contribution to lift is wing-body lift and stabilizer lift
- Wing-body lift is assumed independent, stabilizer requires massive coupling corrections (due to downwash and wake deficit)

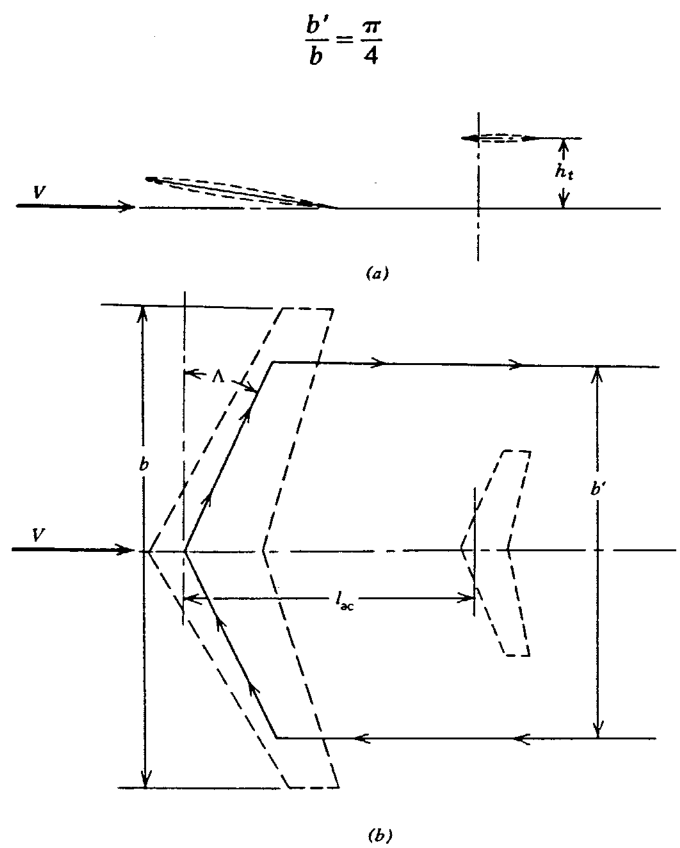
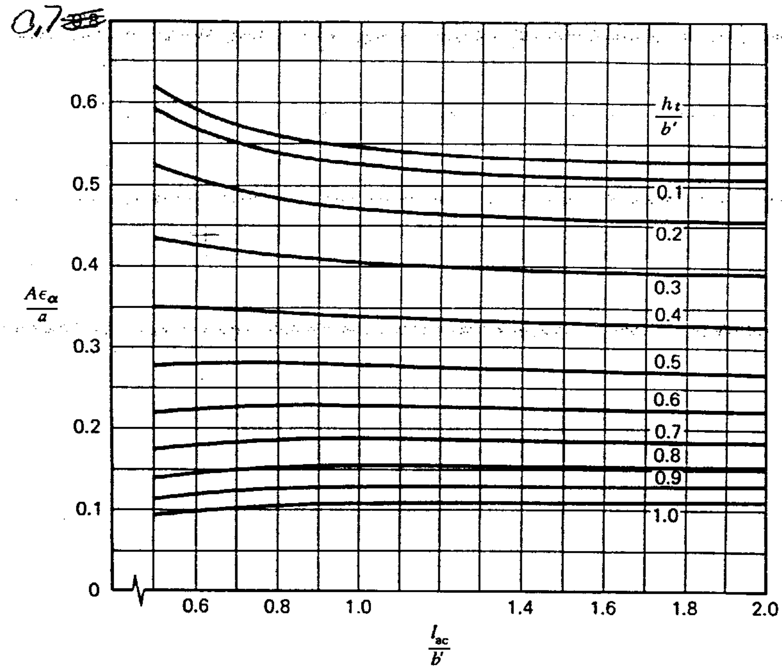
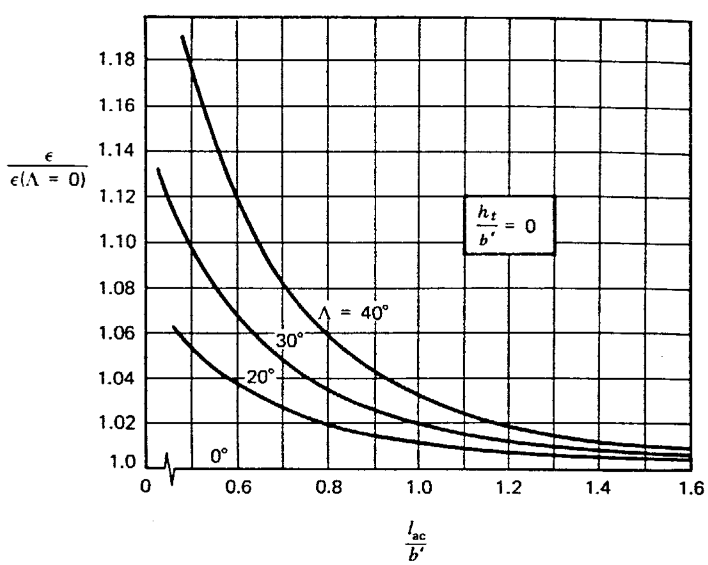


Figure 8.5 Equivalent vortex system for calculating downwash at tail. (*a*) Side view. (*b*) Planform view.



Downwash angle for $\Lambda = 0$.



Correction to downwash angle for sweepback.

```
c      Program for printing table of effective downwash, tindf.f
c
c Walt Lounsbery, 8-13-92
c
c This is translated from a program in HP 9845 BASIC from 10-31-83
c
      program tindf
c b   - wing span, b1 is active, b2 is acted on
c h   - height of passive wing over active wing, wind coord.
c l   - distance from active wing 1/4 mac to passive wing 1/4 mac
c       along free air wind vector
c nar - number of aspect ratios for active wing
c nbrat - number of span ratios (b2/b1)
c nhrat - number of height ratios (h/b1)
c ndrat - number of distance ratios (l/b1)
c art   - active wing aspect ratio breakpoints
c b2b1t - passive/active wing span ratio breakpoints
c hb1t  - height to active wing span ratio breakpoints
c lb1t  - downstream distance to active wing span ratio breakpoints
c lineval - temp storage for downwash values at height ratio brkpts
      parameter (nar=5,nbrat=5,nhrat=6,nlrat=6)
c      parameter (nar=1,nbrat=1,nhrat=1,nlrat=1)
      common /const/ degrad,radeg,pi
      real art(nar),b2b1t(nbrat),hb1t(nhrat),lb1t(nlrat),
     &   lineval(nlrat)
      data art/5.9,6.0,6.5,6.9,7./
      data b2b1t/0.3,0.35,0.4,0.45,0.5/
      data hb1t/0.,0.05,0.10,0.15,0.20,0.25/
      data lb1t/0.3,0.4,0.5,0.6,1.0,2.0/
c      data art/6./
c      data b2b1t/0.35/
c      data hb1t/0.15/
c      data lb1t/2.2/

      pi=4.*atan(1.)
      degrad=pi/180.
      radeg=1./degrad

      open(unit=8,file='dwash.out',status='new')

      do m1=1,nar
        do m2=1,nbrat
          write(8,100) art(m1),b2b1t(m2),(hb1t(i),i=1,nhrat)
  100     format(//t15,'Table of wing effective',
     &       ' downwash functions'//t10,
     &       'Active wing aspect ratio is ',f5.2/
     &       t10,'Passive/active wing span ratio is ',f7.2//
     &       4x,'DISTANCE',t25,'Height/active wing span ratio'/
     &       4x,'  RATIO',6(2x,f6.3,2x))
          do m4=1,nlrat
            do m3=1,nhrat
              b1=1.
              lineval(m3)=fepeff(b1,art(m1),lb1t(m4),
     &                          hb1t(m3),b2b1t(m2))
            end do
            write(8,110) lb1t(m4),(lineval(i),i=1,nhrat)
```

```
 110       format(4x,f6.2,2x,6(f8.3,2x))
         end do
       end do
      end do
      stop
      end

      function fepeff(bl,arl,l,ht,b2)
      real l,ht
      dimension w(20)
      common /const/ degrad,radeg,pi
      nl=15
      n2=15
      dum=sin(degrad*45./float(nl+1))
      yp=.995*b2/2.
      do i=1,nl
        k2=i-1
        w(i)=2.*cos(degrad*45.*(2.*float(k2)+1.)/float(nl+1))*dum
      end do
      call eps(bl,nl,l,ht,w,yp,esum)
      epsilon=0.
      e0=esum
      y0=yp
      do i=1,n2
        yp=b2*cos(degrad*90.*float(i)/float(n2))/2.
        call eps(bl,nl,l,ht,w,yp,esum)
        epsilon=epsilon+(esum*sqrt(b2*b2/4.-yp*yp)+
     &            e0*sqrt(b2*b2/4.-y0*y0))*(y0-yp)
c        write(8,*) '  yp=',yp,'  esum=',esum,'  epsilon=',epsilon
        y0=yp
        e0=esum
      end do
      fepeff=0.81057*bl*epsilon/(b2*b2*(2.+sqrt(arl*arl+4.)))
c      write(8,*) '  bl=',bl,'  epsilon=',epsilon
c      write(8,*) '  b2=',b2,'  arl=',arl
c      write(8,*) '  final fepeff=',fepeff
      end

      subroutine eps(bl,nl,l,ht,w,yp,esum)
      dimension w(20)
      real l,ht
      common /const/ degrad,radeg,pi
c      write(8,*) '  routine eps, bl=',bl,'  nl=',nl
      esum=0.
      do kl=1,nl
        ygam=bl*cos(degrad*90.*float(kl)/float(nl+1))/2.
        call eats(yp,ygam,l,ht,et)
        esum=esum+w(kl)*et
      end do
      return
      end

      subroutine eats(yp,ygam,l,ht,et)
      real l,ht
      t1=ygam-yp
      t2=yp+ygam
      t3=ht*ht+l*l
```

```
      e1=0.
      g1=t3+t1*t1
      g2=t3+t2*t2
      if (abs(t3) .gt. 0.0001) then
        if (abs(g1) .gt. 0.01) e1=t1/sqrt(g1)
        if (abs(g2) .gt. 0.01) e1=e1+t2/sqrt(g2)
        e1=1*e1/t3
      end if
      e2=0.
      g3=ht*ht+t1*t1
      if (abs(g3) .gt. 0.0001) then
        if (abs(g1) .gt. 0.001) e2=1/sqrt(g1)
        e2=t1*(e2+1.)/g3
      end if
      e3=0.
      g4=ht*ht+t2*t2
      if (abs(g4) .gt. 0.0001) then
        if (abs(g2) .gt. 0.001) e3=1/sqrt(g2)
        e3=t2*(e3+1.)/g4
      end if
      et=e1+e2+e3
      return
      end
```

$$C_{L_H} = \underbrace{C_{L_{0H}}}_{= 0 \text{ for symmetric, flat stabilizer}} + C_{L_{\alpha_H}}(\alpha + i_H - \epsilon + \tau_E \delta_E)$$

$$\alpha_H = \alpha + i_H - \epsilon$$

$$\epsilon = \epsilon_0 + \frac{d\epsilon}{d\alpha}\alpha$$

$$\text{if } \eta_H = \frac{\bar{q}_H}{\bar{q}}$$

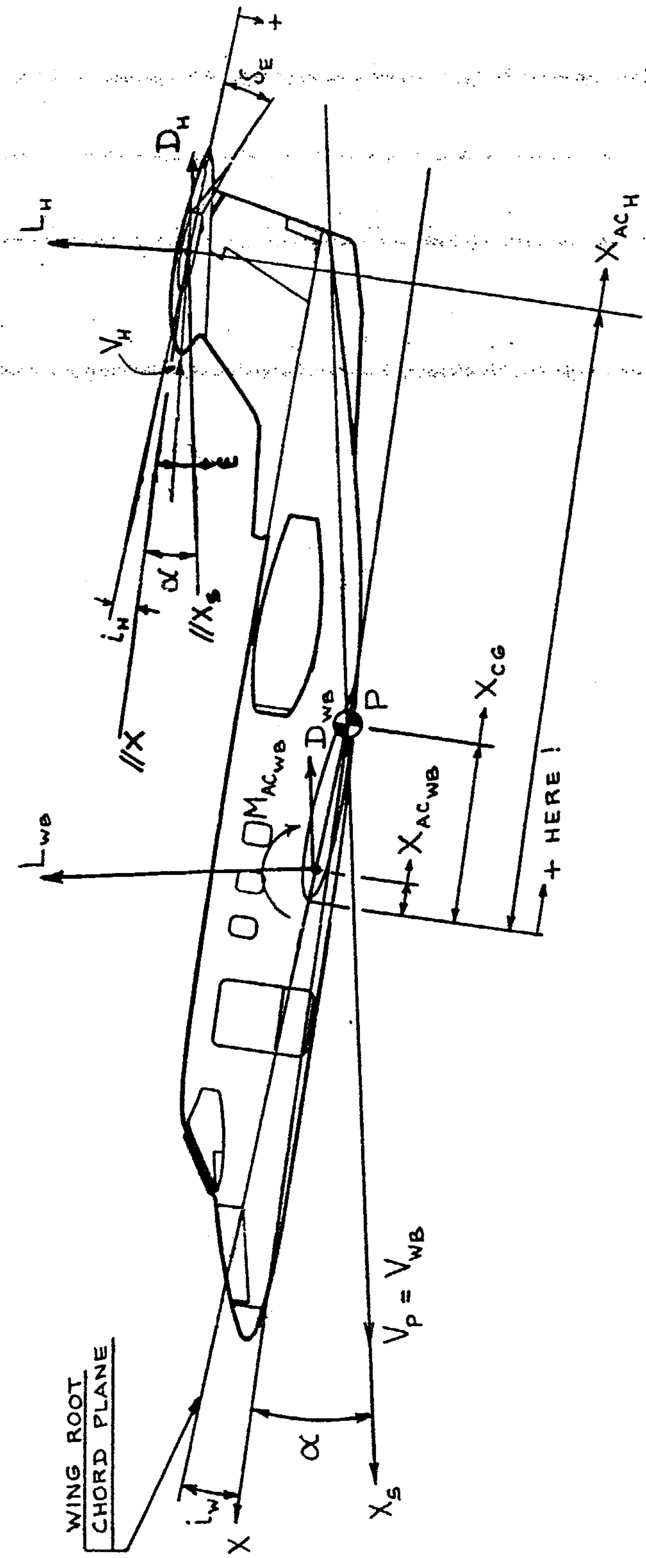
$$C_L = C_{L_{0WB}} + C_{L_{\alpha_{WB}}}\alpha + C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}\left\{\alpha - \left(\epsilon_0 + \frac{d\epsilon}{d\alpha}\alpha\right) + i_H + \tau_E \delta_E\right\}$$

so $C_{L_0} = C_{L_{0WB}} - C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}\epsilon_0$

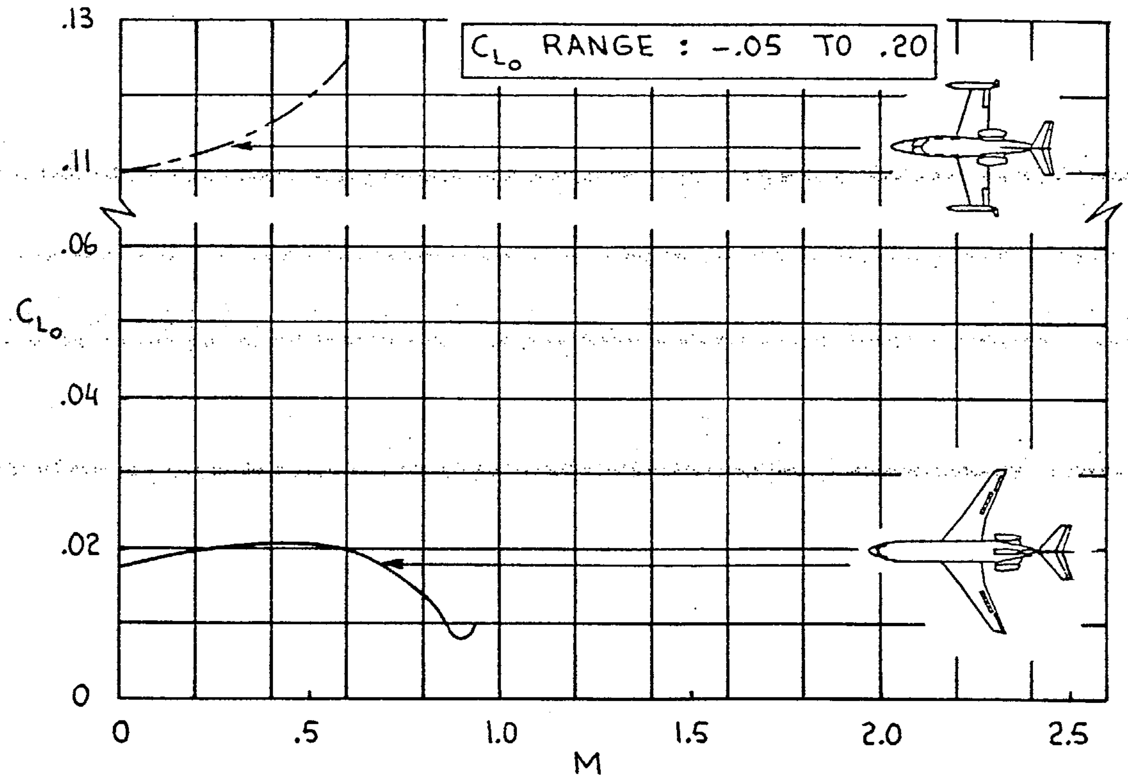
$$C_{L_\alpha} = C_{L_{\alpha_{WB}}} + C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}\left(1 - \frac{d\epsilon}{d\alpha}\right)$$

$$C_{L_{i_H}} = C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}$$

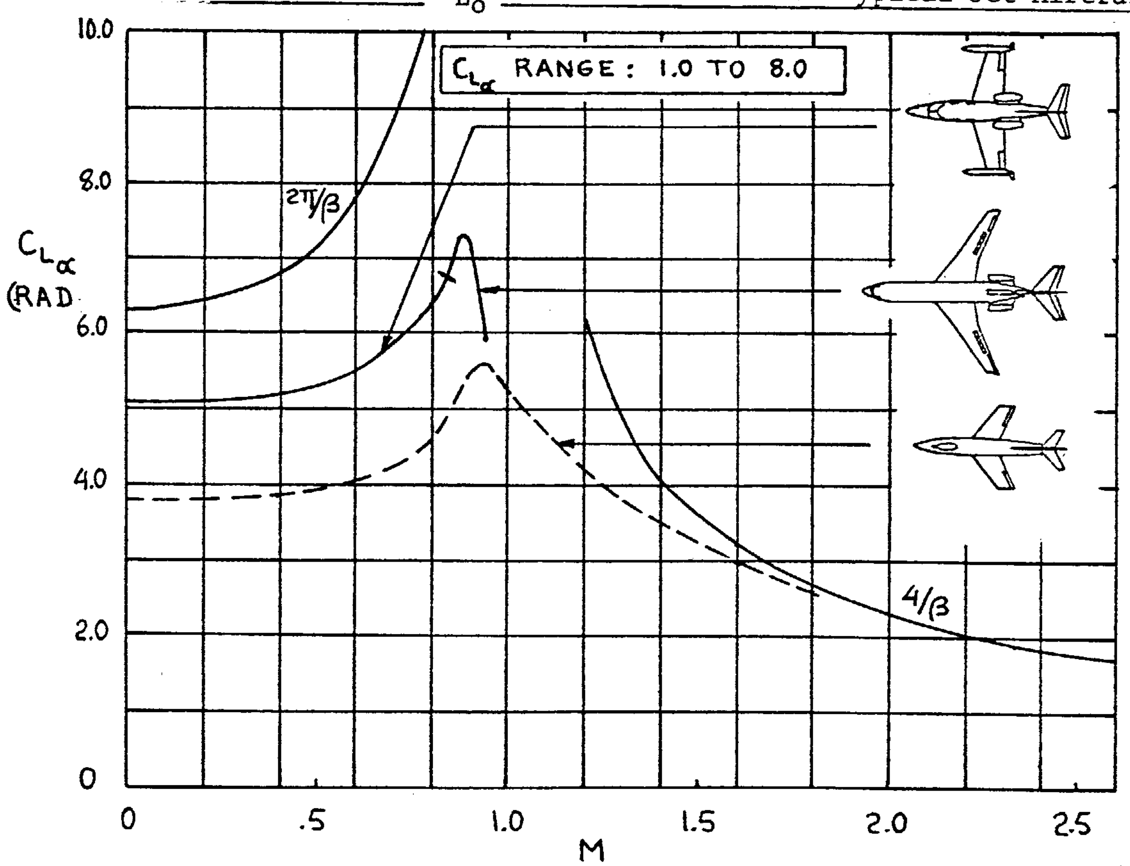
$$C_{L_{\delta_E}} = C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}\tau_E$$



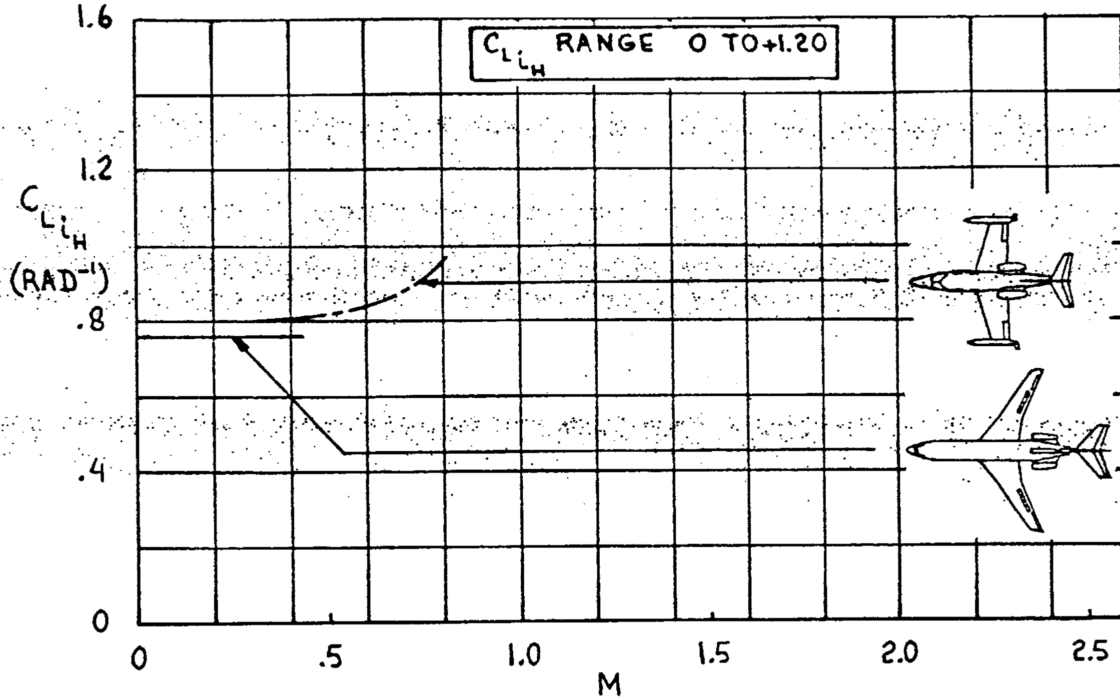
Geometry for Finding Total Airplane Aerodynamic Parameters (Power-off)



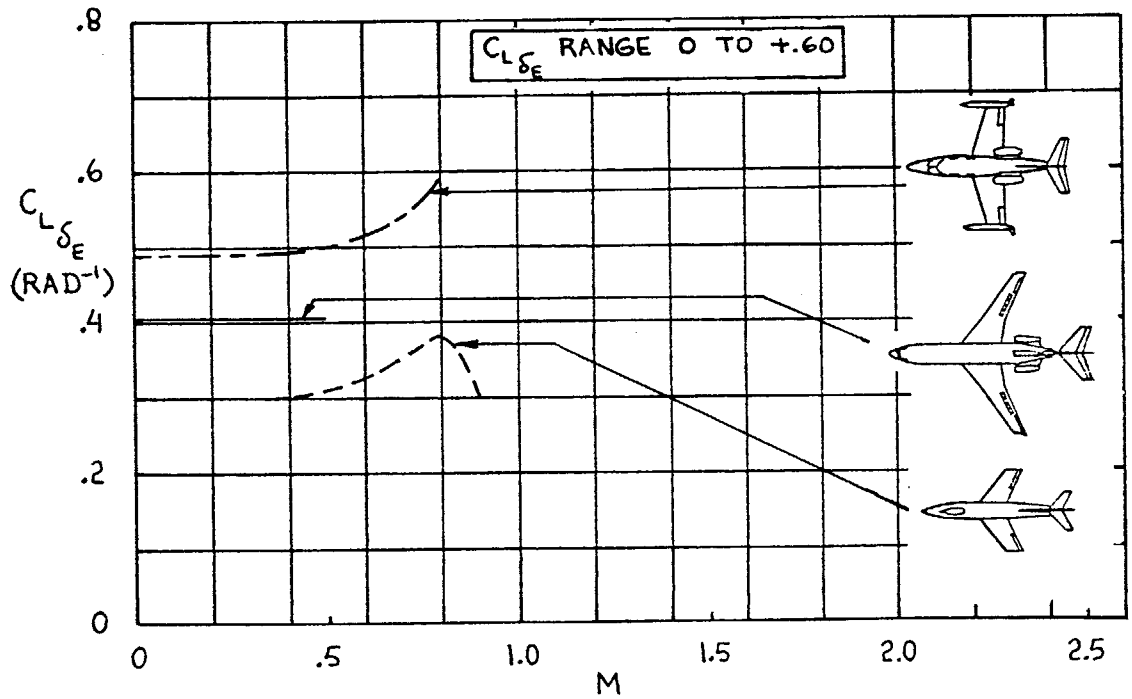
Variation of C_{L_0} with Mach Number for Typical Jet Aircraft



Variation of C_{L_α} with Mach Number for Typical Jet Aircraft



Variation of $C_{L_{i_H}}$ with Mach Number for Typical Jet Aircraft



Variation of $C_{L_{\delta_E}}$ with Mach Number for Typical Jet Aircraft

Session IV
Feb. 9, 1993

Longitudinal Aerodynamic Derivatives

Steady State Derivatives Continued

Pitch Moment

$$M = C_M \bar{q} S \bar{c}$$

$$C_M = C_{M_0} + C_{M_\alpha} \alpha + C_{M_{i_H}} i_H + C_{M_{\delta_e}} \delta_e$$

$$M = C_{M_{ac_{WB}}} \bar{q} S \bar{c} + L_{WB} (x_{CG} - x_{ac_{WB}}) \cos\alpha$$

$$+ D_{WB} (x_{CG} - x_{ac_{WB}}) \sin\alpha - L_H (x_{ac_H} - x_{CG}) \cos(\alpha - \epsilon)$$

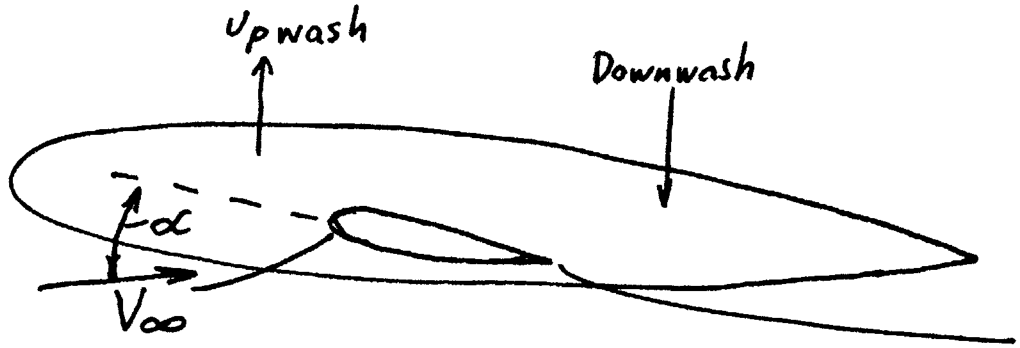
- neglecting tail drag contribution
- using small angle approximation
  $\cos\alpha \doteq 1 \qquad \cos(\alpha - \epsilon) = 1$
- neglecting wing-body drag term
- Normalizing

$$C_M = C_{L_{WB}} \frac{x_{CG} - x_{ac_{WB}}}{\bar{c}} + C_{M_{ac_{WB}}} - C_{L_H} \frac{\bar{q}_H S_H}{\bar{q} S} \frac{x_{ac_H} - x_{CG}}{\bar{c}}$$

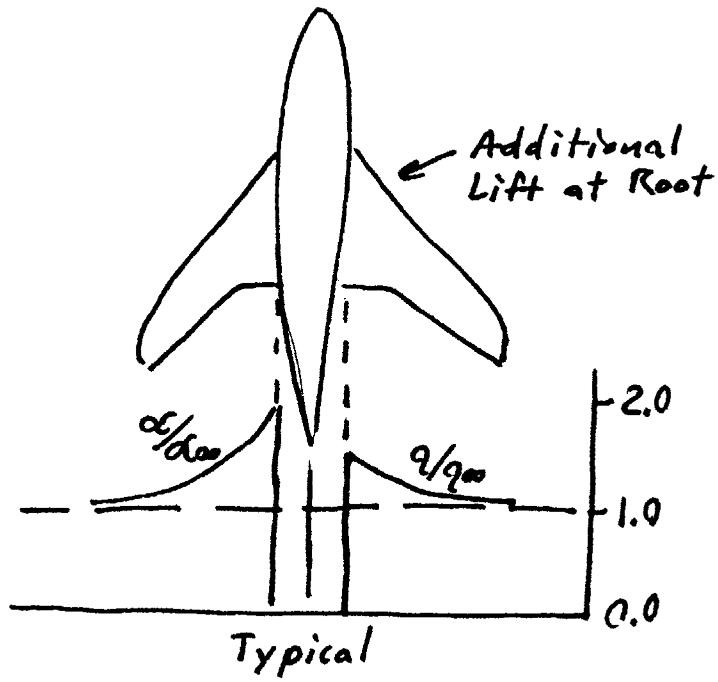
$$C_M = (C_{L_{0_{WB}}} + C_{L_{\alpha_{WB}}} \alpha)(\bar{X}_{CG} - \bar{X}_{ac_{WB}}) + C_{M_{ac_{WB}}}$$

$$- C_{L_{\alpha_H}} \eta_H \frac{S_H}{S} (\bar{X}_{ac_H} - \bar{X}_{CG}) \left\{ \alpha - (\epsilon_0 + \frac{d\epsilon}{d\alpha} \alpha) + i_H + \tau_e \delta_e \right\}$$

- $\bar{X}_{ac_{WB}}$ is influenced by fuselage and any non-wing, non-horizontal tail contribution due to wing induced flow or accelerated flow contributions



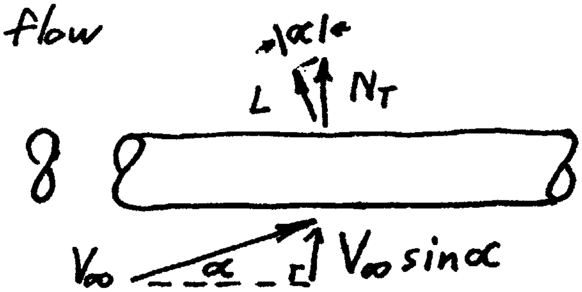
Wing Induced Flow on Fuselage



Fuselage Induced Flow on Wing

- Fuselage-wing interaction produces small corrections to lift ($C_{L\alpha_{WB}} \doteq 1.05\, C_{L\alpha_W}$) but large possible corrections to $C_{m_{0_{WB}}}$ and $C_{m_{\alpha_{WB}}}$

- Fuselage, nacelles, or similar items are assessed by similarity to cylinders in crossflow



$$N_T = q\, A_{REF}\, C_D \qquad \text{Typical } C_D = 0.4 \rightarrow 0.9$$

$$q = \frac{1}{2}\rho\,(V_\infty \sin\alpha)^2 \qquad \text{Based on 'plan' area}$$

$$L = N_T \sin\alpha \qquad C_L = \frac{L}{q_\infty A_{REF}} = C_D \sin^3\alpha$$

- Not a big influence! $\sin^3 10° = 0.00521$

Back to components of moment equation...

$$C_{M_0} = C_{M_{ac_{WB}}} + C_{L_{0_{WB}}}(-\bar{x}_{ac_{WB}} + \bar{x}_{cg}) + C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}(\bar{x}_{ac_H} - \bar{x}_{CG})\varepsilon_0$$

$$C_{M_{i_H}} = -C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}(\bar{x}_{ac_H} - \bar{x}_{CG})$$

Let $\bar{V}_H = \frac{S_H}{S}(\bar{x}_{ac_H} - \bar{x}_{CG})$, then $C_{M_{i_H}} = -C_{L_{\alpha_H}}\eta_H \bar{V}_H$

$$C_{M_{\delta_E}} = -C_{L_{\alpha_H}}\eta_H \bar{V}_H \tau_E$$

$$C_{M_\alpha} = C_{L_{\alpha_{WB}}}(\bar{x}_{cg} - \bar{x}_{ac_{WB}}) - C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}(\bar{x}_{ac_H} - \bar{x}_{cg})\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

C_{M_α} is complex, so...

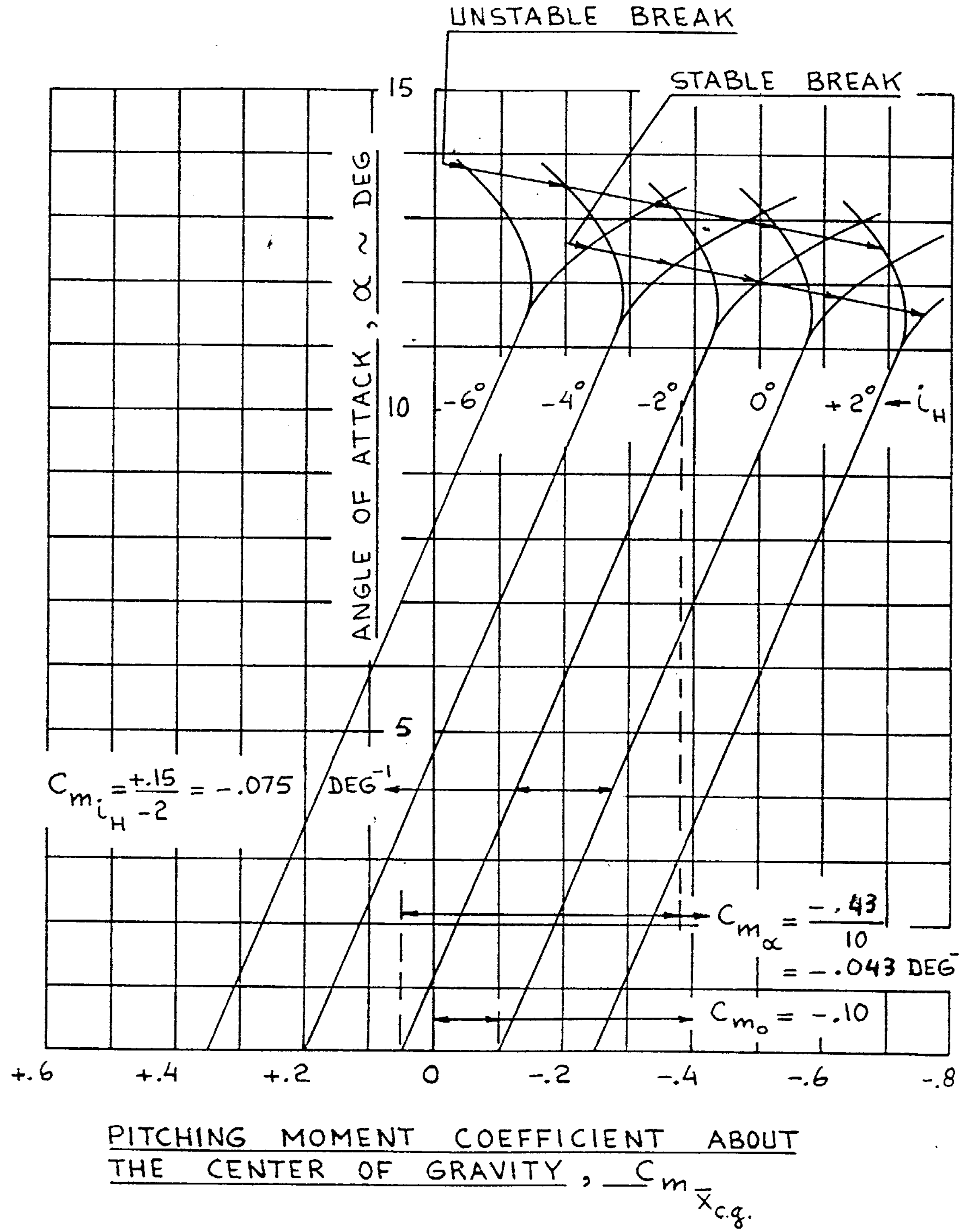
define $\bar{x}_{ac} = x_{cg}\Big)_{C_{M_\alpha}=0}$ "neutral point"

$$\bar{x}_{ac} = \frac{\bar{x}_{ac_{WB}} + \frac{C_{L\alpha_H}}{C_{L\alpha_{WB}}}\eta_H \frac{S_H}{S}\bar{x}_{ac_H}\left(1-\frac{d\epsilon}{d\alpha}\right)}{1 + \frac{C_{L\alpha_H}}{C_{L\alpha_{WB}}}\eta_H \frac{S_H}{S}\left(1-\frac{d\epsilon}{d\alpha}\right)}$$

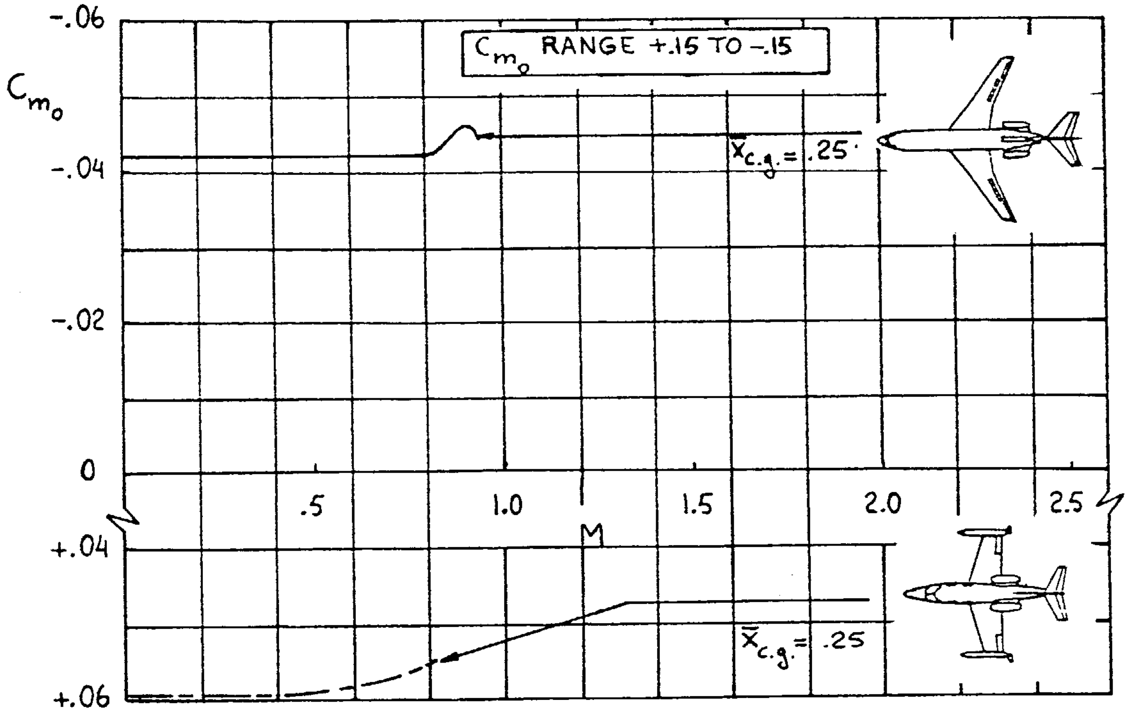
$$C_{M_\alpha} = C_{L_\alpha}\left(\bar{x}_{cg} - \bar{x}_{ac}\right)$$

"static margin" $\bar{x}_{cg} - \bar{x}_{ac} = \dfrac{C_{M_\alpha}}{C_{L_\alpha}} = \dfrac{d\,C_M}{d\,C_L}$

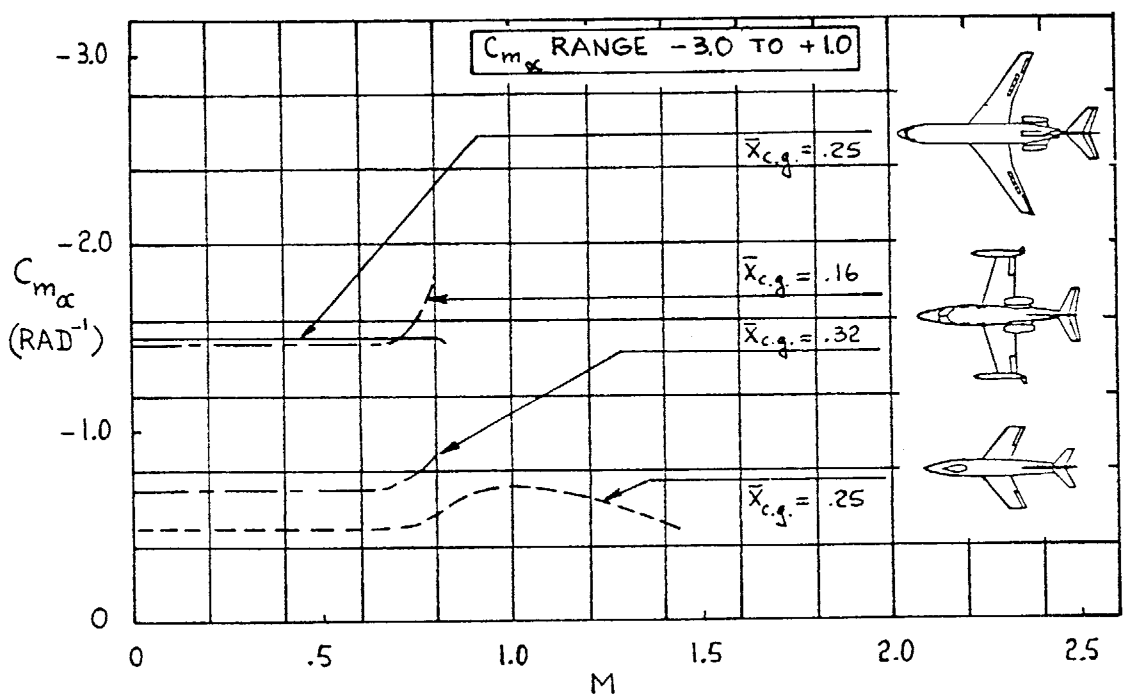
- Slope of pitching moment versus lift coefficient is related to stability and location of aircraft aerodynamic center.



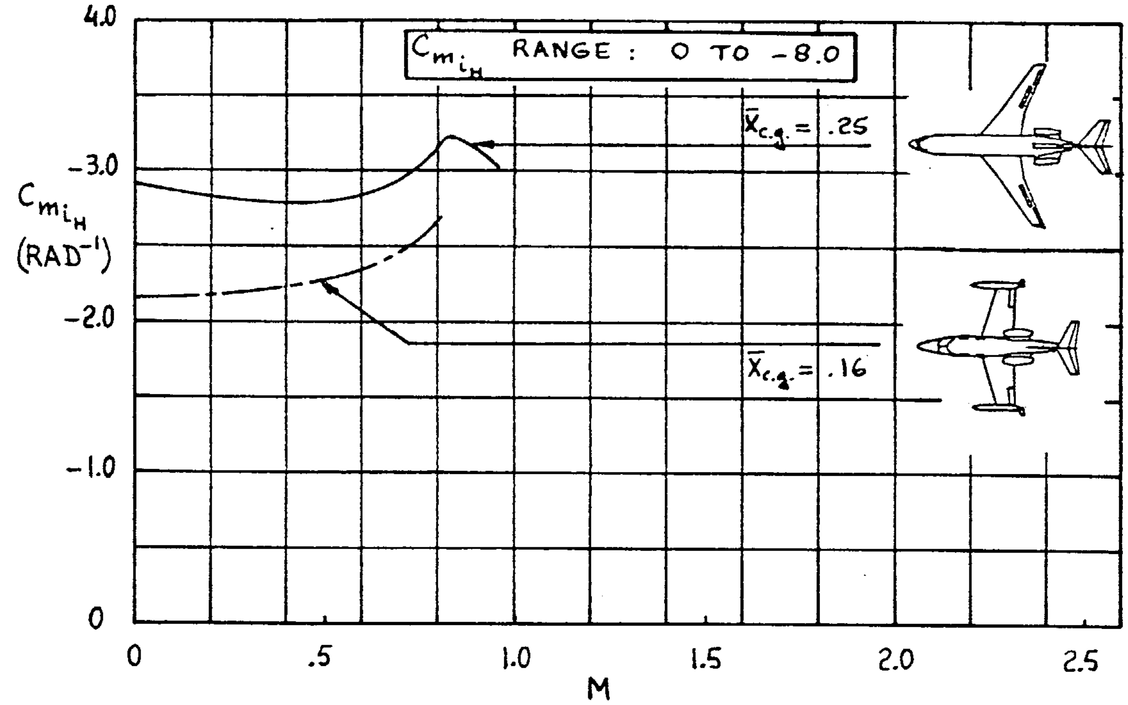
Typical Example of the Pitching Moment Slope of an Airplane



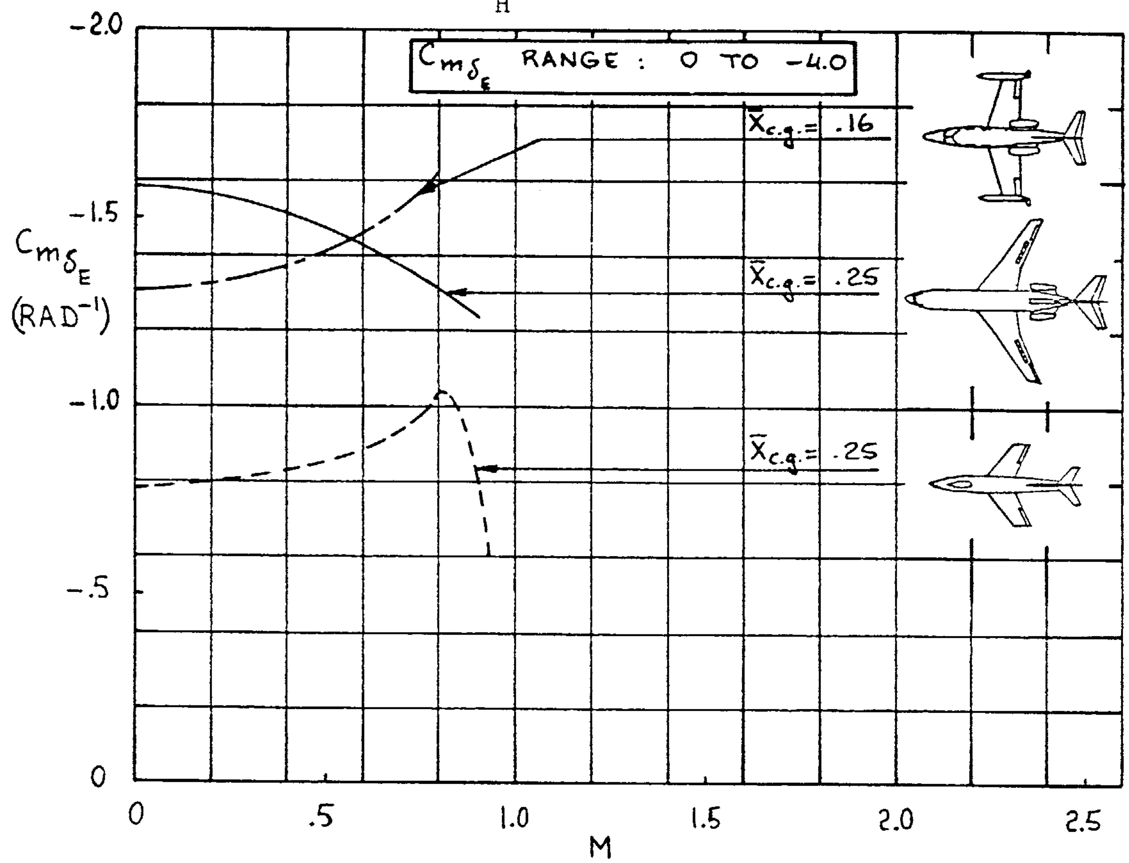
Variation of C_{m_o} with Mach Number for Typical Jet Aircraft



Variation of C_{m_α} with Mach Number for Typical Jet Aircraft

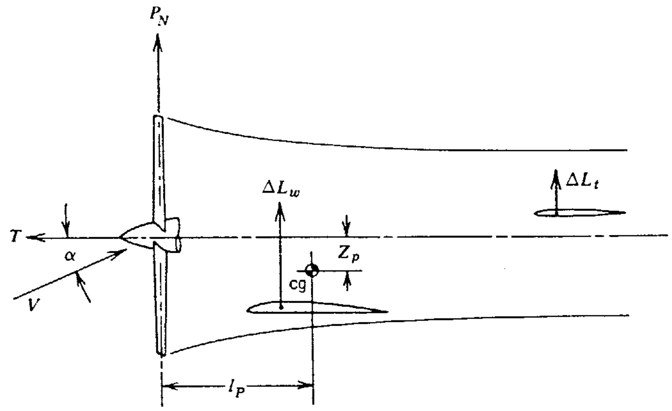


Variation of $C_{m_{i_H}}$ with Mach Number for Typical Jet Aircraft

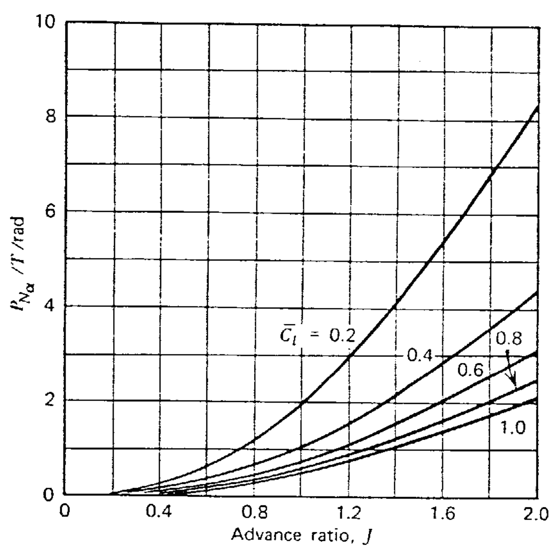


Variation of $C_{m_{\delta_E}}$ with Mach Number for Typical Jet Aircraft

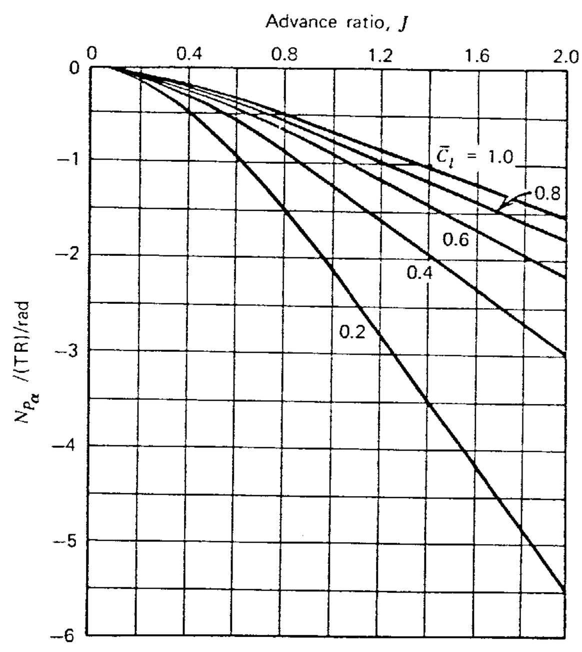
Propulsion Installation Effects



Effect of propeller forces and slipstream on longitudinal static stability.



Ratio of propeller normal force derivative to thrust.



Ratio of propeller moment derivative to product of thrust and radius.

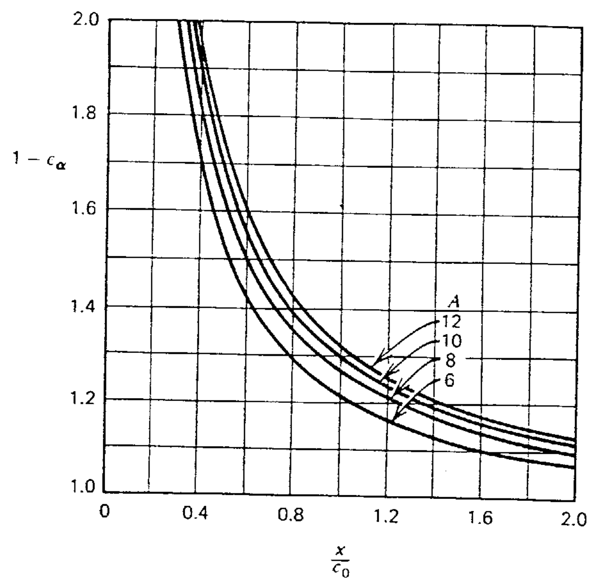


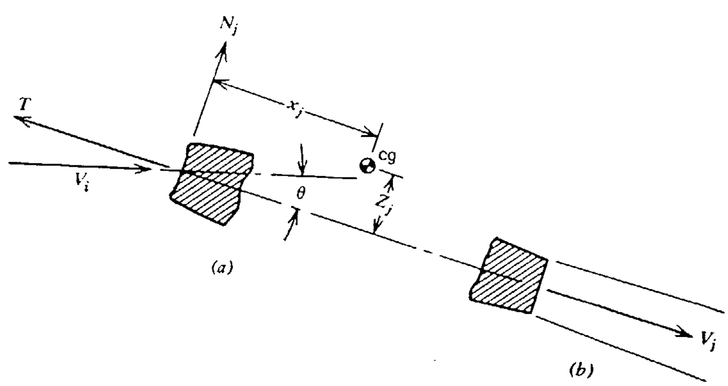
Figure 8.22c Correction for upwash ahead of a wing. Model of figure 8.5 assumed using an unswept elliptic wing. c_0 = root chord, x = distance ahead of quarter-chord line.

- Propwash effects are most important at low speeds and high power settings

For jet engines...

$$\Delta C_M = \left(\frac{T}{qS}\right)\frac{z_j}{\bar{c}}$$

$$N_j = m_j V_i \theta \qquad \theta \text{ in radians} \qquad \theta = \alpha - \epsilon_i$$



The normal force produced by a jet engine. (a) Intake. (b) Tail pipe.

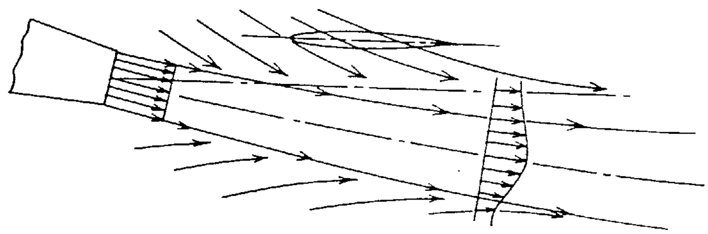
$$\Delta C_{M_\alpha} = \left(\frac{m_j V_i}{q S}\right) \frac{x_j}{\bar{c}} (1 - \epsilon_{i\alpha})$$

since $m_j = \rho_i A_i V_i$ (roughly)

and $m_j = f(T)$

then $\Delta C_{M_\alpha} = f(T, V_i, \epsilon_{i\alpha})$

Jet efflux produces induced angles of attack by entrainment or direct blowing.



Flow inclination resulting from flow entrainment.

Again, jet effects can be most challenging at low speeds where large angles of attack and power settings produce the largest changes from power off trim.

Longitudinal Static Stability

- Define static stability as the tendancy to return to an equilibrium state from small disturbances.
- Exact behavior is not derived

Level Flight - No Controls

$$L = W = ((C_{L_\alpha}\alpha + C_{L_0})\bar{q}S + \text{Thrust terms}$$

$$M = 0 = ((C_{M_0} + C_{M_\alpha}\alpha)\bar{q}S\bar{c} + \text{Thrust terms}$$

$$T - D = 0 = T - (C_{D_0} + C_{D_\alpha}\alpha)\bar{q}S$$

$$\text{or} \quad 0 = T - ((C_{D_0} + KC_L^2)\bar{q}S$$

Neglect thrust terms for now

For trimmed flight $\frac{W}{\bar{q}S} = C_{L_0} + C_{L_\alpha}\alpha \qquad \alpha = \frac{\frac{W}{\bar{q}S} - C_{L_0}}{C_{L_\alpha}}$

if $C_{L_0} = 0$ (α referenced to aircraft zero lift α)

then $\alpha = \frac{\frac{W}{\bar{q}S}}{C_{L_\alpha}}$ or $\bar{q}S = \frac{W}{C_{L_\alpha}\alpha}$

$$0 = ((C_{M_0} + C_{M_\alpha}\alpha)\bar{q}S\bar{c} = ((C_{M_0} + C_{M_\alpha}\alpha)\frac{W\bar{c}}{C_{L_\alpha}\alpha}$$

$$\cancel{0 = \frac{C_{M_0}}{C_{L_\alpha}}} \quad 0 = \frac{C_{M_0}}{\alpha} + C_{M_\alpha} \qquad \alpha = -\frac{C_{M_0}}{C_{M_\alpha}} = \frac{\frac{W}{\bar{q}S}}{C_{L_\alpha}}$$

- Changing trim α or speed means altering aircraft C_{M_0}
- For positive stability (α increases, C_M decreases)

  $C_{M_\alpha} < 0$

$\Rightarrow$ for positive α, $C_{M_0} > 0$

A full treatment of static stability would concern variation about trim state of:

- α
- γ
- V

Let's look at speed stability, or $\frac{dF_{Ax}}{dV} < 0$

(ie more thrust, more speed)

$$F_{Ax} = T - D = T - (C_{D_0} + kC_L^2)\bar{q}S$$

Level flight, $C_{L_0} = 0$ $\quad C_L = \frac{W}{\bar{q}S}$

$$F_{Ax} = T - (C_{D_0} + k\frac{W^2}{\bar{q}^2S^2})\bar{q}S = T - C_{D_0}\bar{q}S - \frac{kW^2}{\bar{q}S}$$

$\bar{q} = \frac{\rho}{2}V^2$, Let $\frac{dT}{dV} \doteq 0$

$$\frac{dF_{Ax}}{dV} = -C_{D_0}\rho SV + \frac{4kW^2}{S\rho}V^{-3}$$

Example: let $\rho = 0.00237$ $\quad C_{D_0} = 0.12$ $\quad k = \frac{1}{\pi AR}$ $\quad AR = 4$ $\quad k = 0.0796$

$W/S = 10$ $\quad W = 500$ $\quad S = 50$ $\quad C_{L_{MAX}} = 2 \Rightarrow V_{MIN} = 65$ ft/s

V	$-C_{D_0}\rho SV$	$\frac{4kW^2}{S\rho V^3}$	$\frac{dF_{Ax}}{dV}$	
65	-0.9243	2.445	1.521	
75	-1.0665	1.592	0.526	↑Unstable
80	-1.1376	1.312	0.174	
85	-1.2087	1.094	-0.115	
100	-1.422	0.672	-0.75	↓Stable
150	-2.133	0.199	-1.934	

This shows the back side of the power curve!

Neutral Point

Adding elevator terms to the preceding equations

$$C_{M\delta_e}\delta_e \quad \text{and} \quad C_{L\delta_e}\delta_e$$

and solving for $\dfrac{\partial \delta_e}{\partial C_L} = \dfrac{-C_{M\alpha}}{C_{L\alpha}C_{M\delta_e} - C_{M\alpha}C_{L\delta_e}}$

if $C_{M\alpha} = 0$ then $\dfrac{\partial \delta_e}{\partial C_L} = 0$

The stick fixed neutral point is the same as the aircraft neutral point, and is that cg position where lift coefficient is not altered by elevator.

Using $C_L = \dfrac{W}{\bar{q}S}$ $\quad \dfrac{\partial C_L}{\partial V} = -\dfrac{4W}{\rho S V^3}$

then $\dfrac{\partial \delta_e}{\partial V} = \dfrac{4W}{\rho S V^3}\dfrac{C_{M\alpha}}{C_{L\alpha}C_{M\delta_e} - C_{M\alpha}C_{L\delta_e}}$

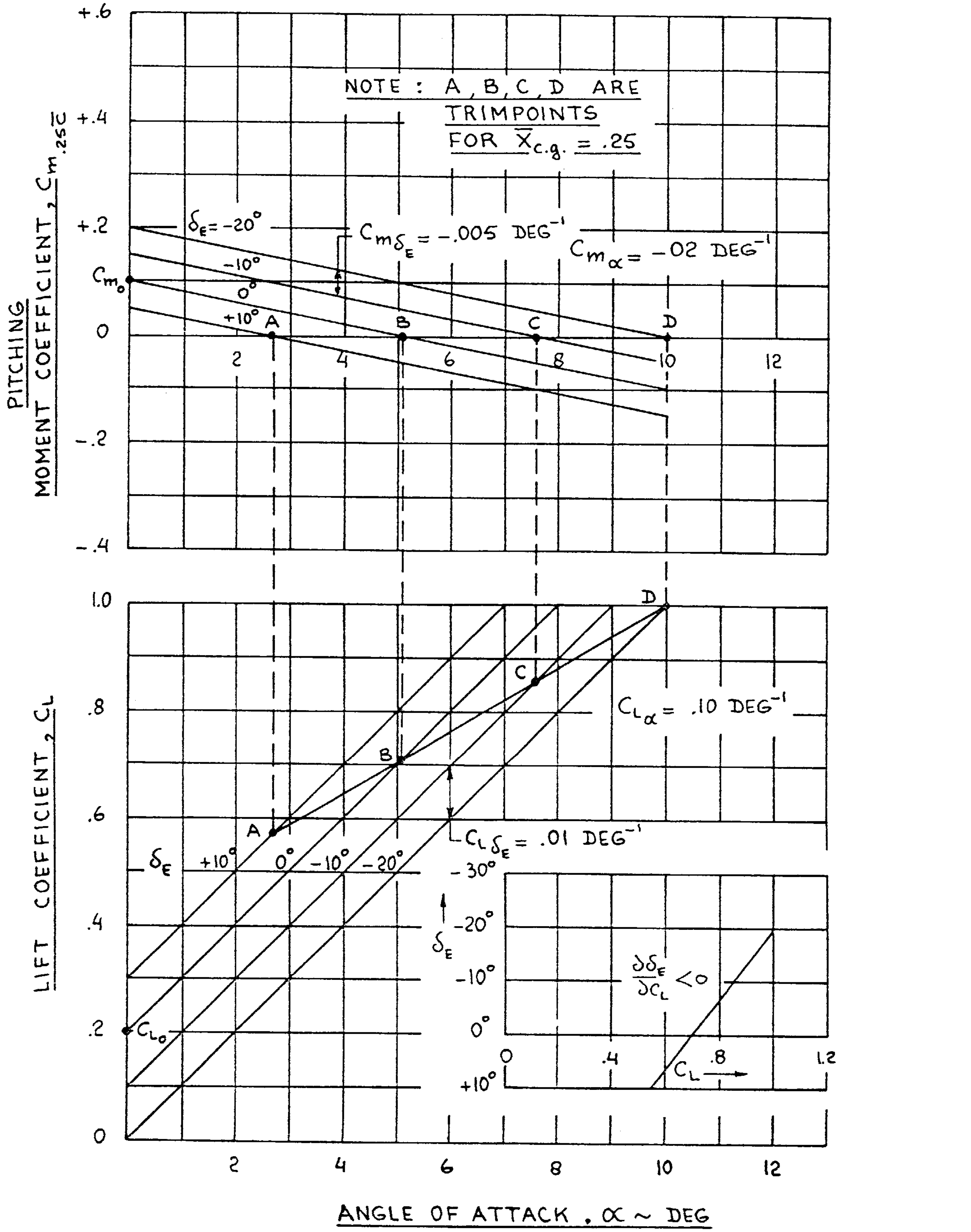
Usually $|C_{L\delta_e}C_{M\alpha}| << |C_{L\alpha}C_{M\delta_e}|$

So $\quad \dfrac{\partial \delta_e}{\partial V} > 0$ if $C_{M\alpha} < 0$

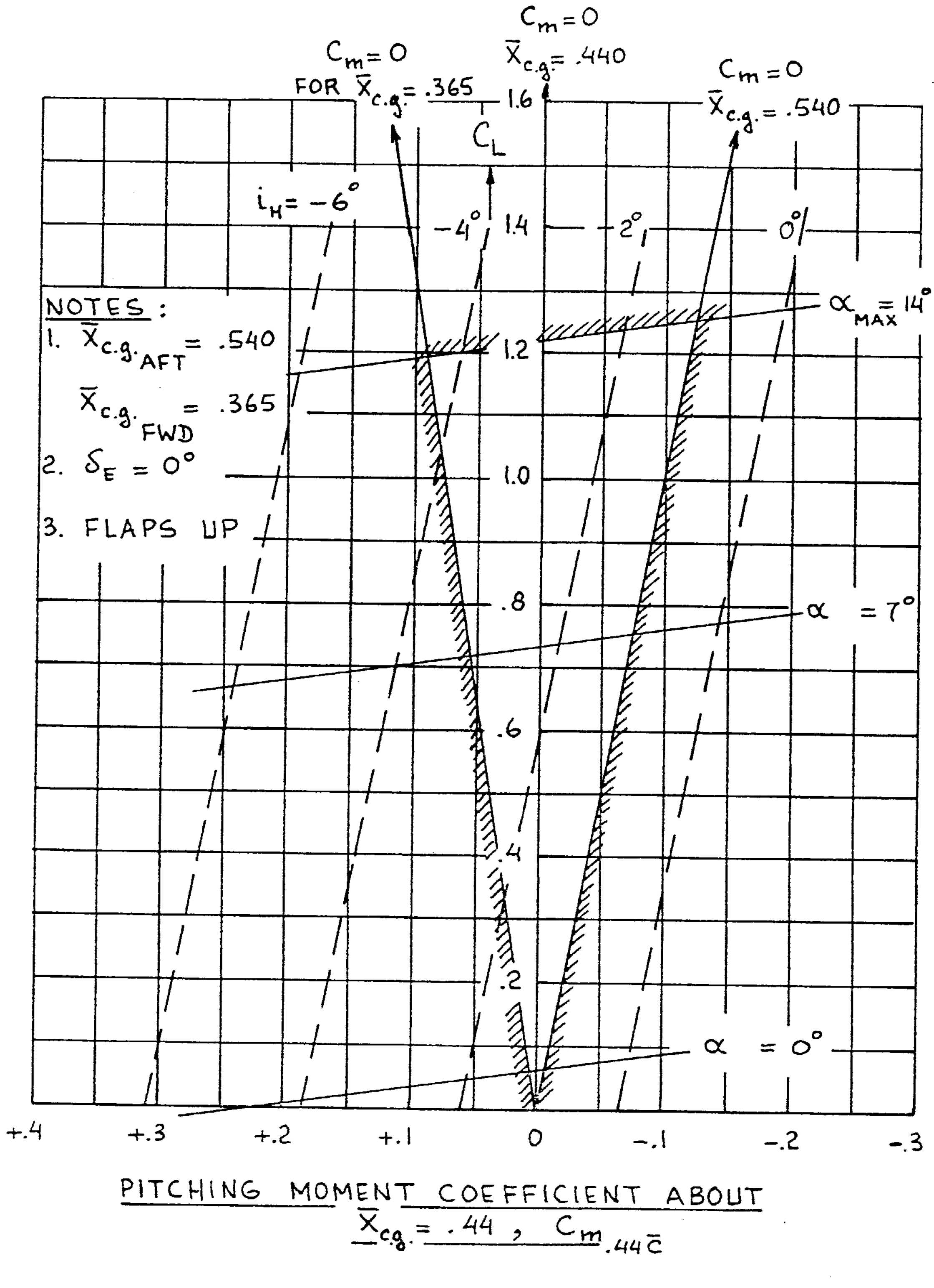
Elevator push (TE down) is required for speed increase.

And if $\quad C_{M\alpha} = 0 \Rightarrow \dfrac{\partial \delta_e}{\partial V} = 0$

So a lot of things go to pot at the neutral point!



Determination of Trim Conditions



Determinations of Trim Conditions for a Light Airplane with Flaps Up

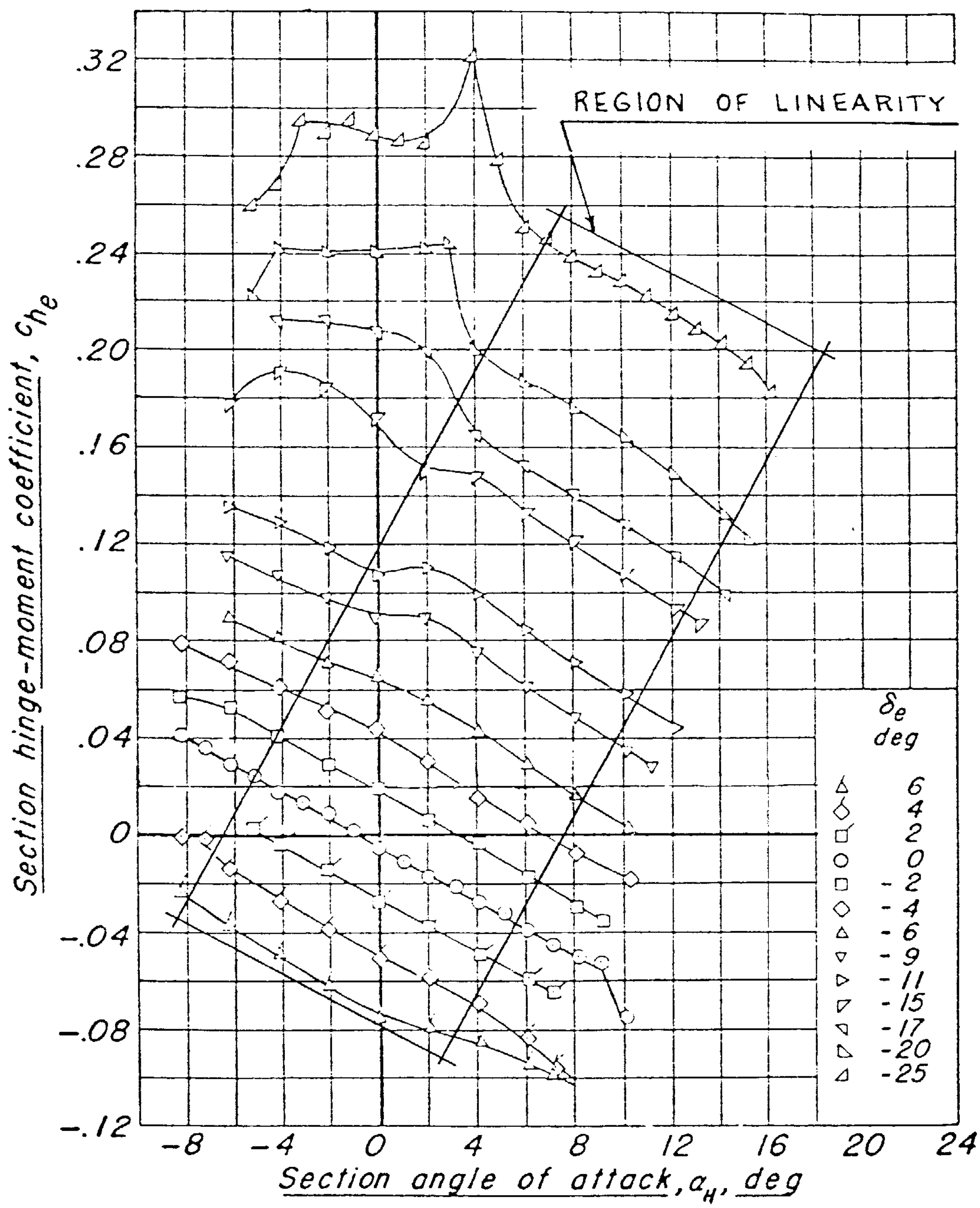
Stick Force Trim

Stick Free Neutral Point

$H.M. = \bar{q}_H S_e \bar{c}_e C_h$

Usually $C_h = C_{h_0} + C_{h_\alpha} \alpha_H + C_{h_{\delta_E}} \delta_E + C_{h_{\delta_t}} \delta_t$

NOTE: ELEVATOR CHORD RATIO .30



Section Elevator Hinge-Moment Characteristics for a Constant Chord Model of the NACA 64A010 Airfoil Section at $R_N = 3 10^6$ and M=.12 (Reproduced from NACA TN 3497)

For a symmetric tail airfoil $C_{h_o} = 0$

$$\alpha_H = \alpha - \epsilon + i_H = \alpha\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_H - \epsilon_o$$

neglecting tab contribution, for a direct geared stick

$$F_S = \bar{q}_H S_e \bar{c}_e G \left\{ C_{h_o} + C_{h_\alpha}\left\{\alpha\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_H - \epsilon_o\right\} + C_{h_{\delta_E}} \delta_E \right\}$$

For trim where

$$\alpha_{TRIM} = \alpha_{o_{TRIM}} + \frac{\partial \alpha}{\partial C_L} C_{L_{TRIM}}$$

$$\delta_{E_{TRIM}} = \delta_{o_{TRIM}} + \frac{\partial \delta_E}{\partial C_L} C_{L_{TRIM}}$$

$$\alpha_{o_{TRIM}} = \frac{-C_{L_o} C_{M_{\delta_E}} + C_{M_o} C_{L_{\delta_E}} - i_H \left(C_{L_{i_H}} C_{M_{\delta_E}} - C_{M_{i_H}} C_{L_{\delta_E}}\right)}{C_{L_\alpha} C_{M_{\delta_E}} - C_{M_\alpha} C_{L_{\delta_E}}}$$

$$\delta_{o_{TRIM}} = \frac{-C_{L_\alpha} C_{M_o} + C_{M_\alpha} C_{L_o} - i_H \left(C_{L_\alpha} C_{M_{i_H}} - C_{M_\alpha} C_{L_{i_H}}\right)}{C_{L_\alpha} C_{M_{\delta_E}} - C_{M_\alpha} C_{L_{\delta_E}}}$$

Using $C_{L_{TRIM}} = \frac{W}{\bar{q} S}$ and $\frac{\bar{q}_H}{\bar{q}} = \eta_H$

$$F_S = \eta_H \bar{q} S_E \bar{c}_E G \left\{ C_{h_o} + C_{h_\alpha}\left(1 - \frac{d\epsilon}{d\alpha}\right)\alpha_{o_{TRIM}} + C_{h_\alpha}(i_H - \epsilon_o) + C_{h_{\delta_E}} \delta_{o_{TRIM}} \right\}$$
$$+ S_E \bar{c}_E G \eta_H \left\{ \frac{\partial \alpha}{\partial C_L} \frac{W}{S} C_{h_\alpha}\left(1 - \frac{d\epsilon}{d\alpha}\right) + \frac{\partial \delta_E}{\partial C_L} \frac{W}{S} C_{h_{\delta_E}} \right\}$$

Stick Force Gradient with Speed

$$\frac{\partial F_S}{\partial V} = \rho \eta_H V S_E \bar{c}_E G \left[C_{h_o} + C_{h_\alpha}\left\{\alpha_{o_{TRIM}}\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_H - \epsilon_o\right\} + C_{h_{\delta_E}} \delta_{o_{TRIM}} \right]$$

For stick force stability $\frac{\partial F_S}{\partial V} < 0$

For FAR minimum $\frac{\partial F_S}{\partial V} \leq \frac{1 \text{ lb}}{6 \text{ KTS}}$

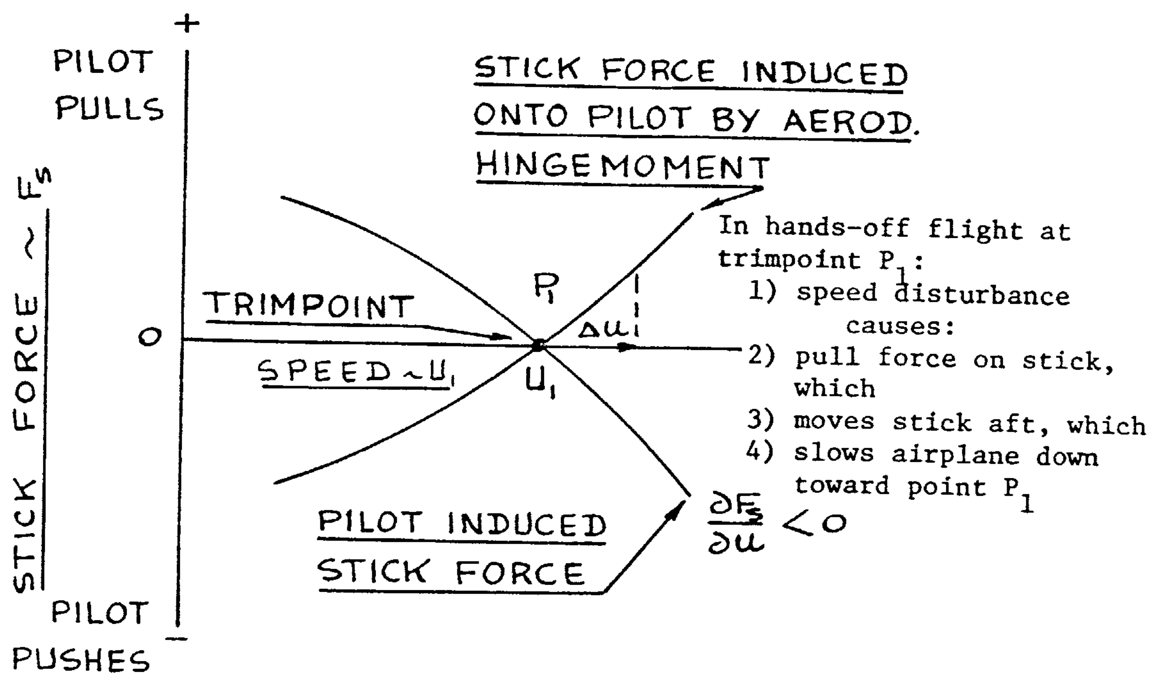
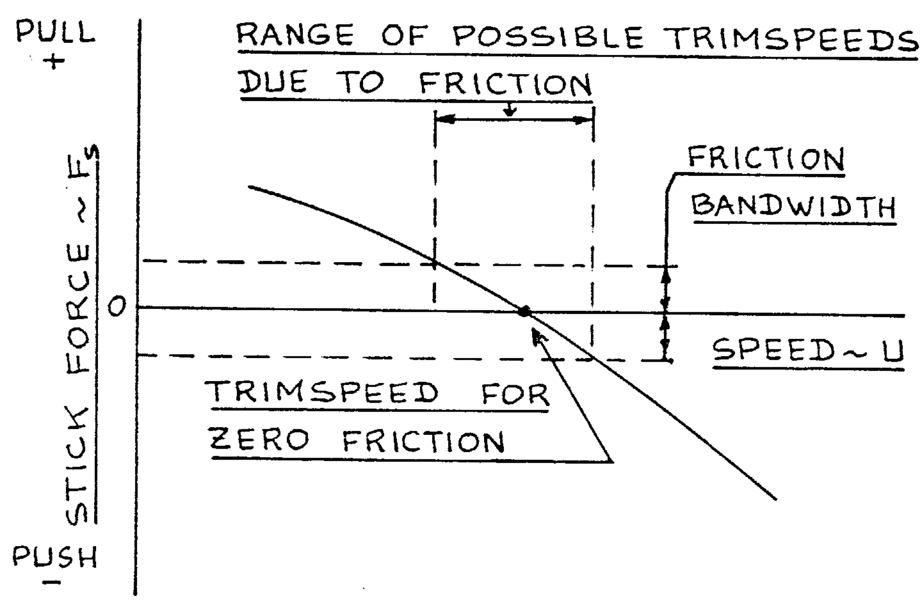


Illustration of Stable Stick Force Behavior

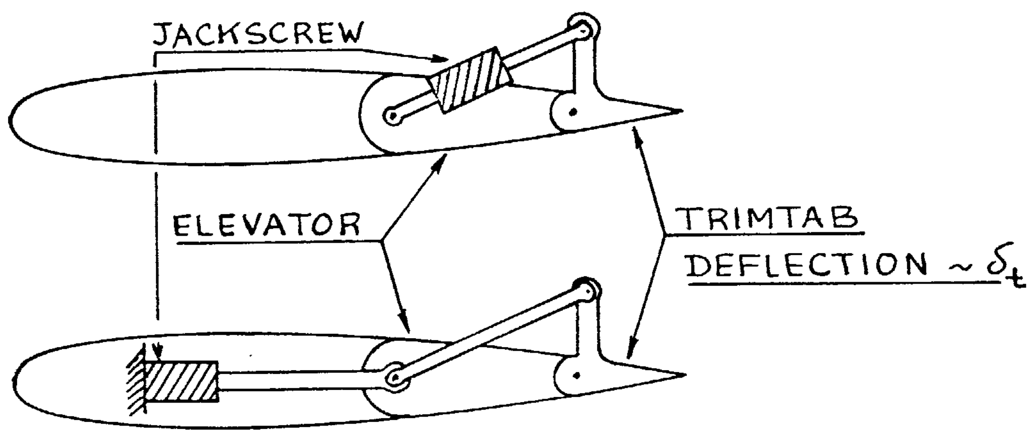


Effect of Friction on Trim Speed Return Characteristics

- C_{h_0} can be altered to enhance stability!
- i_H is effective at accomplishing stick force trim (per equation).
- A trim tab provides a hinge moment bias for trim:

$$C_{h_0} = C_{h_{0_{BASIC}}} + C_{h_{\delta_T}} \delta_T$$

Also effects apparent stability!



Example of an Elevator with Two Types of Trim Tab Configurations

Stick Free Neutral Point

Floating elevator means

$$C_{h_0} + C_{h_\alpha}\alpha_H + C_{h_{\delta_E}}\delta_E = 0$$

so

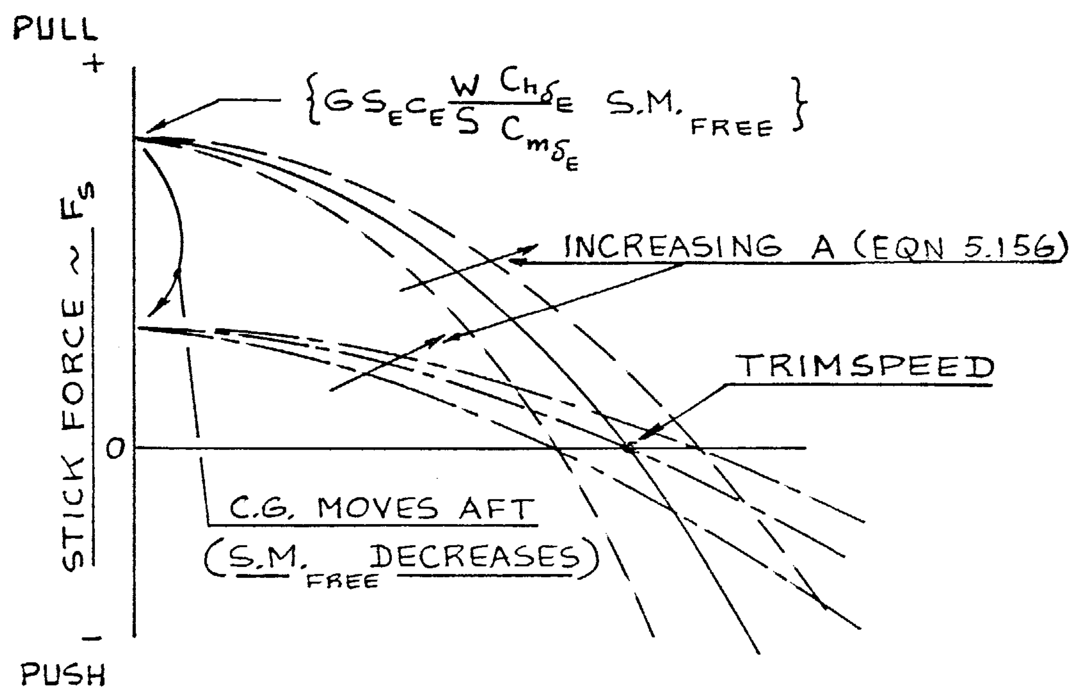
$$\frac{\partial \delta_E}{\partial \alpha} = -\frac{C_{h_\alpha}}{C_{h_{\delta_E}}}\frac{\partial \alpha_H}{\partial \alpha} = -\frac{C_{h_\alpha}}{C_{h_{\delta_E}}}\left(1 - \frac{d\epsilon}{d\alpha}\right)$$

and

$$C_{M_{\alpha}\,\substack{STICK\\FREE}} = C_{L_{\alpha_{WB}}}(\bar{X}_{cg} - \bar{X}_{ac_{WB}}) - C_{L_{\alpha_H}}\eta_H \frac{S_H}{S}(\bar{X}_{ac_H} - \bar{X}_{cg}) \cdot \left\{\left(1 - \frac{d\epsilon}{d\alpha}\right) + \tau_E \frac{\partial \delta_E}{\partial \alpha}\right\}$$

giving

$$\left.\bar{X}_{cg}\right)_{\substack{C_{M_\alpha}=0\\ STICK\ FREE}} = N.P._{\substack{STICK\\FREE}} = \frac{\bar{X}_{ac_{WB}} + \frac{C_{L_{\alpha_H}}}{C_{L_{\alpha_{WB}}}}\eta_H \frac{S_H}{S}\bar{X}_{ac_H}\left(1 - \frac{d\epsilon}{d\alpha}\right)\left(1 - \frac{C_{h_\alpha}\tau_E}{C_{h_{\delta_E}}}\right)}{1 + \frac{C_{L_{\alpha_H}}}{C_{L_{\alpha_{WB}}}}\eta_H \frac{S_H}{S}\left(1 - \frac{d\epsilon}{d\alpha}\right)\left(1 - \frac{C_{h_\alpha}\tau_E}{C_{h_{\delta_E}}}\right)}$$



Example of Variations in Stick-Force with Speed Due to Changes in Stick-Free Static Margin and A.

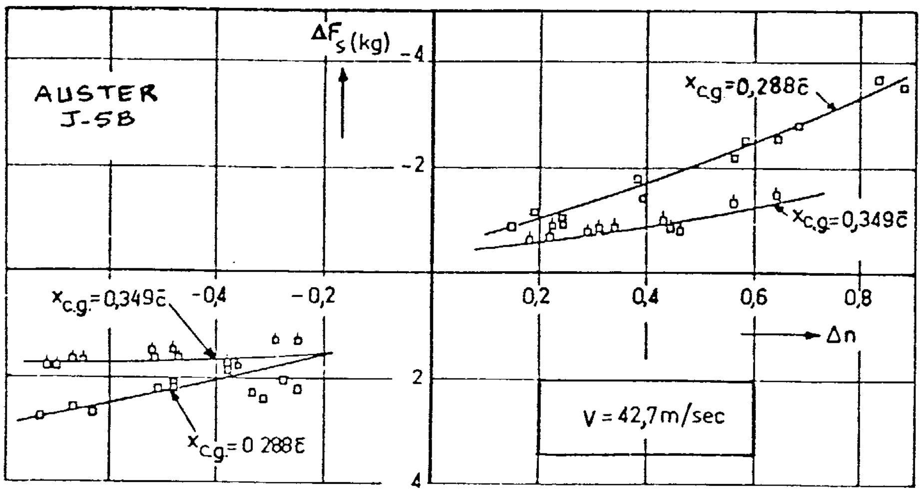
- N.P.-Free is generally forward of stick fixed neutral point

Maneuver Point

- Maneuver point is defined as cg for $\frac{\partial F_s}{\partial n} = 0$

Stick-free maneuver point is:

$$M.P._{FREE} = NP_{FREE} - \left(1 - \frac{C_{h\alpha}\tau_E}{C_{h\delta_E}}\right)\frac{\rho S \bar{c} g}{4W} C_{m_q}$$



Example of the Effect of Center-of-Gravity Location on the Variation of Stick-force with Load-Factor (Copied from Reference 5,5)

Lateral-Directional
Steady State Aerodynamic Derivatives

- Derivatives relate to Side Force, Roll Moment, and Yaw Moment
- Variables are:
  - β
  - δ_A, δ_{SP}, δ_R
  - $\bar{q}$
  - Mach and Reynolds Number

Side Force

$$F_{A_Y} = C_Y \bar{q} S$$

$$C_Y = C_{Y_0} + C_{Y_\beta} \beta + C_{Y_{\delta_R}} \delta_R + C_{Y_{\delta_A}} \delta_A + C_{Y_{\delta_{SP}}} \delta_{SP}$$

- Note how side force is normalized to wing area
- C_{Y_0} is usually $\approx$ zero
- Other coefficients are often non linear and influenced by everything (ie α, ϵ, control cross coupling), behave poorly
- Coefficients are best determined in a wind tunnel and confirmed in flight test.
- $C_{Y_{\delta_{SP}}}$ is associated with a swept hingeline, other contributions due to "secondary" flow

Estimation of Side Force Coefficients

$\underline{C_{Y_\beta}}$

- Wing-body-horizontal tail contribution is small relative to more vertical lifting surfaces

$$C_{Y_\beta} = C_{Y_{\beta_{WB}}} + C_{Y_{\beta_V}} \qquad C_{Y_{\beta_V}} >> C_{Y_{\beta_{WB}}}$$

$$C_{L_V} = C_{L_{\alpha_V}}\left(1 - \frac{d\sigma}{d\beta}\right)\beta \qquad \text{let } \eta_V = \frac{\bar{q}_V}{\bar{q}}$$

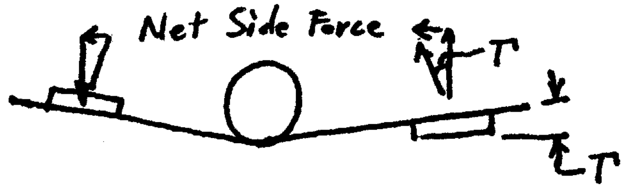
$$C_{Y_{\beta_V}} = -C_{L_{\alpha_V}}\left(1 - \frac{d\sigma}{d\beta}\right)\eta_V \frac{S_V}{S}$$

$\underline{C_{Y_{\delta_R}}}$

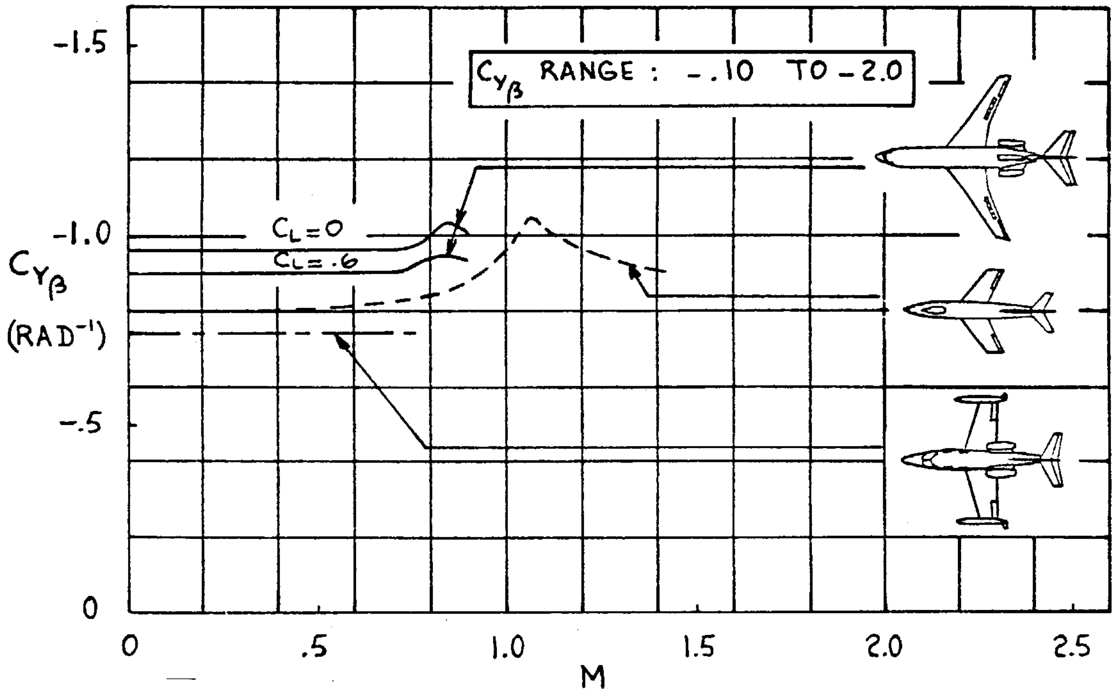
$$C_{Y_{\delta_R}} = C_{L_{\alpha_V}} \alpha_{\delta_R} \eta_V \frac{S_V}{S} \qquad \text{where } \alpha_{\delta_R} = \frac{C_{L_{\delta_R}} \delta_R}{C_{L_\alpha}}$$

$\underline{C_{Y_{\delta_A}}, C_{Y_{\delta_{SP}}}}$

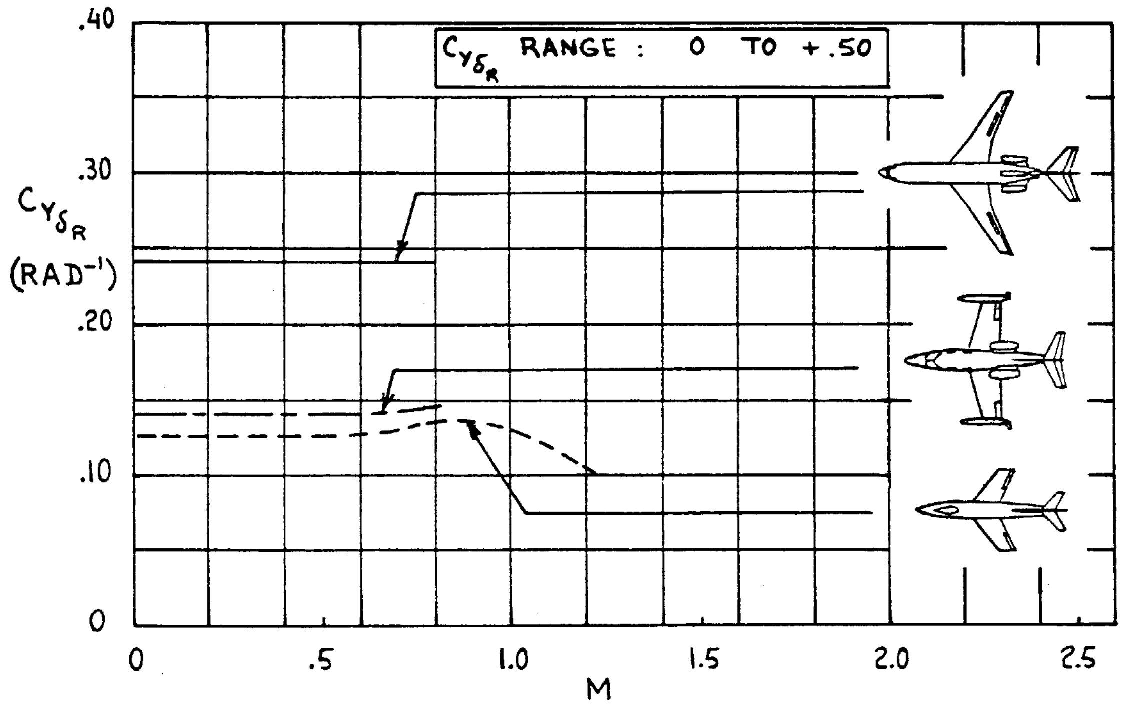
- $C_{Y_{\delta_A}}$ usually ≈ 0, but dihedral is a factor



- $C_{Y_{\delta_{SP}}}$ is usually significant, but cannot be well estimated.



Variation of C_{y_β} with Mach Number for Typical Jet Aircraft



Variation of $C_{y_{\delta_R}}$ with Mach Number for Typical Jet Aircraft

Yaw Moment

$$N_A = C_n \bar{q} S b$$

$$C_n = C_{n_0} + C_{n_\beta}\beta + C_{n_{\delta_R}}\delta_R + C_{n_{\delta_A}}\delta_A + C_{n_{\delta_{SP}}}\delta_{SP}$$

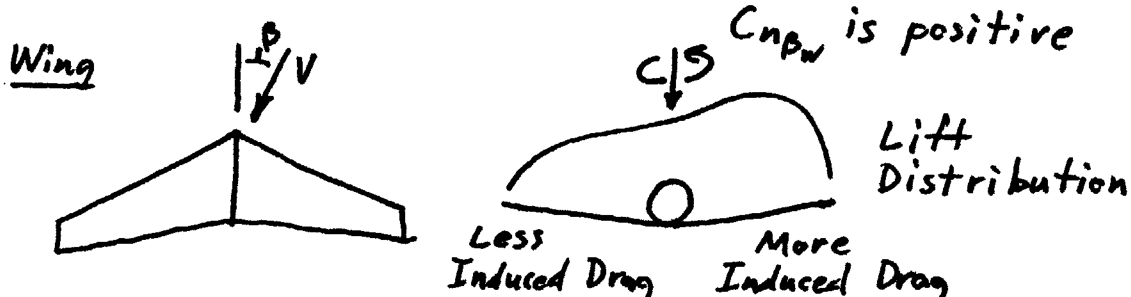
- Note how yaw moment is normalized to wing area and wing span
- C_{n_0} is usually $\simeq$ zero
- Other coefficients can behave poorly in critical flight phases
- Coefficients are best determined in a wind tunnel and confirmed by flight test

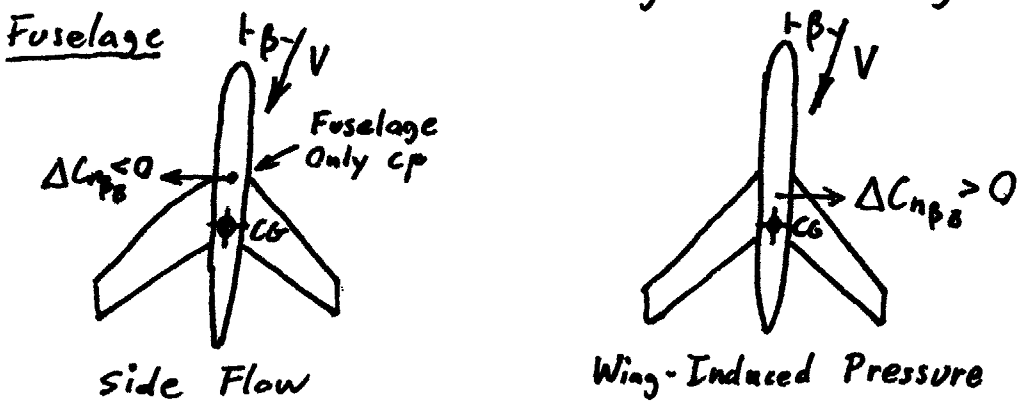
Estimation of Yaw Moment Coefficients

C_{n_β}

$$C_{n_\beta} = C_{n_{\beta_{WB}}} + C_{n_{\beta_V}}$$

$$C_{n_{\beta_{WB}}} = C_{n_{\beta_W}} + C_{n_{\beta_B}}$$





let $X_{v_s} \equiv$ distance from cg to vertical tail cp (¼ chord mac) along x stability axis

$$C_{n_{\beta_v}} = C_{L_{\alpha_v}} \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v X_{v_s}}{S\, b}$$

let $\bar{V}_v = \dfrac{S_v X_{v_s}}{S\, b}$

$C_{n_{\delta_R}}$

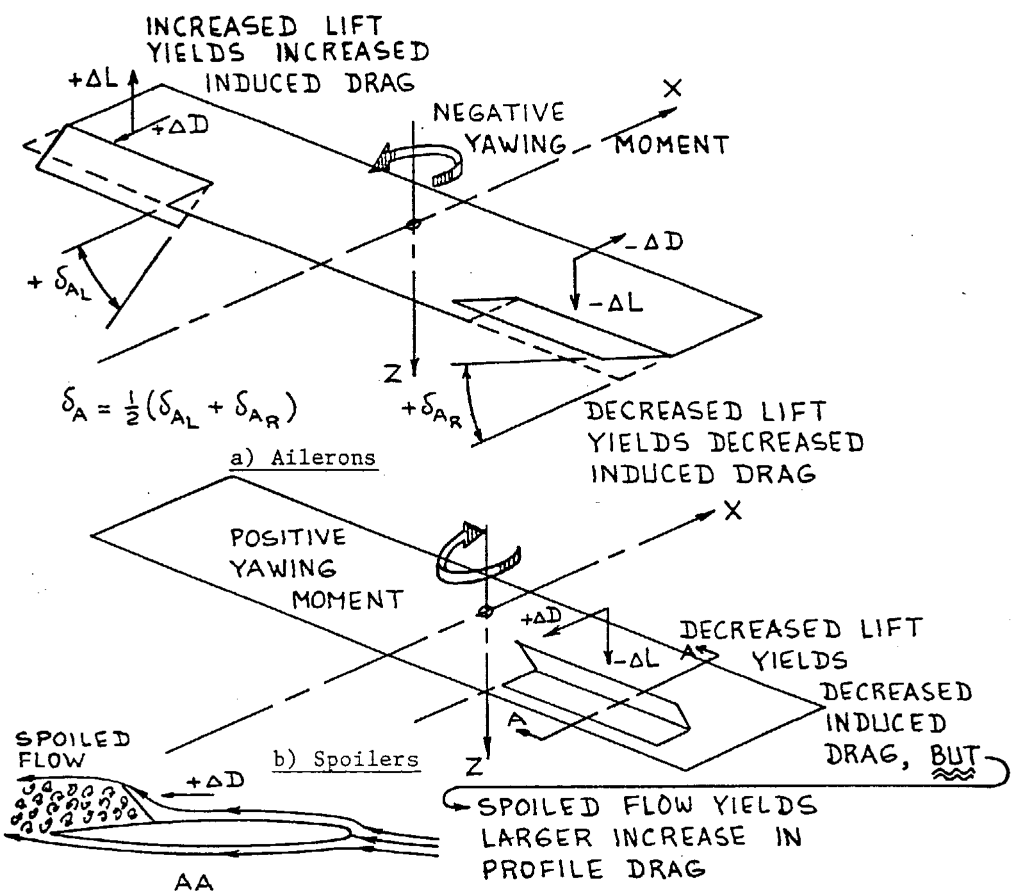
$$C_{n_{\delta_R}} = -C_{L_{\alpha_v}} \alpha_{\delta_R} \eta_v \bar{V}_v$$

$C_{n_{\delta_A}}$

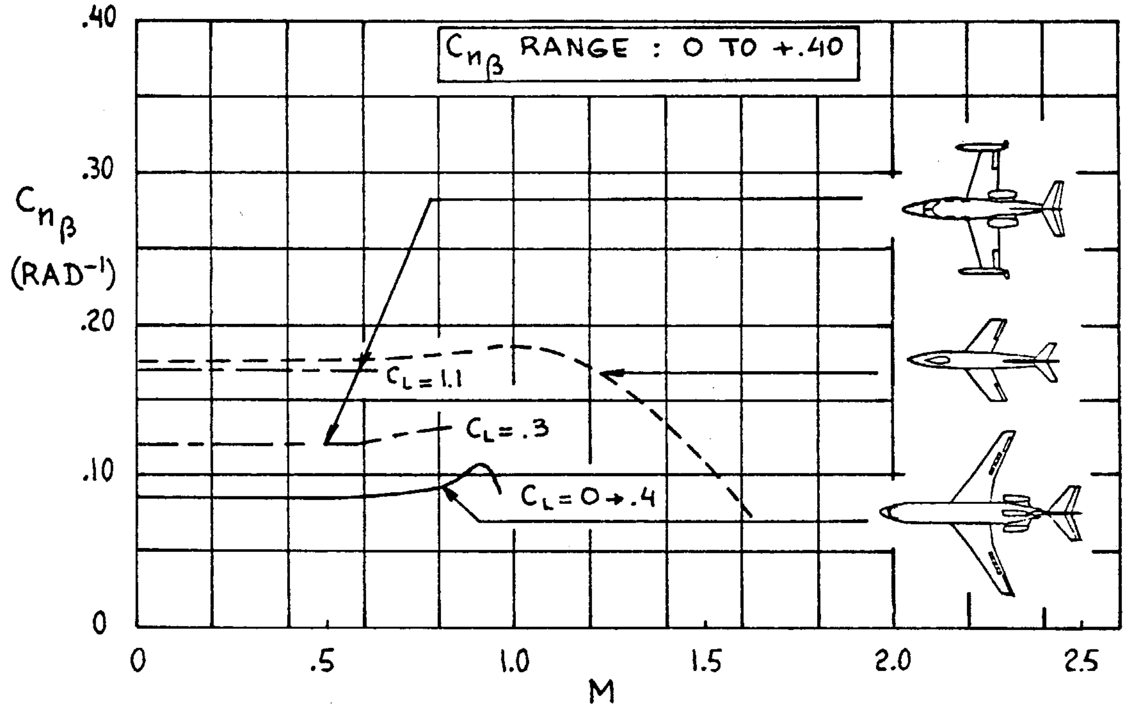
- Best found from wind tunnel data
- Contribution of asymmetric profile and induced drag
- Undesirable contribution

$C_{n_{\delta_{SP}}}$

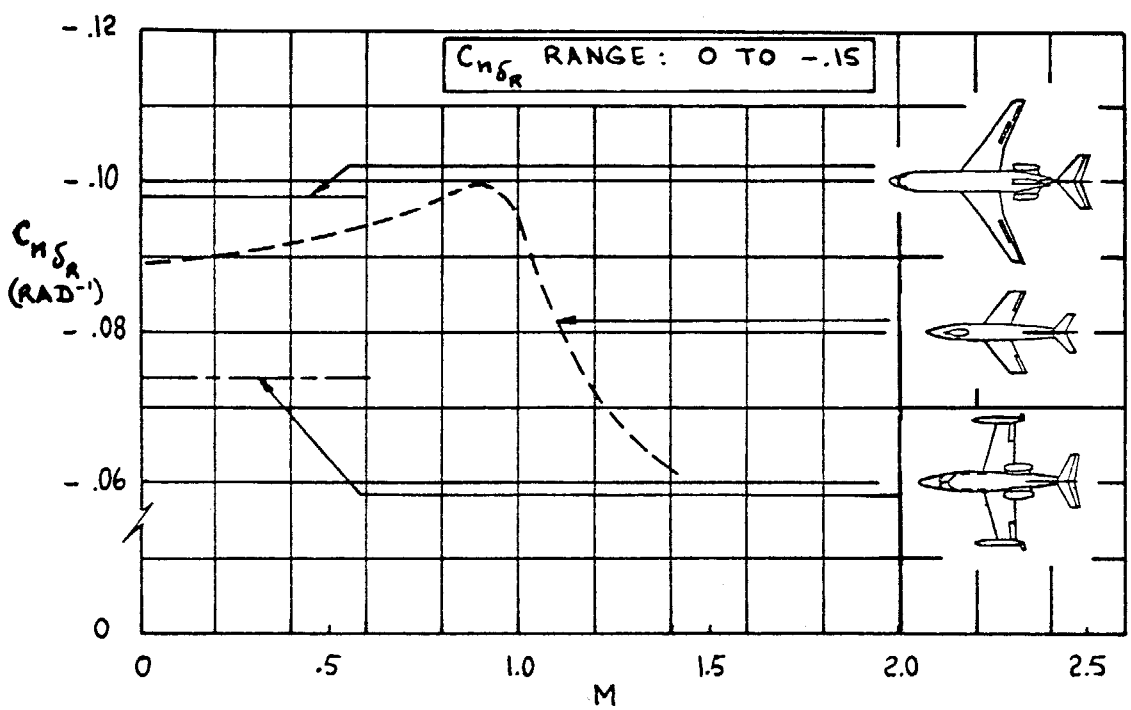
- Mainly due to asymmetric profile drag
- Operates in proper sense
- Best determined in wind tunnel



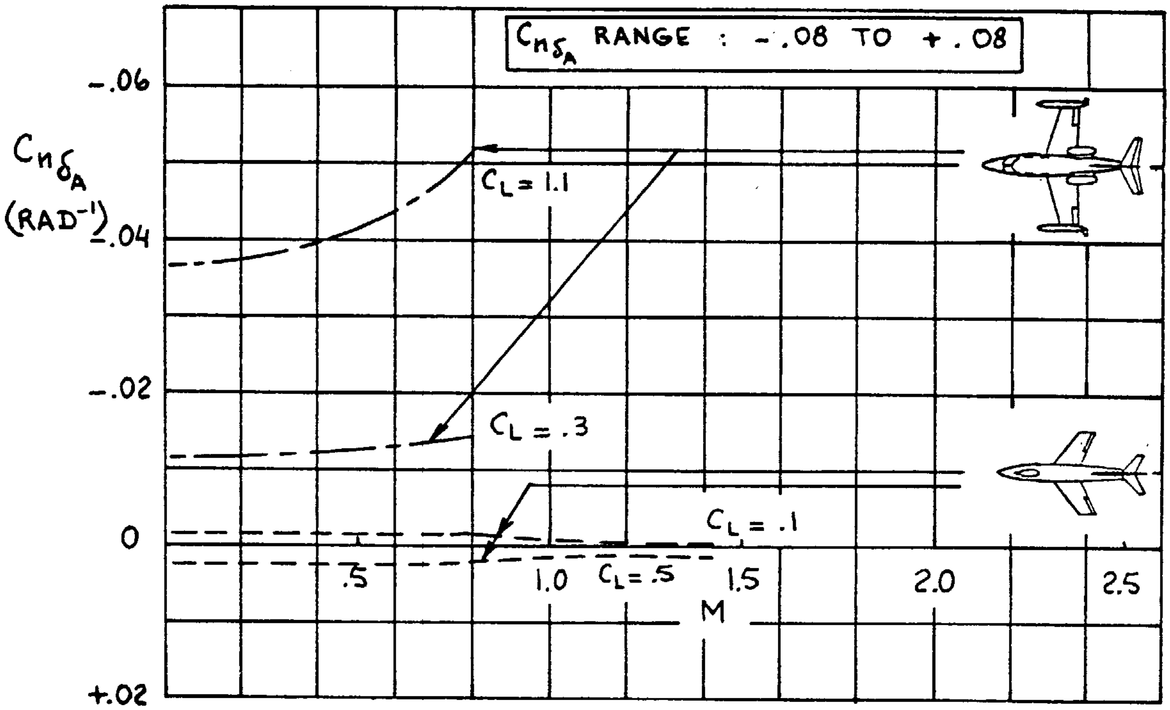
Physical Explanation of Yawing Moment due to Three Lateral Control Devices



Variation of C_{n_β} with Mach Number for Typical Jet Aircraft



Variation of $C_{n_{\delta_R}}$ with Mach Number for Typical Jet Aircraft



Variation of $C_{n_{\delta_A}}$ with Mach Number for Typical Jet Aircraft

Roll Moment

$$L_A = C_\ell \bar{q} S b$$

$$C_\ell = C_{\ell_0} + C_{\ell_\beta}\beta + C_{\ell_{\delta_A}}\delta_A + C_{\ell_{\delta_{SP}}}\delta_{SP} + C_{\ell_{\delta_R}}\delta_R$$

- C_{ℓ_0} is usually $\simeq$ zero
- Other coefficients can behave poorly
- Coefficients are best determined in a wind tunnel and confirmed by flight test.

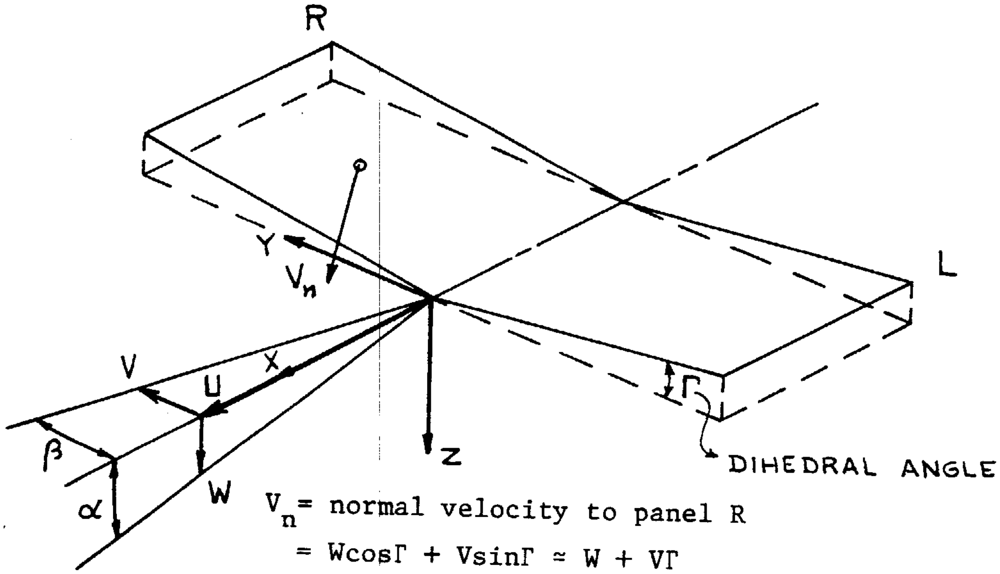
NOTE: + Common caveats for all lateral-directional coefficients!

+ Control contributions provide extreme coupling!

Estimation of Roll Moment Coefficients

C_{ℓ_β} (Dihedral Effect)

$$C_{\ell_\beta} = C_{\ell_{\beta_{WB}}} + C_{\ell_{\beta_H}} + C_{\ell_{\beta_V}}$$



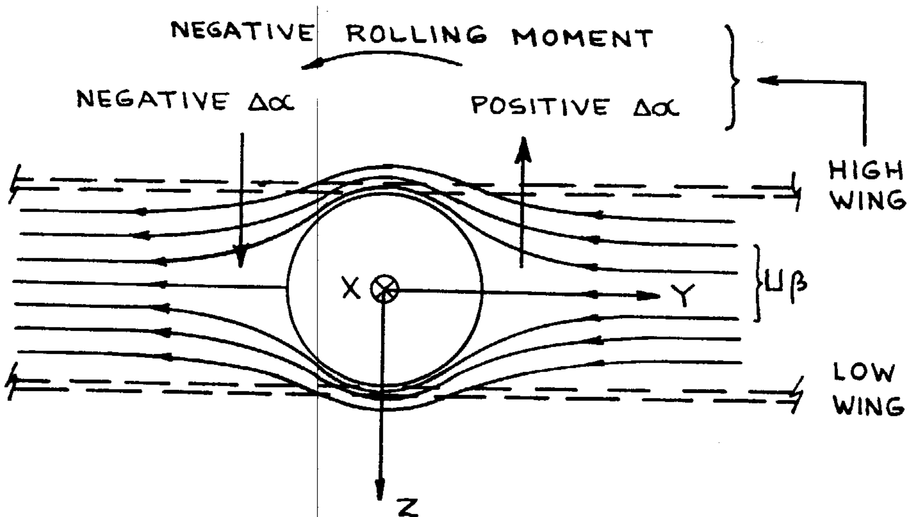
V_n = normal velocity to panel R

$= W\cos\Gamma + V\sin\Gamma \simeq W + V\Gamma$

$\Delta\alpha$ of panel R due to dihedral is:

$\Delta\alpha \simeq \frac{V\Gamma}{V_p} \simeq \frac{V_p\beta\Gamma}{V_p} = \beta\Gamma$ and this produces the lift which in turn produces the rolling moment

Physical Explanation of Rolling Moment due to Sideslip as Affected by Geometric Dihedral



Physical Explanation of Rolling Moment due to Sideslip as Affected by Wing Position on the Fuselage

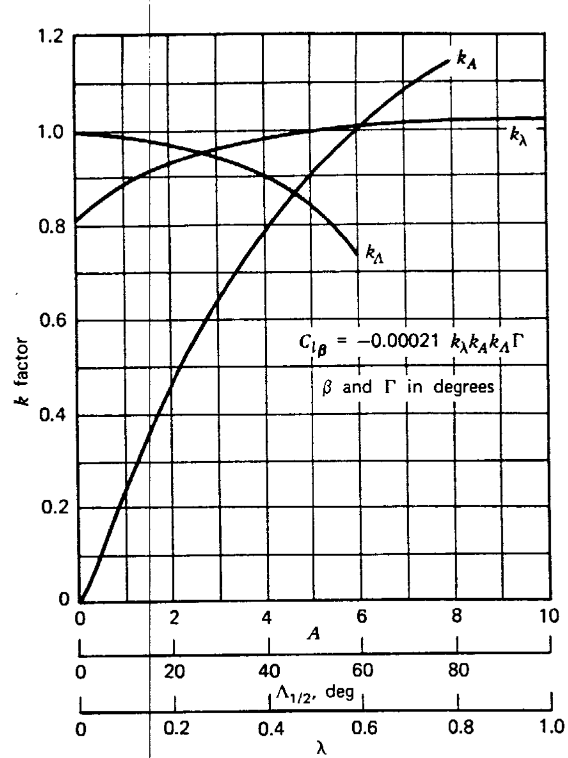
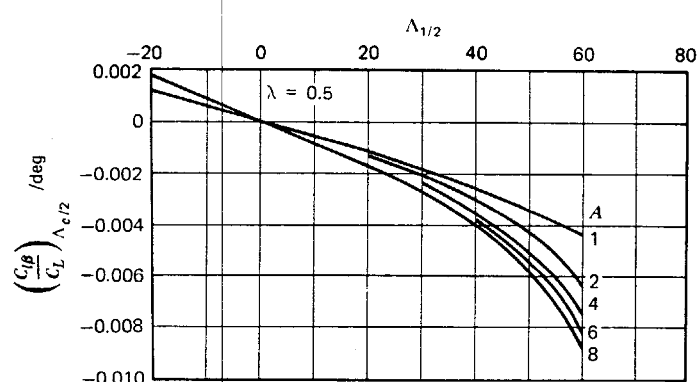


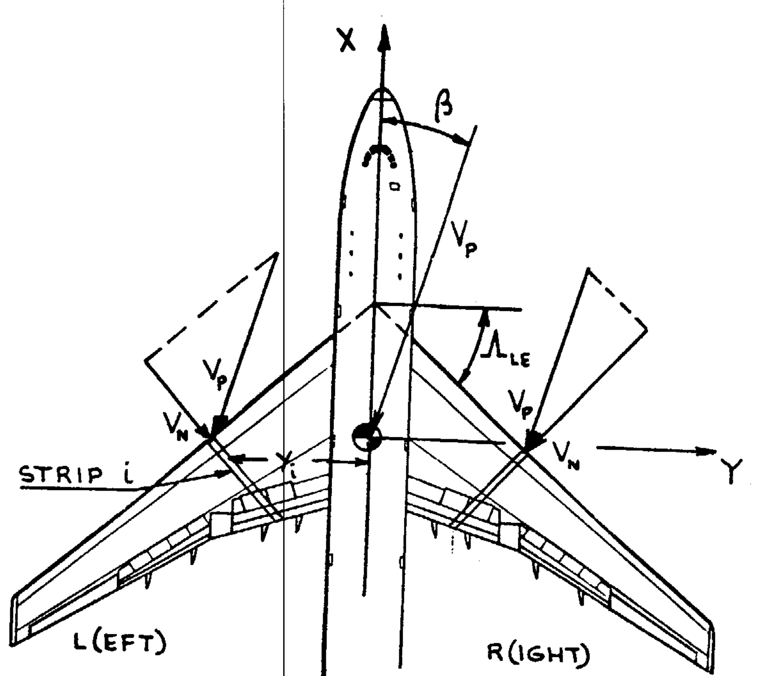
Figure 8.39 Effect of wing dihedral angle on C_{l_β}. Note that factors represent relative effects of varying each parameter independently from the normal case. $\lambda = 0.5$, $A = 6.0$, $\Lambda_{1/2} = 0$.

- Effect of wing sweep for a linearly tapered wing (theoretical)

$$C_{l_{\beta w}}\Big)_{\Lambda} = -\frac{1+2\lambda}{3(1+\lambda)}\, C_L \tan\Lambda$$



Effect of wing sweep on C_{l_β}.

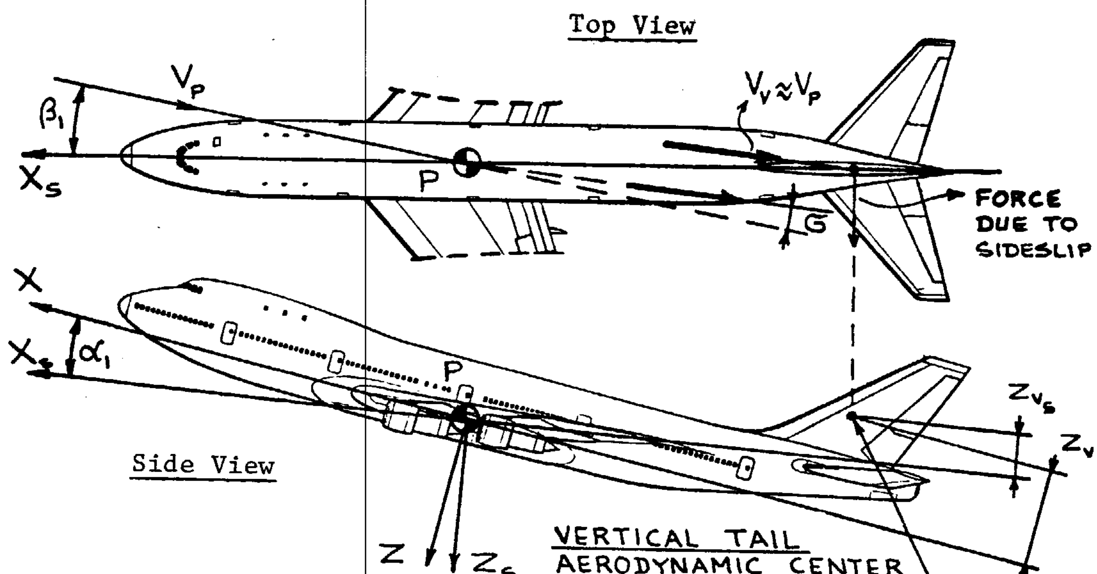


Physical Explanation of Rolling Moment due to Sideslip as Affected by Wing Sweep Angle

$\underline{C_{\ell_{\beta_H}}}$

- Similar to $C_{\ell_{\beta_W}}$
- Referenced $\left.C_{\ell_{\beta_H}}\right)_{REF} = C_{\ell_{\beta_H}} \left(\dfrac{\bar{q}_H S_H b_H}{\bar{q} S b} \right)$

$\underline{C_{\ell_{\beta_V}}}$

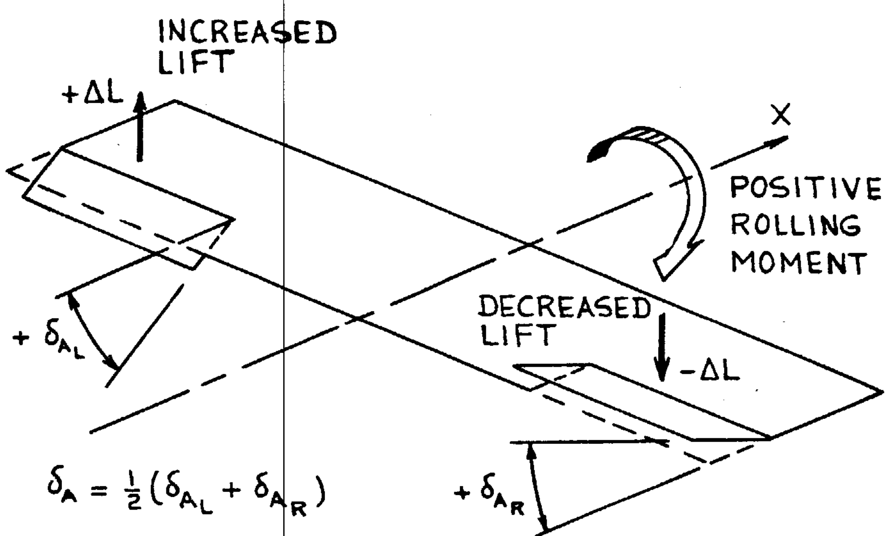


Physical Explanation of Rolling Moment due to Sideslip as Affected by the Vertical Tail

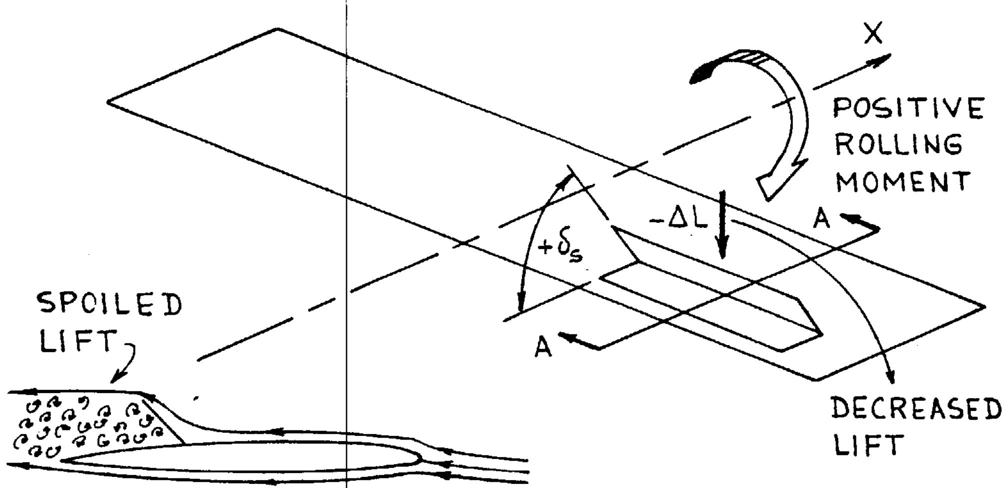
$$C_{l_{\beta_V}} = -C_{L_{\alpha_V}}\left(1 - \frac{d\sigma}{d\beta}\right)\beta\,\eta_V \frac{S_V}{S}\frac{z_{V_S}}{b}$$

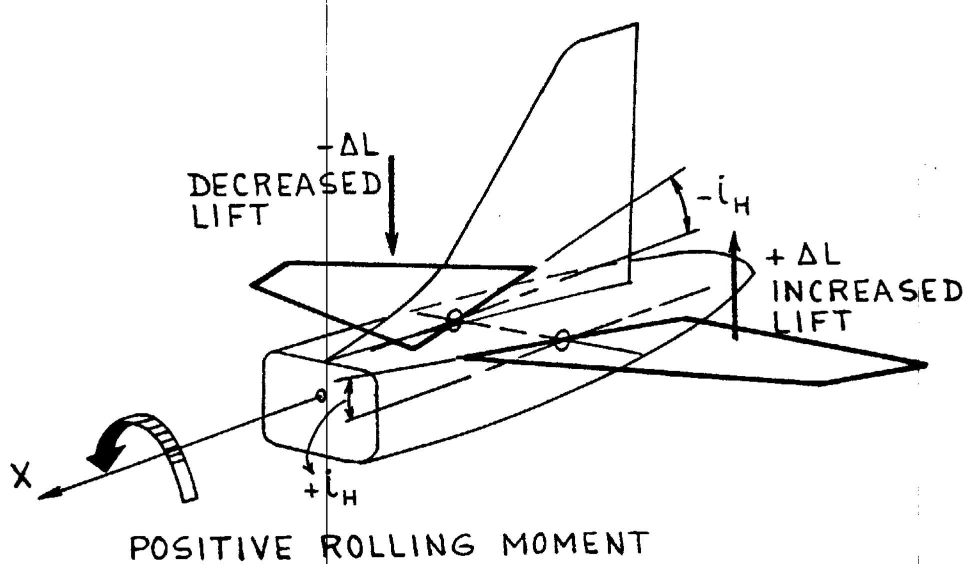
<u>$C_{l_{\delta_A}}, C_{l_{\delta_{SP}}}$</u>

- Analytical methods available for ailerons
- Spoilers and other devices require testing



$$\delta_A = \tfrac{1}{2}(\delta_{A_L} + \delta_{A_R})$$

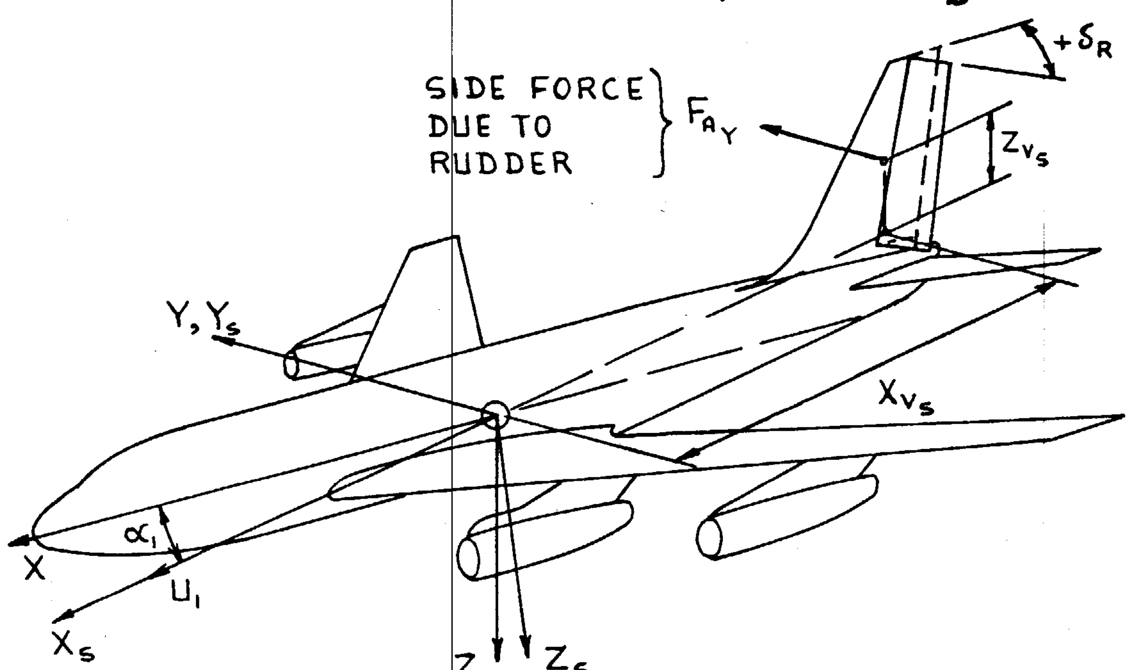




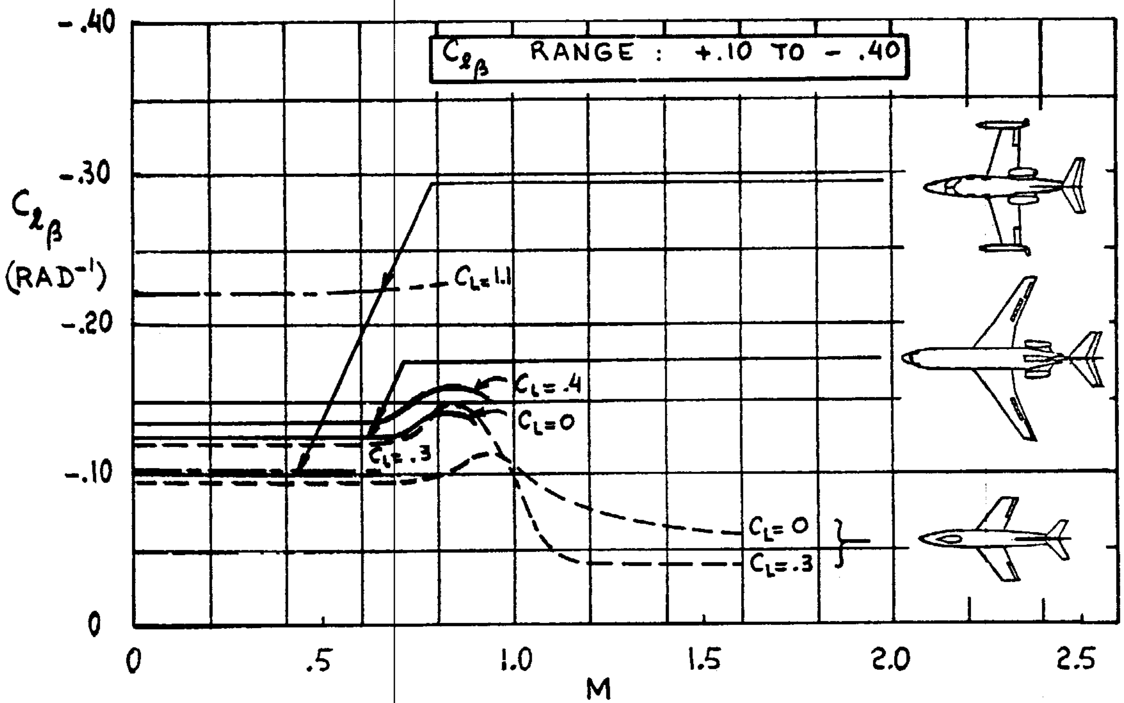
c) Differential Stabilizer

$C_{l_{\delta_R}}$ • Effect of rudder contribution similar to $C_{l_{\beta_V}}$

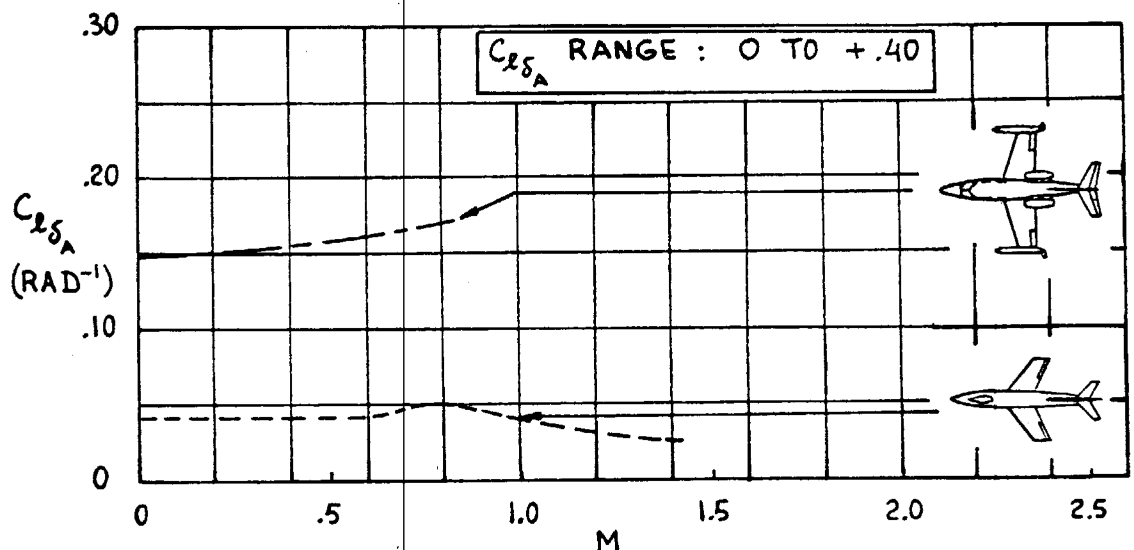
$$C_{l_{\delta_R}} = C_{y_{\delta_R}} \frac{z_{VS}}{b}$$



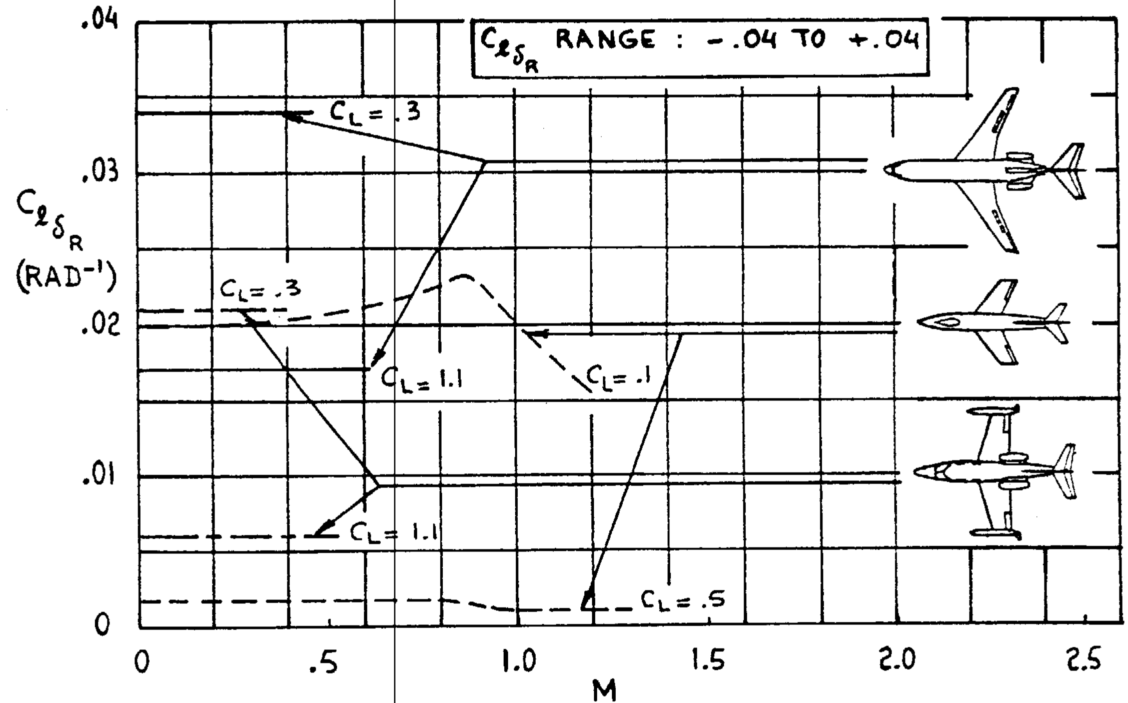
Physical Explanation of Rolling Moment due to Rudder



Variation of C_{ℓ_β} with Mach Number for Typical Jet Aircraft



Variation of $C_{\ell_{\delta_A}}$ with Mach Number for Typical Jet Aircraft



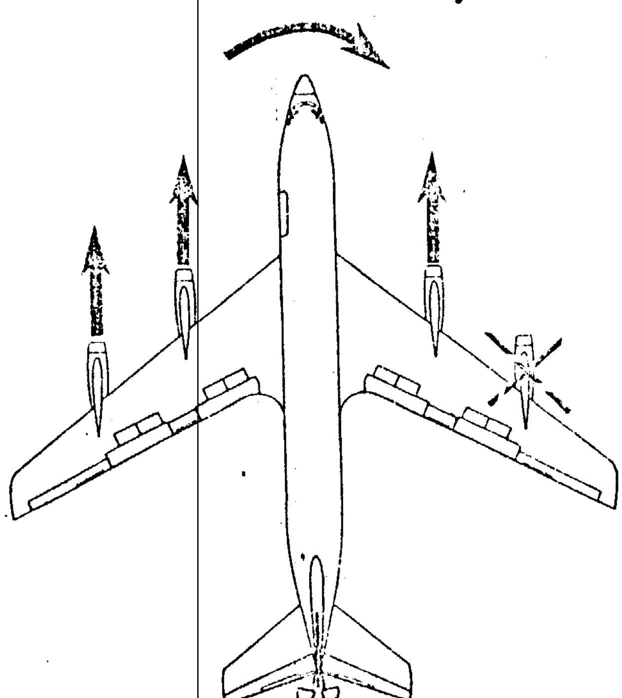
Variation of $C_{\ell_{\delta_{\cancel{A}R}}}$ with Mach Number for Typical Jet Aircraft

Lateral-Directional Static Stability

- Roll away from sideslip, $C_{\ell_\beta} < 0$
- Yaw into sideslip, $C_{n_\beta} > 0$ (weathercocking)
- Static stability not as complicated (or as important) as static longitudinal stability
- Static ~~or~~ control characteristics are very important

Static Lateral-Directional Control

- Control of asymmetric flight condition
- Causes of asymmetry:
  - Engine failure — asymmetric maneuvers
  - Jammed or failed controls
  - Fuel imbalance
  - Hung stores
  - Asymmetric geometry (hatches, pods, bumps)
- Engine failure is a critical case:
  - Reduced performance
  - Engine failure in takeoff – V_{MCG}
  - Engine failure in takeoff flare, landing approach, go-around – V_{MCA}, V_{MCL}
  - Yaw-roll coupling and spoiler pitch coupling requires full fidelity (6 DOF) for analysis



Ground Minimum Control Speed

- Aircraft constrained to $\phi = 0$
- Gear reactions ignored in static analysis
- Compute static balance at recovery condition for rough estimate, $\beta = 0°$
- Recovery speed is $1.03\, V_{MCG}$

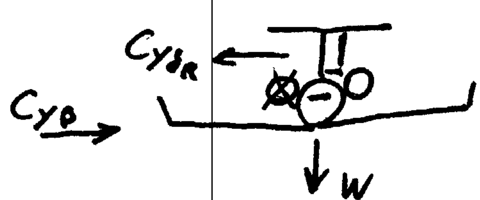
Moment Balance $\quad \dfrac{T \cdot y_e}{q S b} = \overline{V}_v \, \Delta C_{L_v}$ ← due to rudder

$$\text{or} \quad q_{BAL} = \frac{T \cdot y_e}{S b}\left(\frac{1}{\overline{V}_v \cdot \Delta C_{L_v}}\right) = \frac{T \cdot y_e}{S_v l_v} \frac{1}{\Delta C_{L_v}}$$

Air Minimum Control Speed

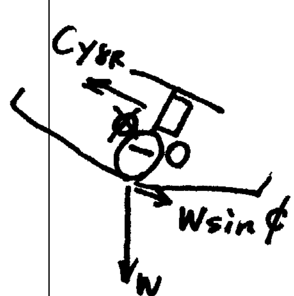
- Aircraft allowed up to 5° bank
- Some sideslip allowed, cannot be excessive

for $\phi = 0$



sideslip relieves sideforce

for $\phi \neq 0$



roll relieves sideforce
rudder authority improved

$$C_{n_e} = \frac{T \cdot y_e}{q S b}$$

$$C_{n_\beta}\beta + C_{n_{\delta_R}}\delta_R + \cancelto{0}{C_{n_{\delta_W}}\delta_W} + C_{n_e} = 0$$

$$C_{l_\beta}\beta + \cancelto{0}{C_{l_{\delta_R}}\delta_R} + C_{l_{\delta_W}}\delta_W + \cancelto{0}{C_{l_e}} = 0$$

$$C_{Y_\beta}\beta + C_{Y_{\delta_R}}\delta_R + \cancelto{0}{C_{Y_{\delta_W}}\delta_W} + \cancelto{0}{C_{Y_e}} + \frac{W}{qS}\sin\phi = 0$$

$$\beta = -\left[\frac{C_{Y_{\delta_R}}\delta_R + \frac{W}{qS}\sin\phi}{C_{Y_\beta}}\right]$$

$$\delta_R = \frac{\left[\frac{C_{n_\beta}}{C_{Y_\beta}}\left(\frac{W}{qS}\right)\sin\phi - C_{n_e}\right]}{\left[C_{n_{\delta_R}} - \frac{C_{n_\beta}}{C_{Y_\beta}} C_{Y_{\delta_R}}\right]}$$

$$\delta_W = -\left[\frac{C_{l_\beta}\beta}{C_{l_{\delta_W}}}\right]$$

Crosswind Landing

$$\beta_{MAX} \approx \sin^{-1}\left[\frac{V_{XWIND}}{V_{APP}}\right]$$

$$C_{n_\beta}\beta + C_{n_{\delta_R}}\delta_R + \cancelto{0}{C_{n_{\delta_W}}\delta_W} = 0$$

$$C_{l_\beta}\beta + \cancelto{0}{C_{l_{\delta_R}}\delta_R} + C_{l_{\delta_W}}\delta_W = 0$$

$$C_{Y_\beta}\beta + C_{Y_{\delta_R}}\delta_R + \cancelto{0}{C_{Y_{\delta_W}}\delta_W} + \frac{W}{qS}\sin\phi = 0$$

Authority Estimate

$$C_{n_{\delta_R}}\delta_R = -\bar{V}_V \Delta C_{L_{V_{\delta_R}}} = -C_{n_\beta} \cdot \beta_{XWIND}$$

Touchdown Bank Angle

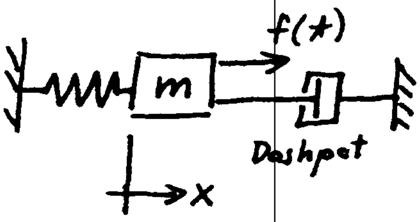
$$\sin\phi \doteq \phi_{RAD} = -\left[\frac{C_{Y_\beta}\beta_{XWIND} + C_{Y_{\delta_R}}\delta_R}{\frac{W}{qS}}\right]$$

Dynamic Stability

Characteristic Behavior can be:

- Divergent
- Convergent
- Oscillatory Divergent
- Oscillatory Convergent

Second Order System (Spring-Mass-Damper)



$$f(t) = m\ddot{x} + c\dot{x} + kx$$

$$\frac{f(t)}{m} = \ddot{x} + \frac{c}{m}\dot{x} + \frac{k}{m}x$$

Let $\omega_n = \sqrt{\frac{k}{m}}$ (undamped natural frequency)

$\zeta = \frac{c}{2\sqrt{km}}$ (damping ratio)

so $\frac{f(t)}{m} = \bar{F}(t) = \ddot{x} + 2\zeta\omega_n\dot{x} + \omega_n^2 x$

Using Laplace transforms

$$\frac{x(s)}{\bar{F}(s)} = \frac{1}{s^2 + 2\zeta\omega_n s + \omega_n^2}$$

Let $\bar{F}(t) = 0$ @ $t < 0$ $\quad \bar{F}(t) = 1.0$ @ $t \geq 0$ (unit step)

so $x(s) = \frac{1}{s(s^2 + 2\zeta\omega_n s + \omega_n^2)}$

solved by finding the roots of the characteristic equation

$$s^2 + 2\xi\omega_n s + \omega_n^2 = 0 = (s-\lambda_1)(s-\lambda_2)$$

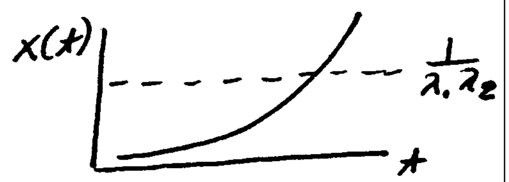
$$x(s) = \frac{A}{s} + \frac{B}{s-\lambda_1} + \frac{C}{s-\lambda_2}$$

Using a table of inverse Laplace Transforms

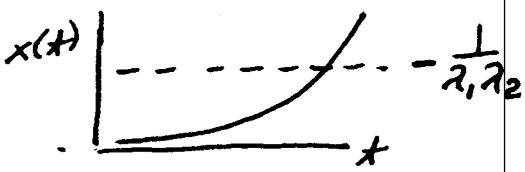
$$x(t) = \frac{1}{\lambda_1\lambda_2} + \frac{1}{\lambda_1(\lambda_1-\lambda_2)} e^{\lambda_1 t} + \frac{1}{\lambda_2(\lambda_2-\lambda_1)} e^{\lambda_2 t}$$

If λ_1 and λ_2 are real

1. λ_1 and λ_2 positive $\Rightarrow$ divergent



2. One root positive, one root negative $\Rightarrow$ divergent



3. Both roots negative $\Rightarrow$ ~~converg~~ convergent

If both λ_1 and λ_2 are complex

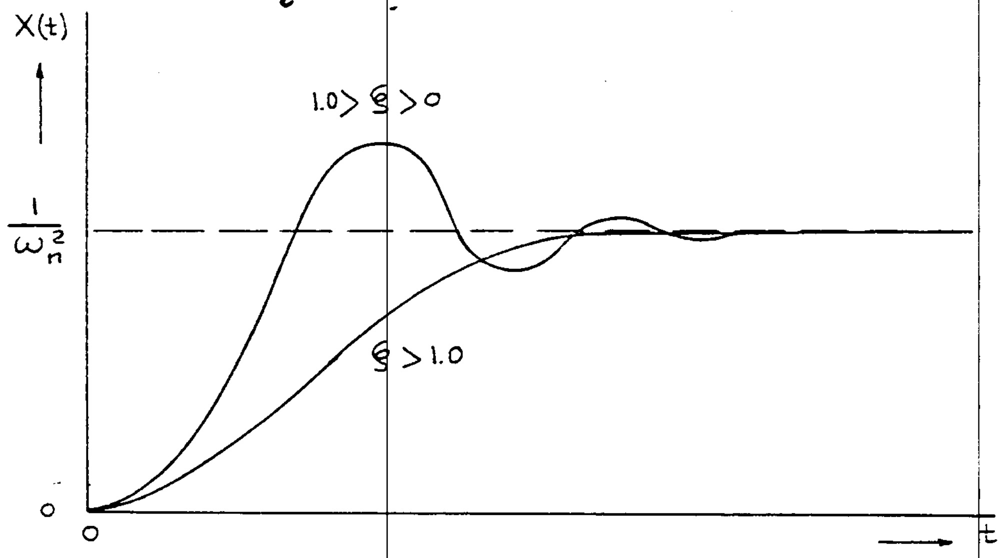
Let $\lambda_1 = n + jw$, $\lambda_2 = n - jw$

$$w = w_n\sqrt{1-\zeta^2} = w_n\sqrt{\frac{k}{m}}$$

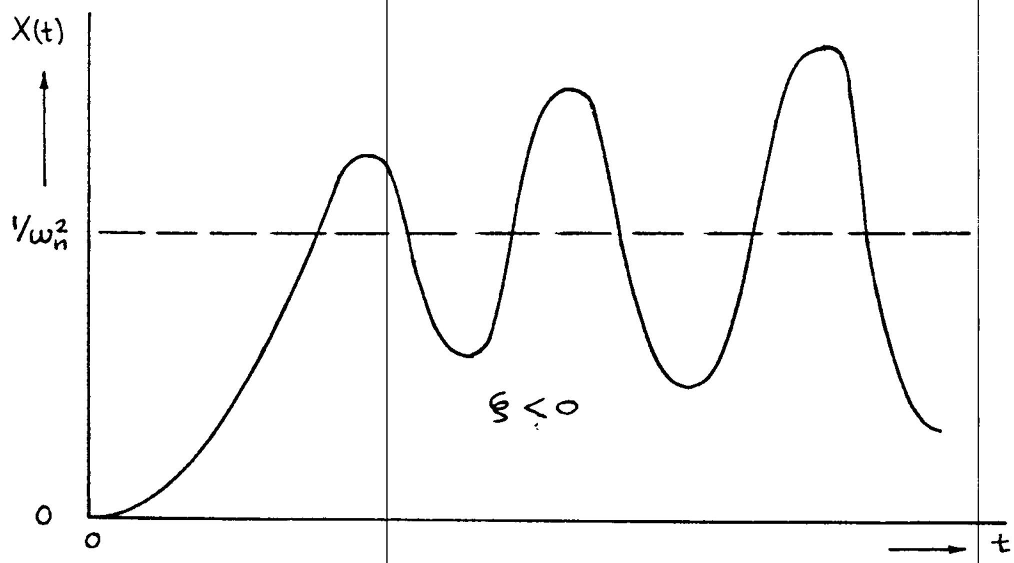
$$x(t) = \frac{1}{w_n^2}\left\{1 - e^{-\zeta w_n t}\sqrt{\frac{1}{1-\zeta^2}}\,\sin(w_n\sqrt{1-\zeta^2}\,t + \phi)\right\}$$

where $\phi = \sin^{-1}\sqrt{1-\zeta^2}$

when $\zeta = 1.0$ $c = 2\sqrt{km}$



Example of Time Response Behavior of Equation (6.70) for $\zeta>1.0$ and for $1.0>\zeta>0$



Example of Time Response Behavior of Equation (6.70) for $\zeta<0$

These properties and the following stability criteria are typical for all systems of linear differential equations. The stability nature of the response is determined entirely by:

1. The type of the roots of the characteristic equation.
2. The sign of the real part of the roots of the characteristic equation.

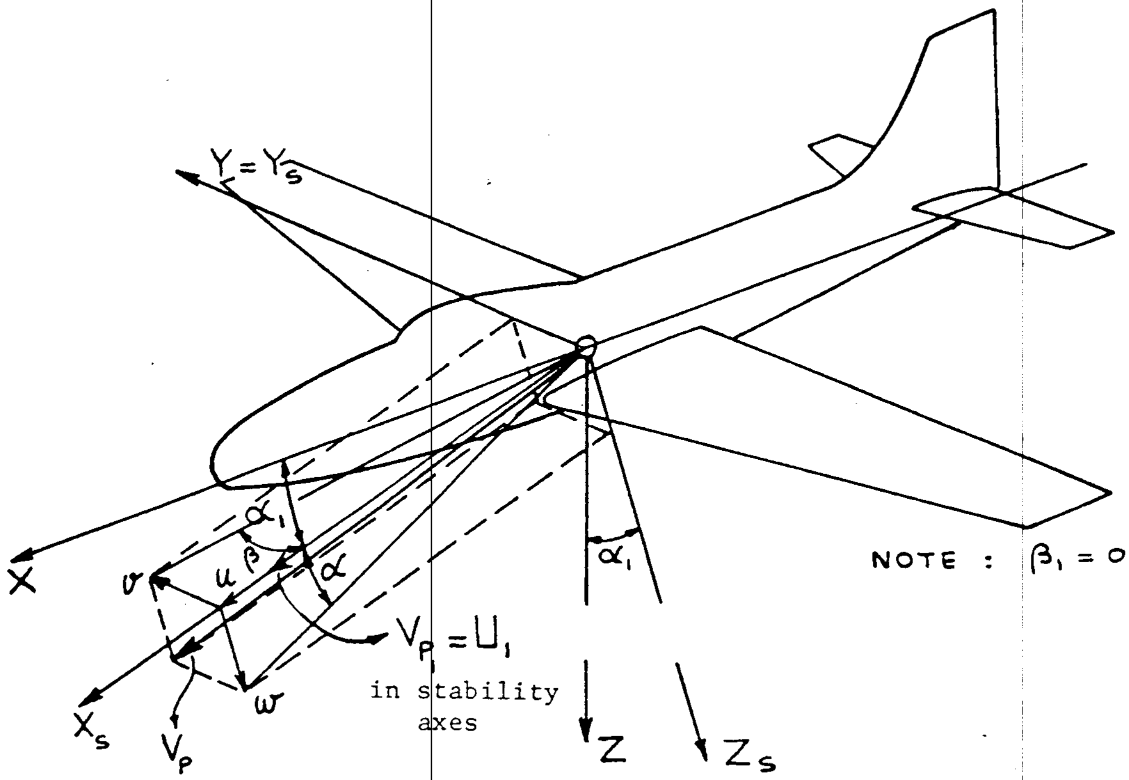
Stability Criteria for a Linear System

1. Stable only if the real parts of the roots of the characteristic equation are negative.
2. Convergent (stable) if the roots are real and negative.
3. Divergent (unstable) if the roots are real and positive.
4. Oscillatory convergent (stable) if the real parts of the roots are negative.
5. Oscillatory divergent (unstable) if the real parts of the roots are positive.

The utility of this analytical approach is why so much effort is expended in pseudo-linearization of aircraft model equations.

Dynamic Stability Derivatives
or Perturbed State Derivatives

(non-steady state)



In vector notation:

$\vec{V}_p = \vec{U}_1 + \vec{u} + \vec{v} + \vec{w}$, where $\vec{V}_p$ is the perturbed state total velocity, $\vec{U}_1$ is the steady state total velocity, while $\vec{u}, \vec{v}, \vec{w}$ are the perturbed velocities

$$\alpha = \arctan \frac{w}{U_1 + u} \simeq \frac{w}{U_1}$$

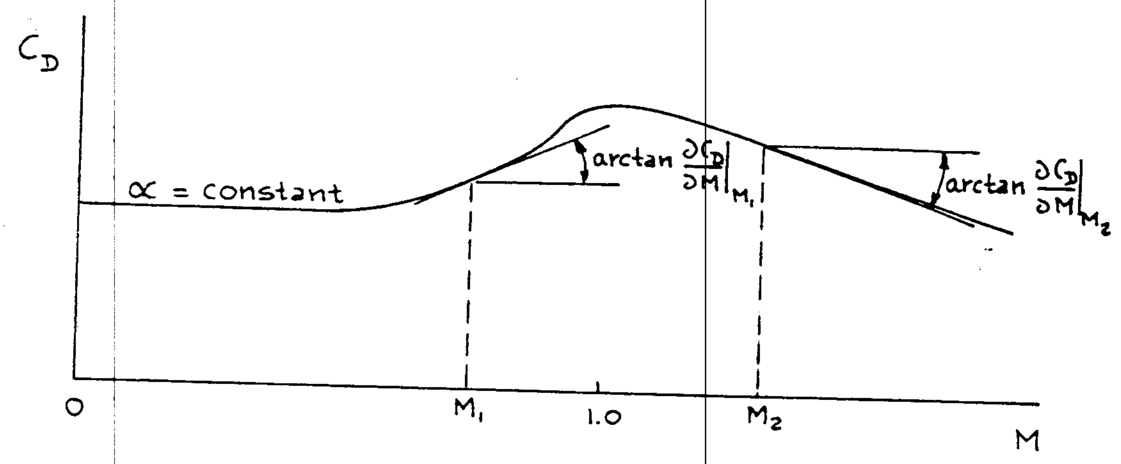
$$\beta = \arctan \frac{v}{U_1 + u} \simeq \frac{v}{U_1}$$

Interpretation of Perturbed State Velocities and Angles

u Perturbation $(C_{L_u}, C_{D_u}, C_{m_u})$

$$\frac{\partial F_{A_x}}{\partial\left(\frac{u}{U_1}\right)} = -(C_{D_u} + 2C_{D_1})\bar{q}_1 S$$

since $C_{D_u} = M_1 \dfrac{\partial C_D}{\partial M}$



Example of Determination of $\partial C_D/\partial M$

$$\frac{\partial F_{A_z}}{\partial\left(\frac{u}{U_1}\right)} = -(C_{L_u} + 2C_{L_1})\bar{q}_1 S$$

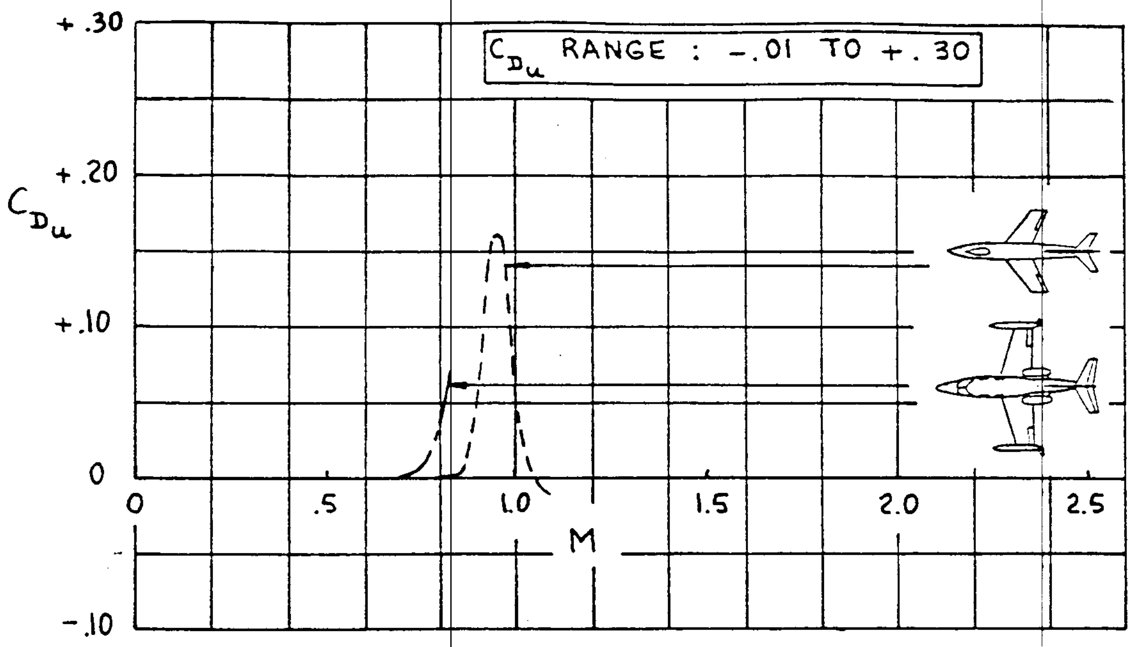
typically $C_{L_u} = \dfrac{M_1^2}{1-M_1^2} C_{L_1}$

$$\frac{\partial M}{\partial\left(\frac{u}{U_1}\right)} = (C_{m_u} + 2C_{m_1})\bar{q}_1 S \bar{c}$$

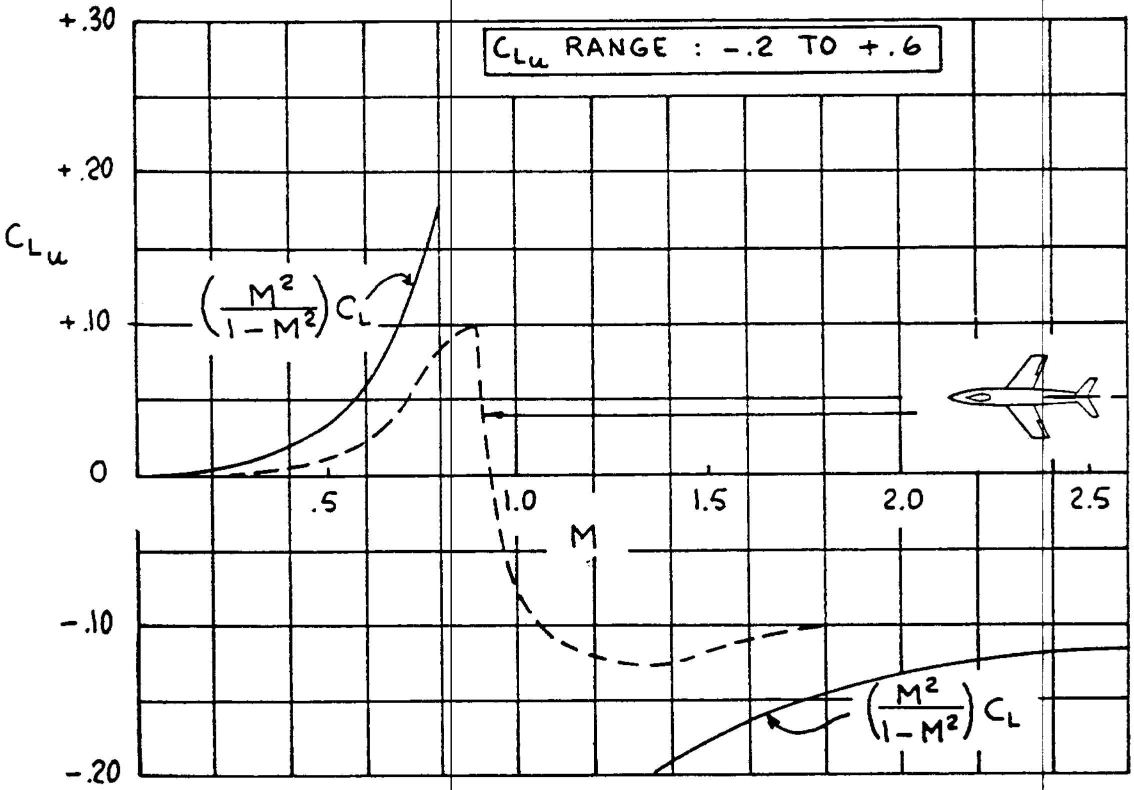
typically, if $\dfrac{\partial C_{m_0}}{\partial M} \approx 0$ $\quad \dfrac{\partial C_m}{\partial M} = -C_{L_1}\dfrac{\partial \bar{x}_{ac}}{\partial M}$

for $M<0$ $\quad \dfrac{\partial \bar{x}_{ac}}{\partial M} > 0 \Rightarrow C_{m_u} < 0$

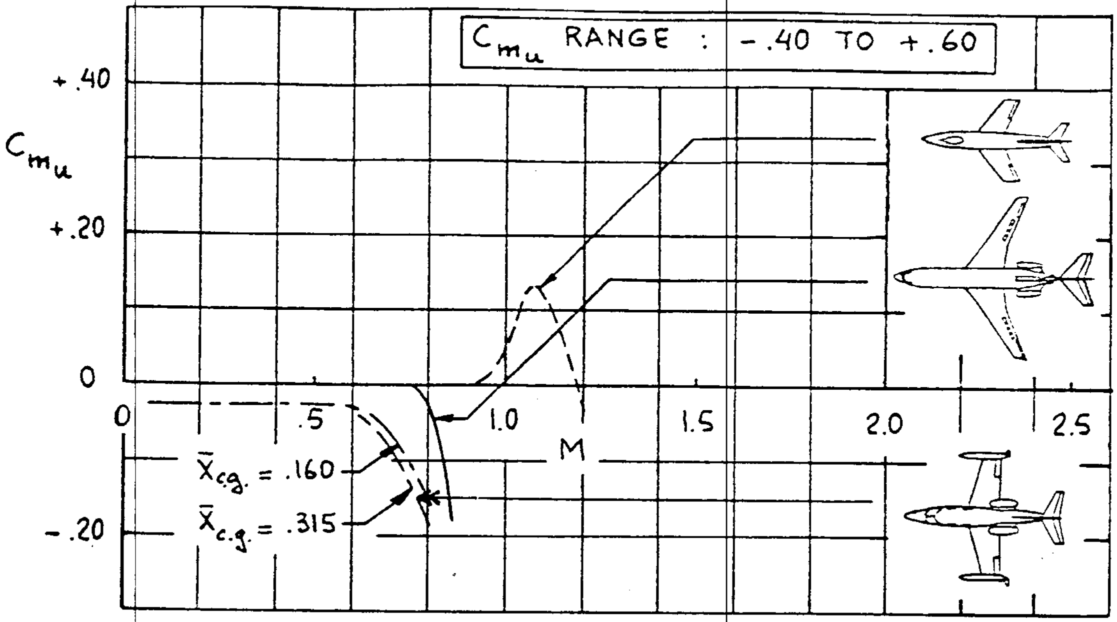
This is transonic tuck



Variation of C_{D_u} with Mach Number for Typical Jet Aircraft



Variation of C_{L_u} with Mach Number for Typical Jet Aircraft

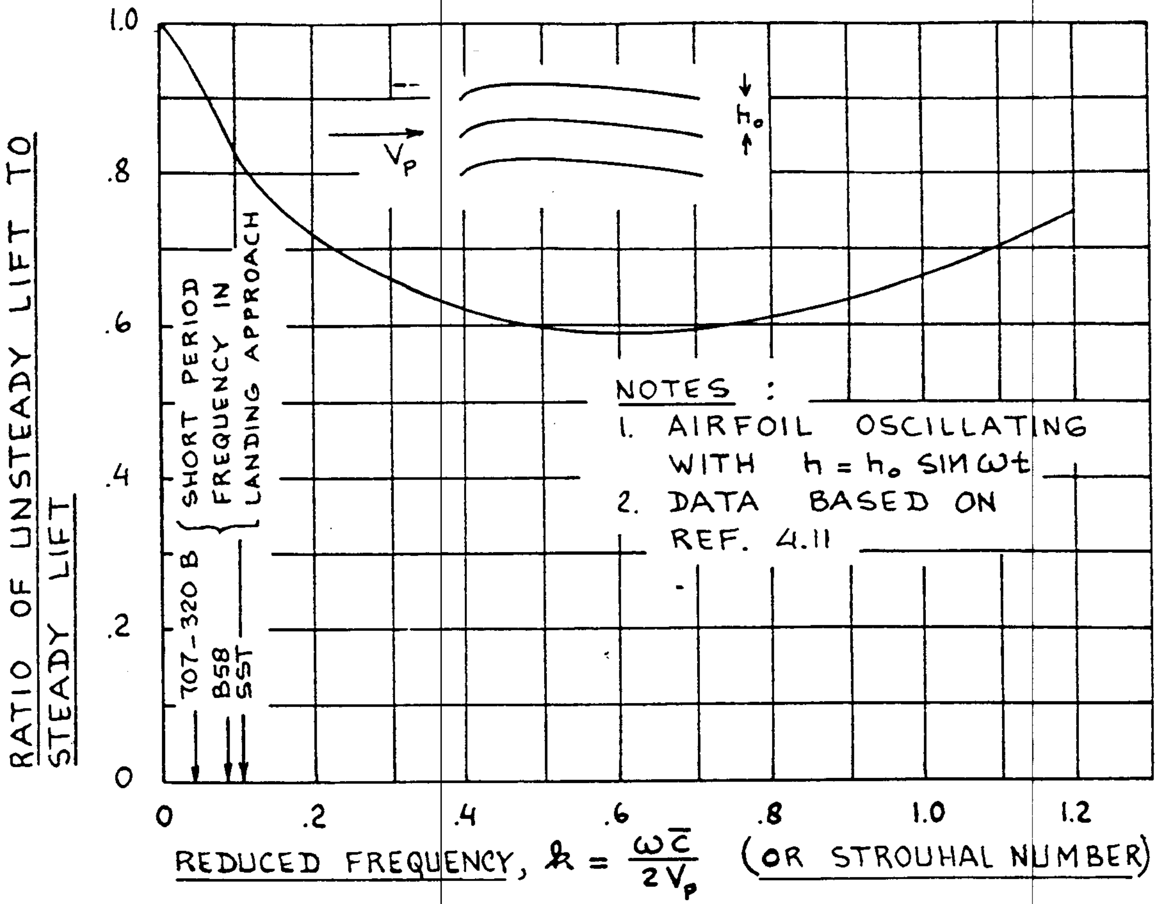


Variation of C_{m_u} with Mach Number for Typical Jet Aircraft

$\dot{\alpha}$ Derivatives ($C_{L_{\dot{\alpha}}}$, $C_{D_{\dot{\alpha}}}$, $C_{m_{\dot{\alpha}}}$)

- Estimation methods are poor
- Potential panel methods are best
- Conventional aircraft $\dot{\alpha}$ influence is explained by downwash lag
- $\dot{\alpha}$ derivatives are dependent on reduced frequency $k = \frac{\bar{c}\dot{\alpha}}{2U_1}$.
  Unsteady lift contribution is neglected by quasi-steady assumption, generally limits $k < 0.04$. Paradox!

Downwash Lag $\quad \Delta\varepsilon = -\frac{d\epsilon}{d\alpha}\dot{\alpha}\Delta t = -\frac{d\epsilon}{d\alpha}\dot{\alpha}\frac{x_H}{U_1}$



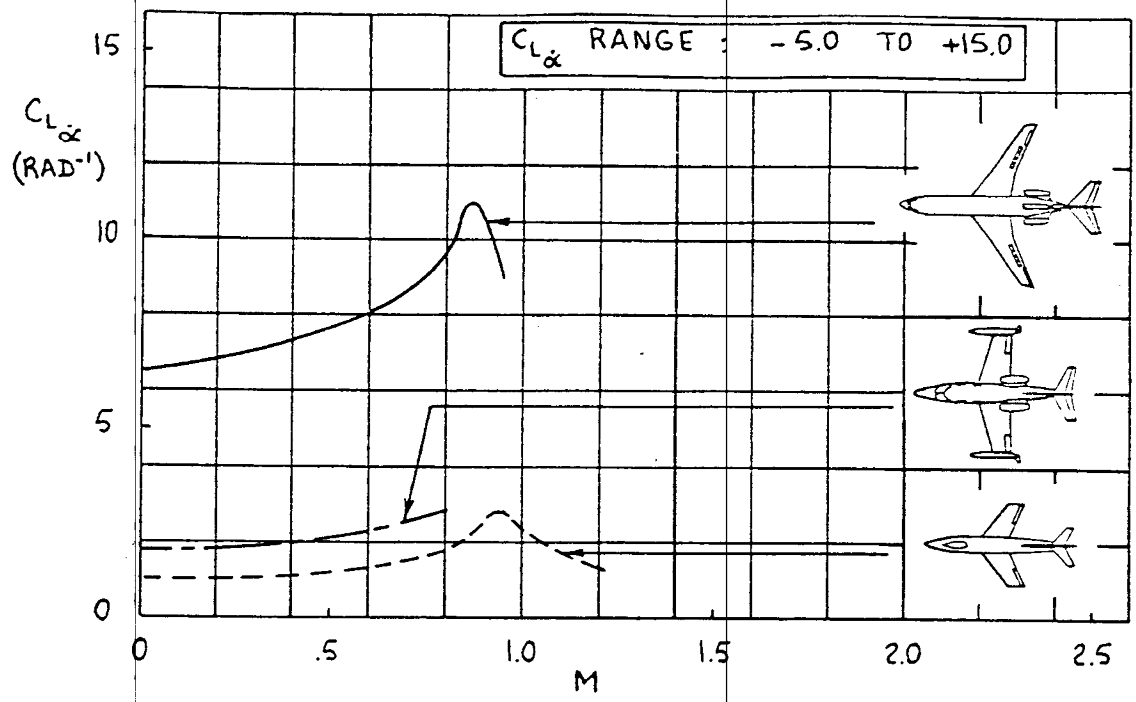
Effect of Frequency of Angle of Attack Oscillation on Lift Attenuation

$$\frac{\partial F_{A_X}}{\partial\left(\frac{\dot{\alpha}\bar{c}}{2U_1}\right)} = -C_{D_{\dot{\alpha}}}\bar{q}_1 S = 0$$

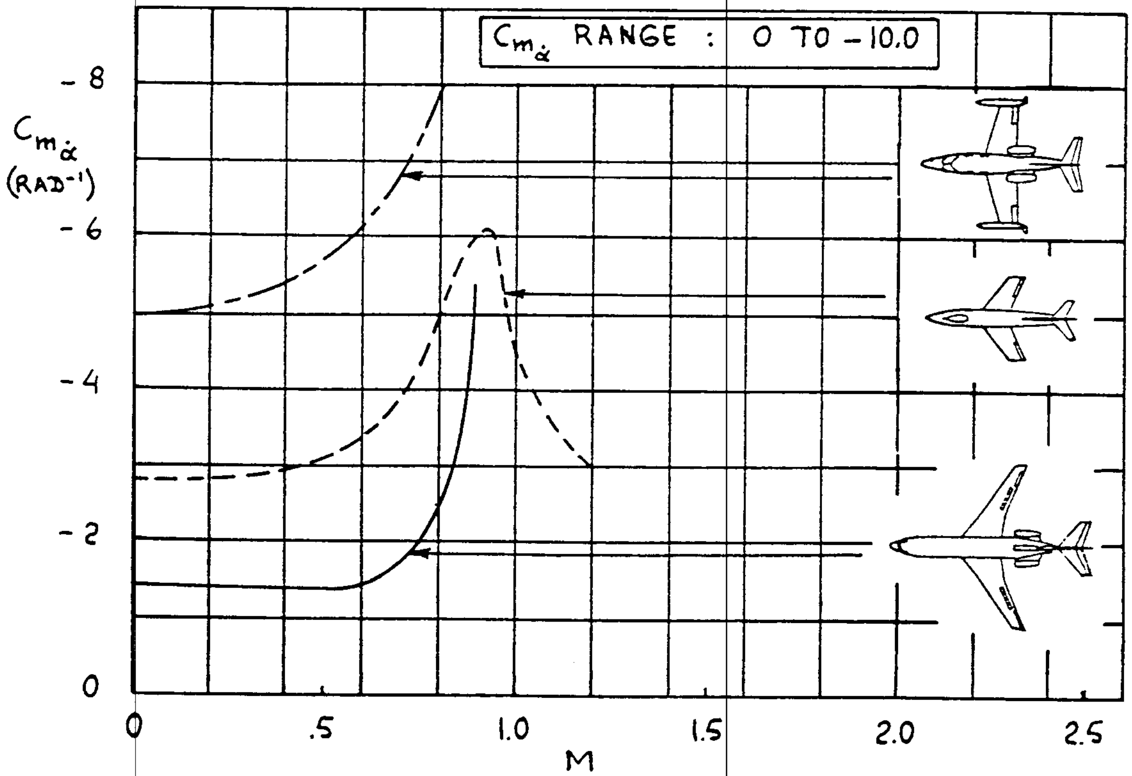
$$\frac{\partial F_{A_Z}}{\partial\left(\frac{\dot{\alpha}\bar{c}}{2U_1}\right)} = C_{z_{\dot{\alpha}}}\bar{q}_1 S = -C_{L_{\dot{\alpha}}}\bar{q}_1 S$$

$$= -2C_{L_{\alpha_H}}\eta_H \bar{V}_H \frac{d\epsilon}{d\alpha}\bar{q}_1 S$$

$$\frac{\partial M_A}{\partial\left(\frac{\dot{\alpha}\bar{c}}{2U_1}\right)} = C_{m_{\dot{\alpha}}}\bar{q}_1 S\bar{c} = -2C_{L_{\alpha_H}}\eta_H \bar{V}_H \frac{x_H}{\bar{c}}\frac{d\epsilon}{d\alpha}\bar{q}_1 S\bar{c}$$



Variation of $C_{L_{\dot\alpha}}$ with Mach Number for Typical Jet Aircraft



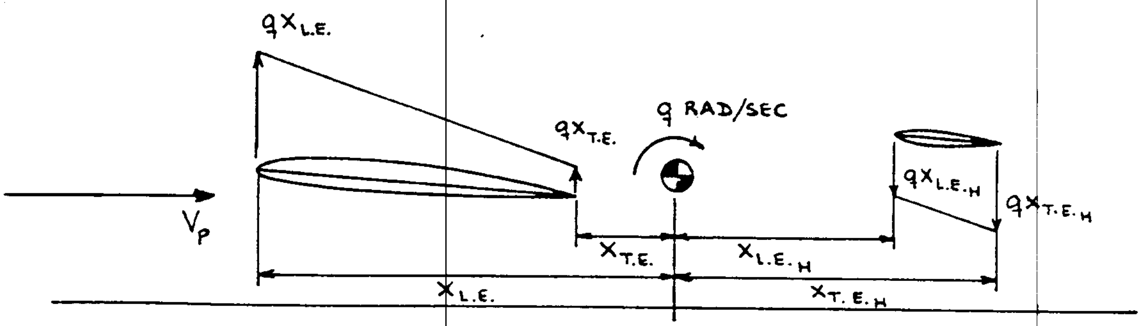
Variation of $C_{m_{\dot\alpha}}$ with Mach Number for Typical Jet Aircraft

q Derivatives (C_{L_q}, C_{D_q}, C_{m_q})

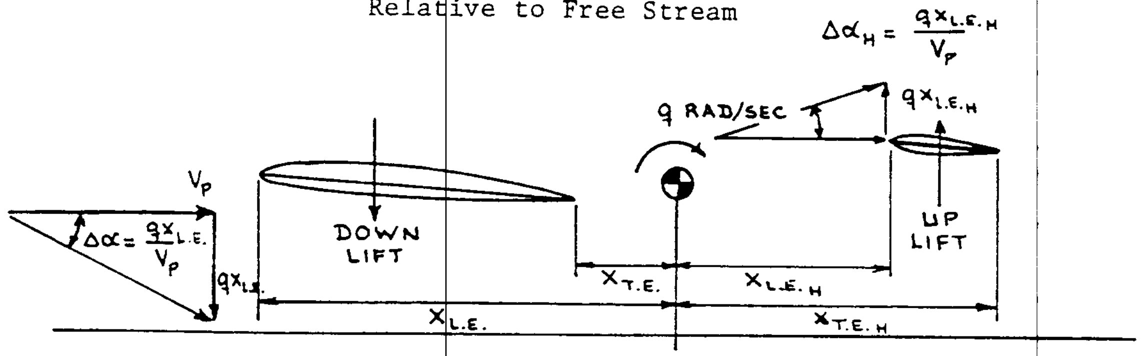
$$\frac{\partial F_{A_X}}{\partial\left(\frac{q\bar{c}}{2U_1}\right)} = C_{X_q}\bar{q}_1 S = -C_{D_q}\bar{q}_1 S = 0$$

$$\frac{\partial F_{A_Z}}{\partial\left(\frac{q\bar{c}}{2U_1}\right)} = -C_{L_q}\bar{q}_1 S \qquad \frac{\partial M_A}{\partial\left(\frac{q\bar{c}}{2U_1}\right)} = C_{m_q}\bar{q}_1 S\bar{c}$$

Surface Velocities About P
in Inertial Space



Surface Velocities About P
Relative to Free Stream



The down-going lift and the up-going lift due to pitch rate, q, result in a total lift due to pitch rate (usually positive) and in a total moment due to pitch rate (always negative)

<u>Physical Explanation of Lift and Pitching Moment due to Pitch Rate</u>

For conventional configurations, the horizontal tail contributes the most...

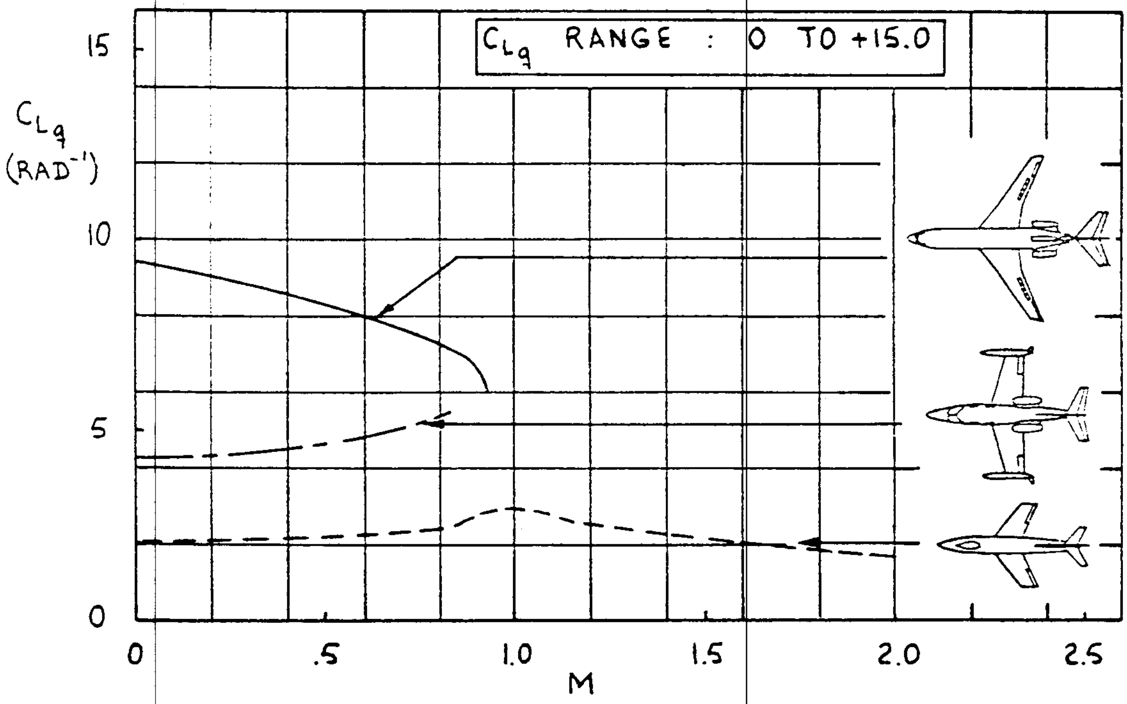
$$\Delta\alpha_H = \frac{X_H q}{U_1}$$

$$C_L)_{H_q} = C_{L\alpha_H}\left(\frac{X_H q}{U_1}\right)\eta_H \frac{S_H}{S}$$

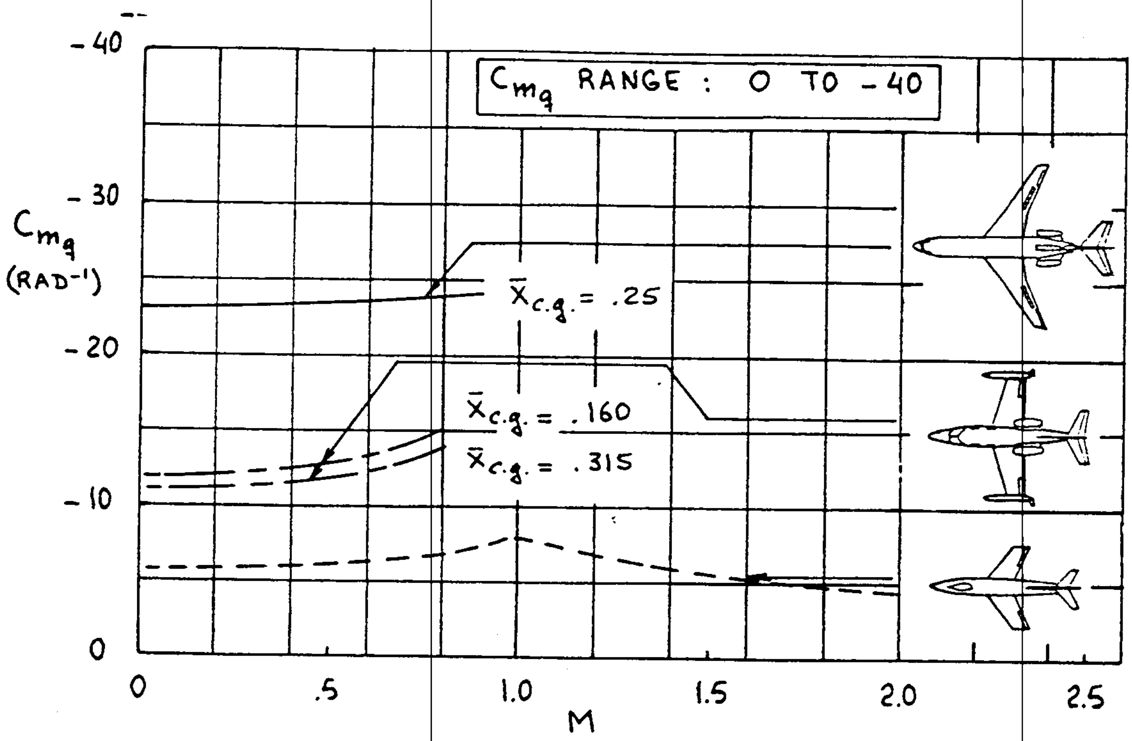
$$\Rightarrow C_{L_q} = 2C_{L\alpha_H}\,\eta_H \bar{V}_H$$

similarly $C_{m_q} = -2C_{L\alpha_H}\,\eta_H \bar{V}_H \frac{X_H}{\bar{c}}$

These are generally $\approx 10\%$ higher than estimate eqns.



Variation of C_{L_q} with Mach Number for Typical Jet Aircraft



Variation of C_{m_q} with Mach Number for Typical Jet Aircraft

$\dot{\beta}$ Derivatives ($C_{y_{\dot{\beta}}}, C_{l_{\dot{\beta}}}, C_{n_{\dot{\beta}}}$)

$$\frac{\partial F_{A_Y}}{\partial\left(\frac{\dot{\beta} b}{2U_1}\right)} = C_{y_{\dot{\beta}}} \bar{q}_1 S \qquad \frac{\partial L_A}{\partial\left(\frac{\dot{\beta} b}{2U_1}\right)} = C_{l_{\dot{\beta}}} \bar{q}_1 S b \qquad \frac{\partial N_A}{\partial\left(\frac{\dot{\beta} b}{2U_1}\right)} = C_{n_{\dot{\beta}}} \bar{q}_1 S b$$

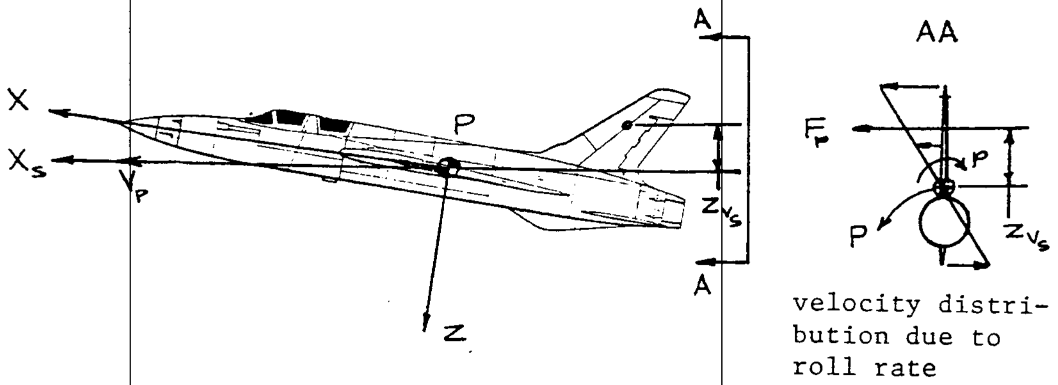
- No direct analogy to downwash lag
- Usually neglected
- Can be significant, even for conventional configurations

p Derivatives ($C_{Y_p}, C_{\ell_p}, C_{n_p}$)

$$\frac{\partial F_{A_Y}}{\partial\left(\frac{pb}{2U_1}\right)} = C_{Y_p}\bar{q}_1 S \qquad \frac{\partial L_A}{\partial\left(\frac{pb}{2U_1}\right)} = C_{\ell_p}\bar{q}_1 S b \qquad \frac{\partial N_A}{\partial\left(\frac{pb}{2U_1}\right)} = C_{n_p}\bar{q}_1 S b$$

$$C_{Y_p} = C_{Y_{p_{WBH}}} + C_{Y_{p_V}} \qquad C_{Y_{p_{WBH}}} \approx 0$$

$$C_{Y_p} = -2\,C_{L_{\alpha_V}}\frac{z_{V_s}}{b}\eta_V\frac{S_V}{S}$$



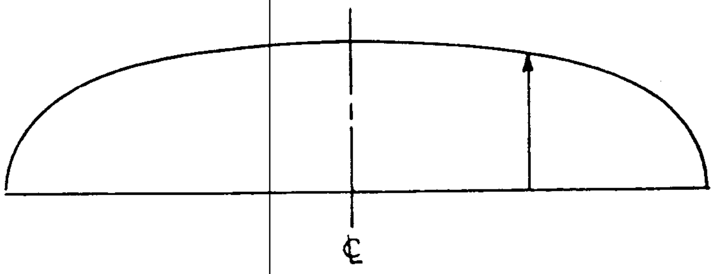
Physical Explanation of Side Force due to Roll Rate

$$C_{\ell_p} = C_{\ell_{p_{WB}}} + C_{\ell_{p_H}} + C_{\ell_{p_V}} \qquad \text{(Roll Damping)}$$

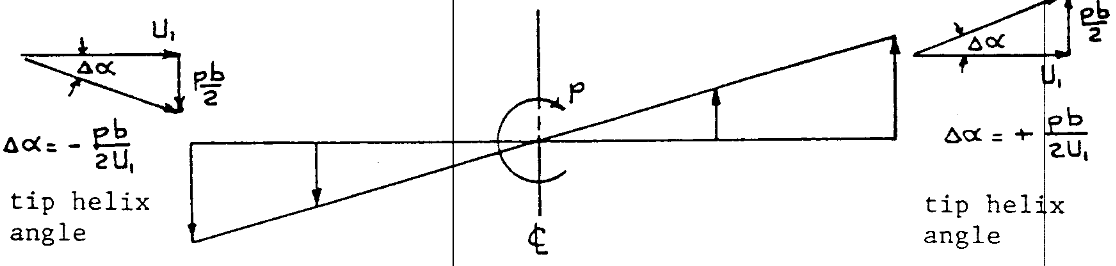
$$C_{\ell_{p_H}} = C_{\ell_{p_H}}\Big)_{\text{REF'D TO HOR. TAIL}}\frac{S_H b_H^2}{S\, b^2}$$

$$C_{\ell_{p_V}} = -2\,C_{L_{\alpha_V}}\left(\frac{z_{V_s}}{b}\right)^2\eta_V\frac{S_V}{S}$$

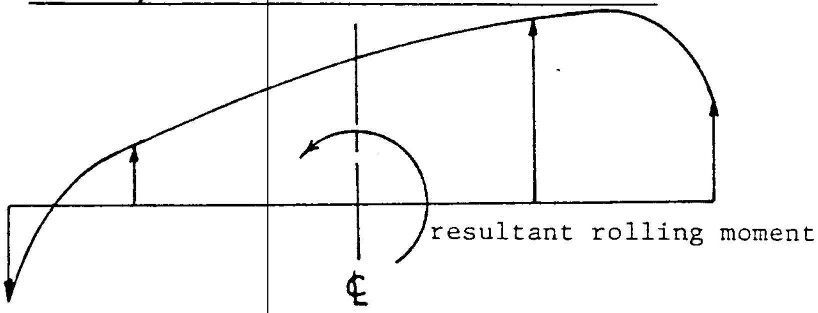
- $C_{\ell_{p_{WB}}}$ and $C_{\ell_{p_H}}$ are determined from plots found in DATCOM and other sources



Normal Lift Distribution

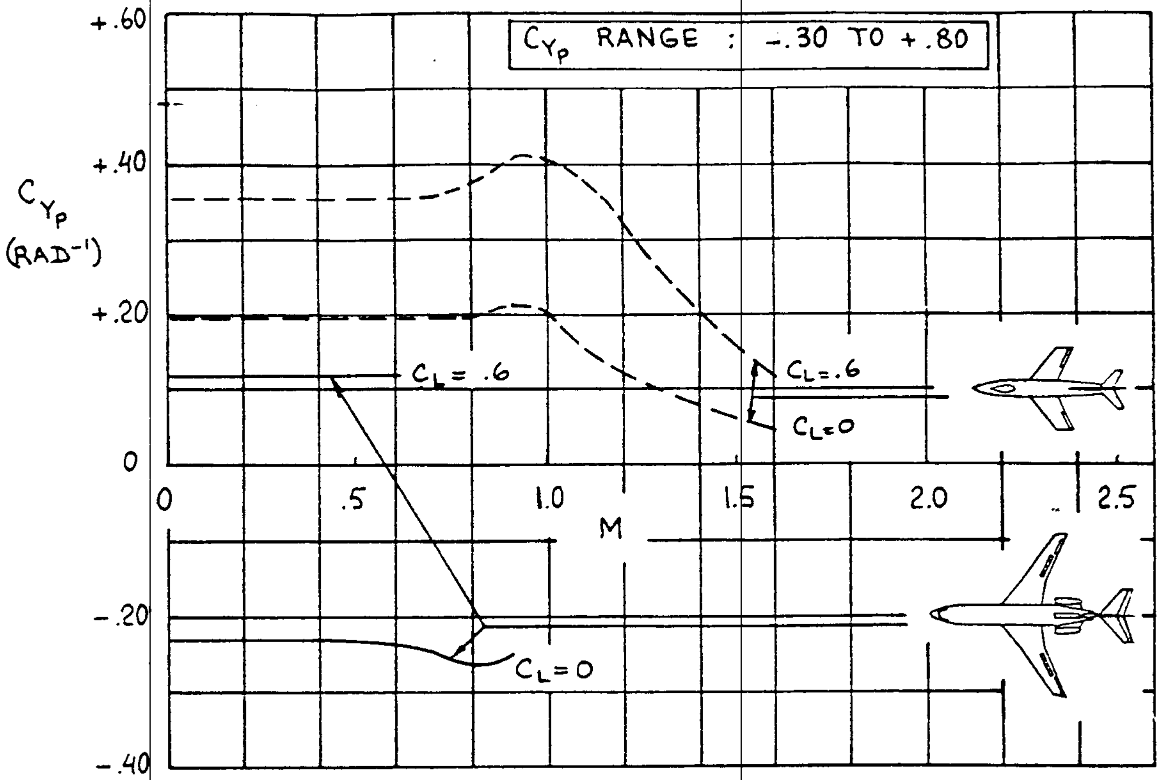


Velocity Distribution due to Roll Rate

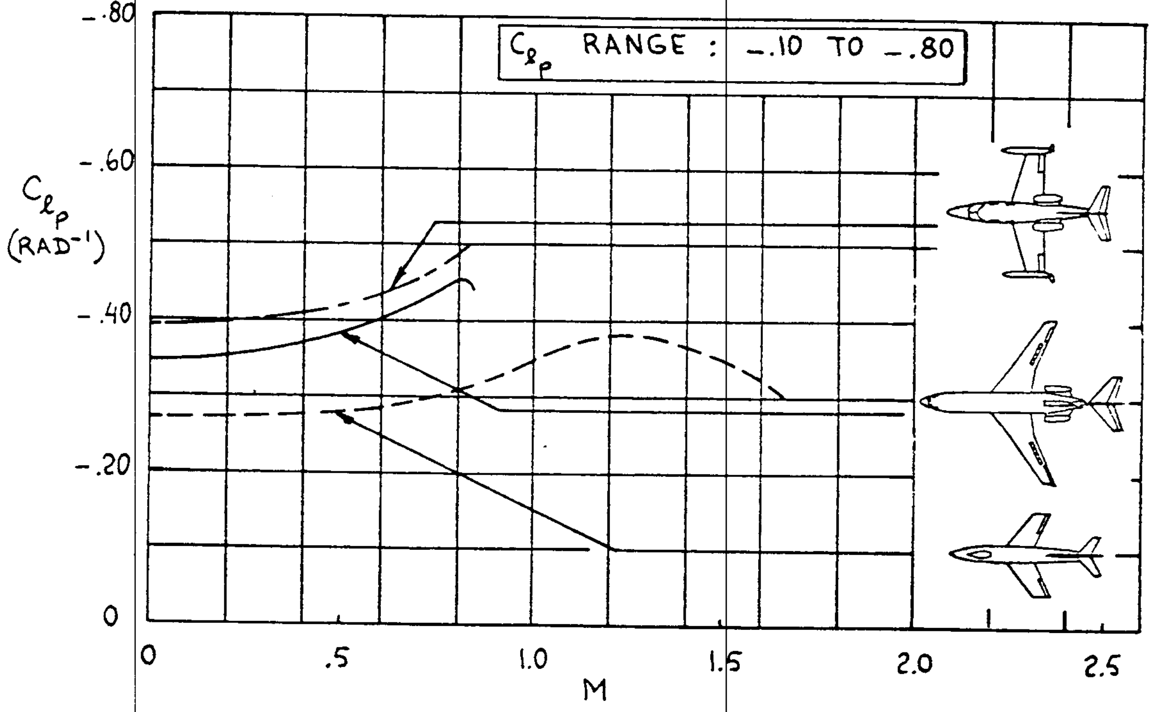


Total Lift Distribution in a Steady
State Roll

Physical Explanation of Rolling Moment due to Roll Rate



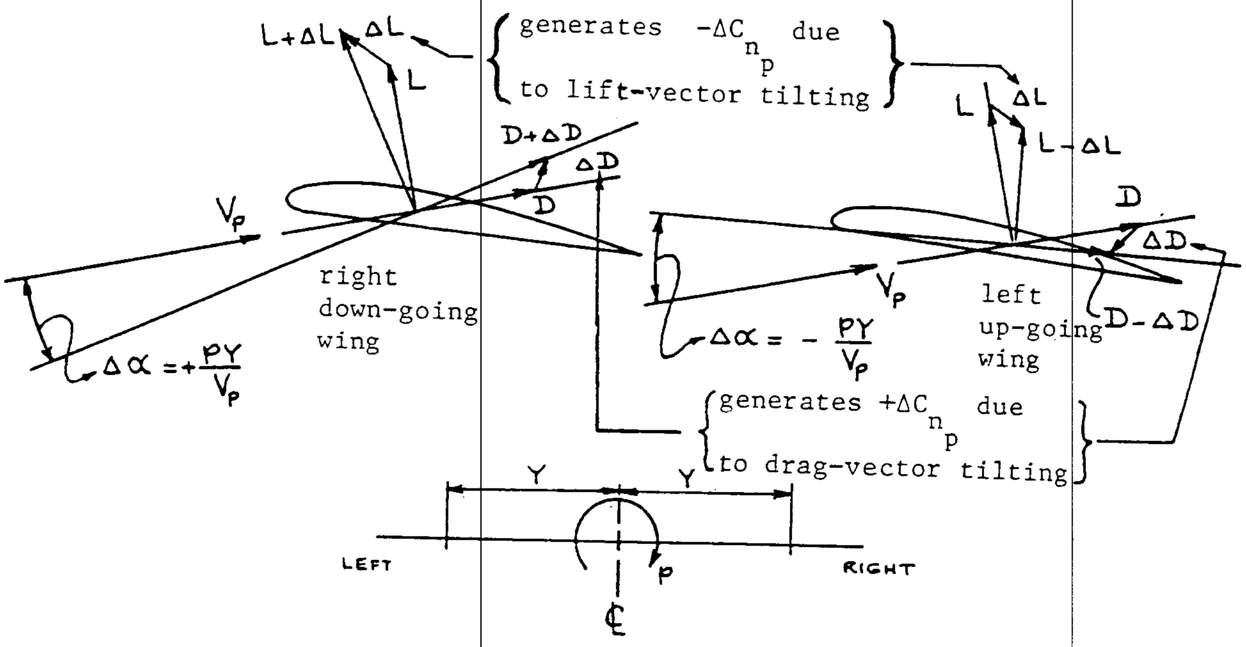
Variation of C_{y_p} with Mach Number for Typical Jet Aircraft



Variation of C_{ℓ_p} with Mach Number for Typical Jet Aircraft

$$C_{n_p} = C_{n_{p_{WB}}} + C_{n_{p_V}}$$

- Horizontal tail generally neglected
- $C_{n_{p_{WB}}}$ can be generated by:
  1. Profile drag effect
  2. Lift vector tilting
  3. Tip suction

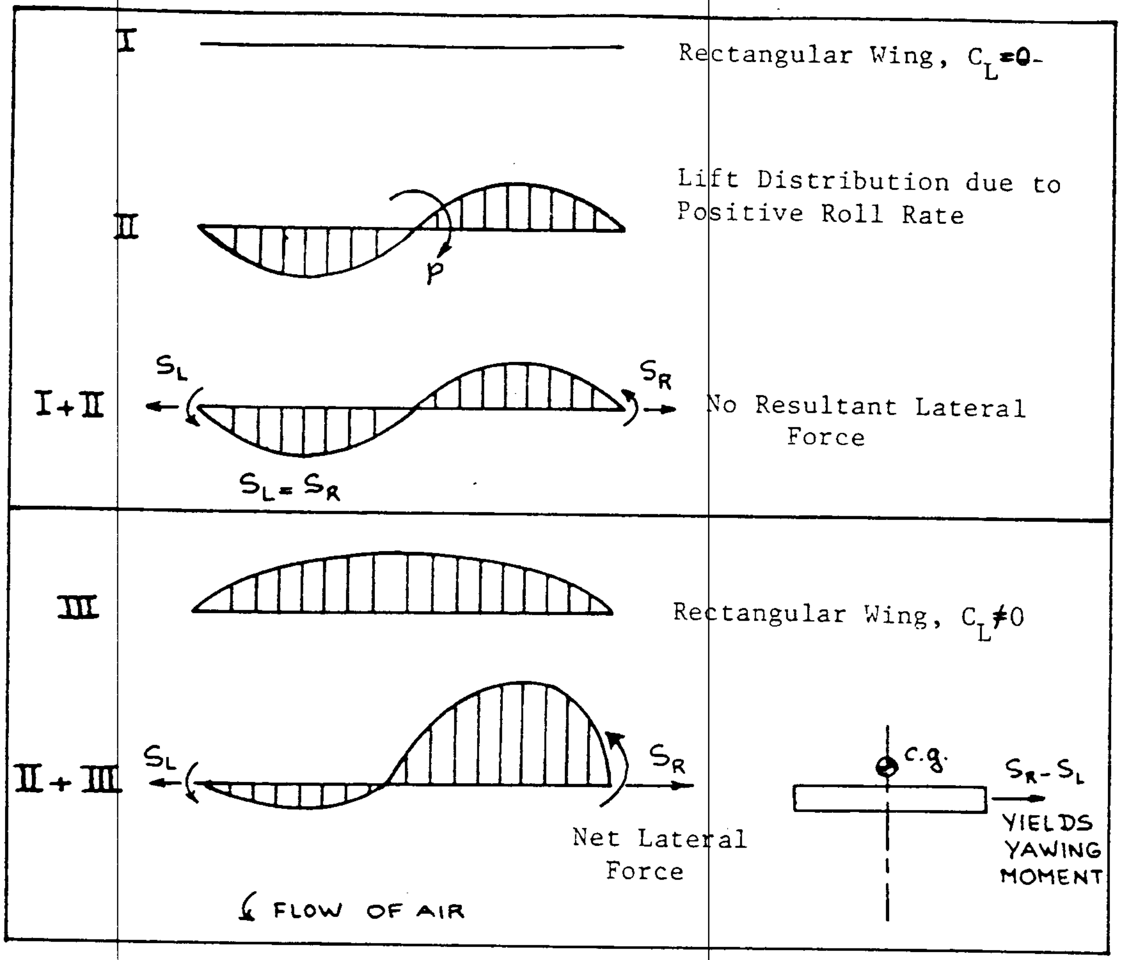


Physical Explanation of the Effect of Lift and Drag Vector Tilting in Steady Rolls on Yawing Moment

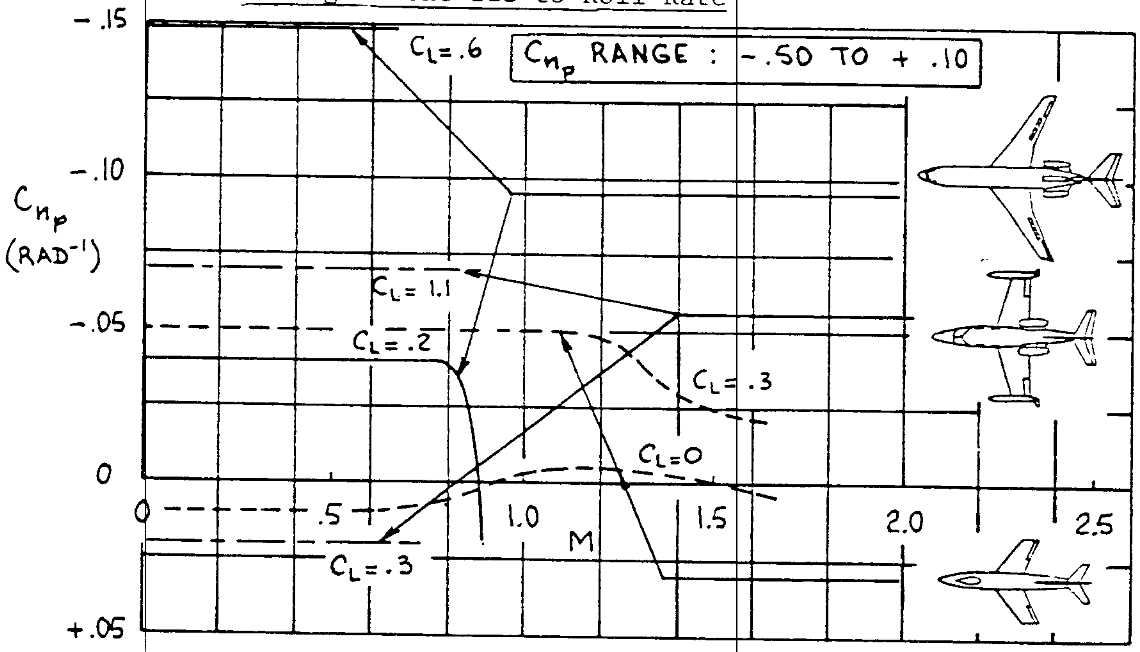
see next page for tip suction diagram

$$C_{n_{p_V}} = 2C_{L_{\alpha_V}} \frac{z_{VS}}{b} \frac{x_{VS}}{b} \eta_V \frac{S_V}{S}$$

- Estimation methods are provided in the cited literature
- Yawing moment due to roll is difficult to estimate.



Physical Explanation of the Effect of Tip Suction on Yawing Moment due to Roll Rate



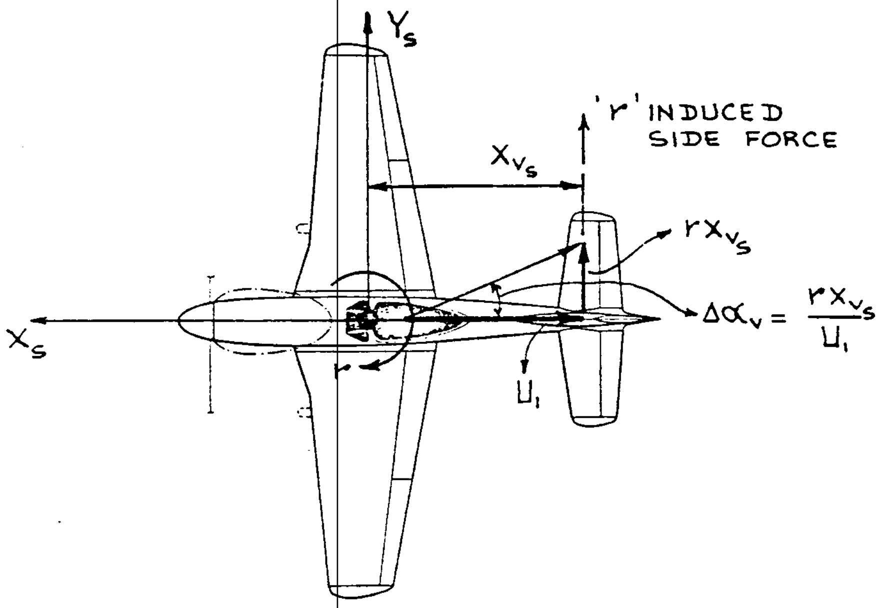
Variation of C_{n_p} with Mach Number for Typical Jet Aircraft

r Derivatives $(C_{y_r}, C_{l_r}, C_{n_r})$

$$\frac{\partial F_{A_Y}}{\partial\left(\frac{rb}{2U_1}\right)} = \bar{C}_{y_r}\bar{q}_1 S \qquad \frac{\partial L_A}{\partial\left(\frac{rb}{2U_1}\right)} = C_{l_r}\bar{q}_1 S b \qquad \frac{\partial N_A}{\partial\left(\frac{rb}{2U_1}\right)} = C_{n_r}\bar{q}_1 S b$$

$$C_{y_r} = C_{y_{r_{WBH}}} + C_{y_{r_V}} \qquad C_{y_{r_V}} >> C_{y_{r_{WBH}}}$$

$$C_{y_{r_V}} = C_{L_{\alpha_V}} \frac{2X_{V_S}}{b} \eta_V \frac{S_V}{S}$$

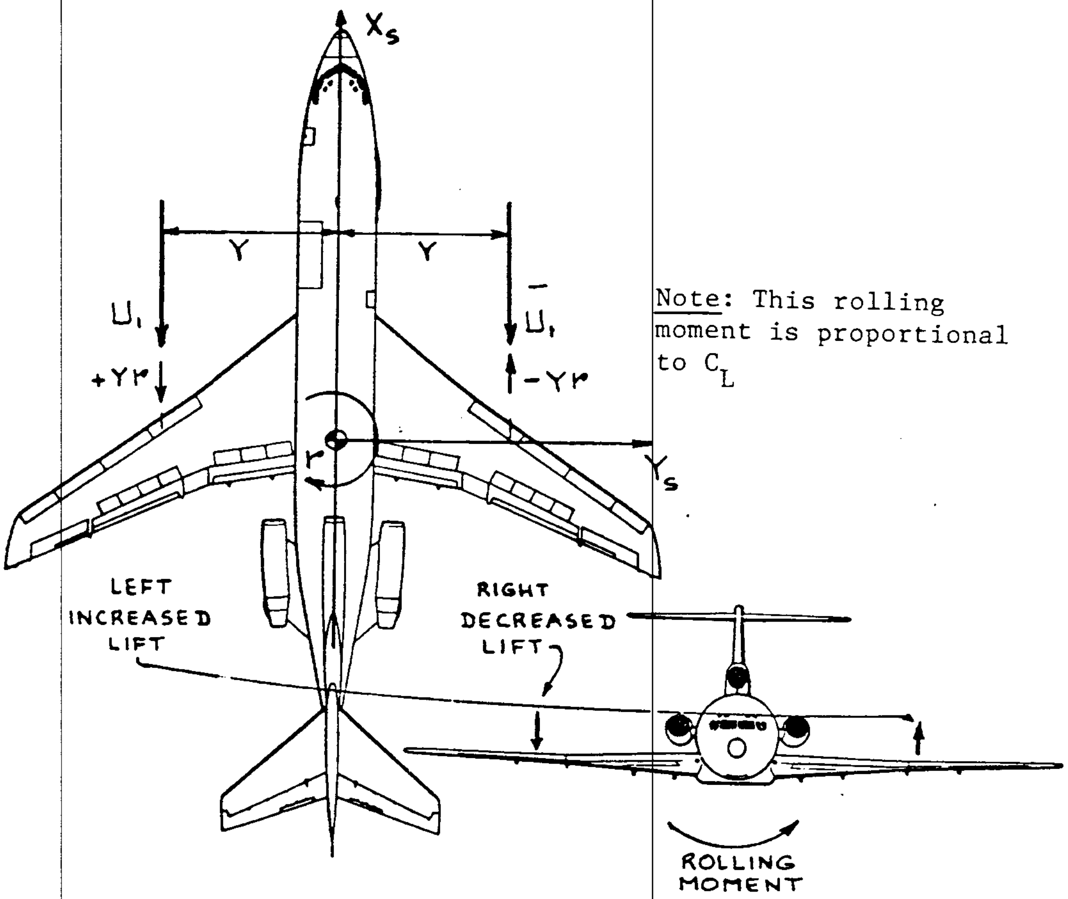


Physical Explanation of Side Force due to Yaw Rate

$$C_{l_r} = C_{l_{r_{WB}}} + C_{l_{r_H}} + C_{l_{r_V}}$$

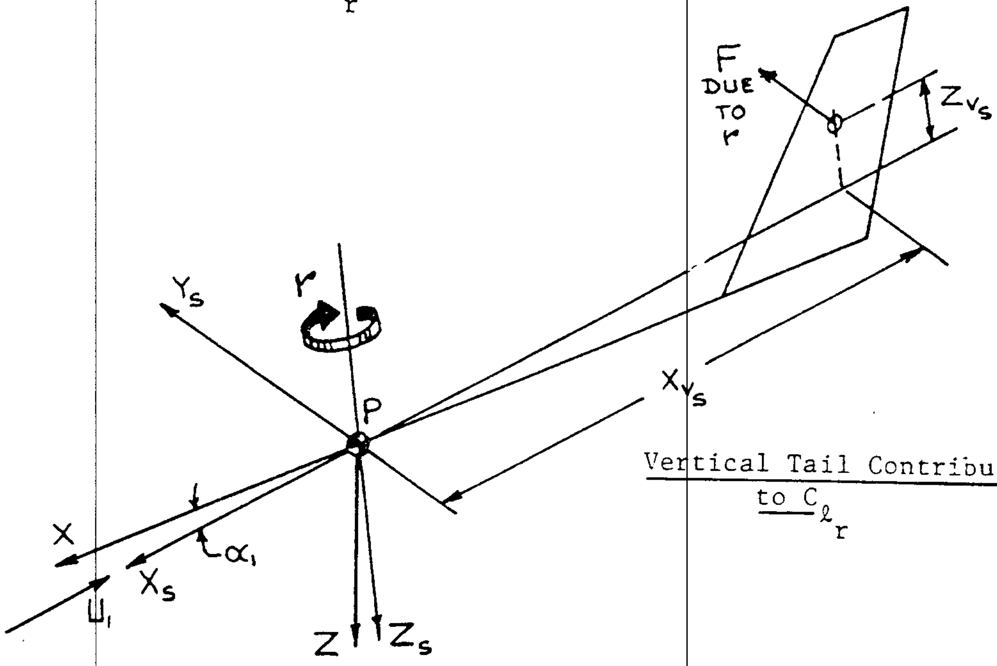
- $C_{l_{r_H}}$ is neglected
- $C_{l_{r_{WB}}}$ can be found in cited literature

$$C_{l_{r_V}} = C_{L_{\alpha_V}} \frac{2X_{V_S} Z_{V_S}}{b^2} \eta_V \frac{S_V}{S}$$



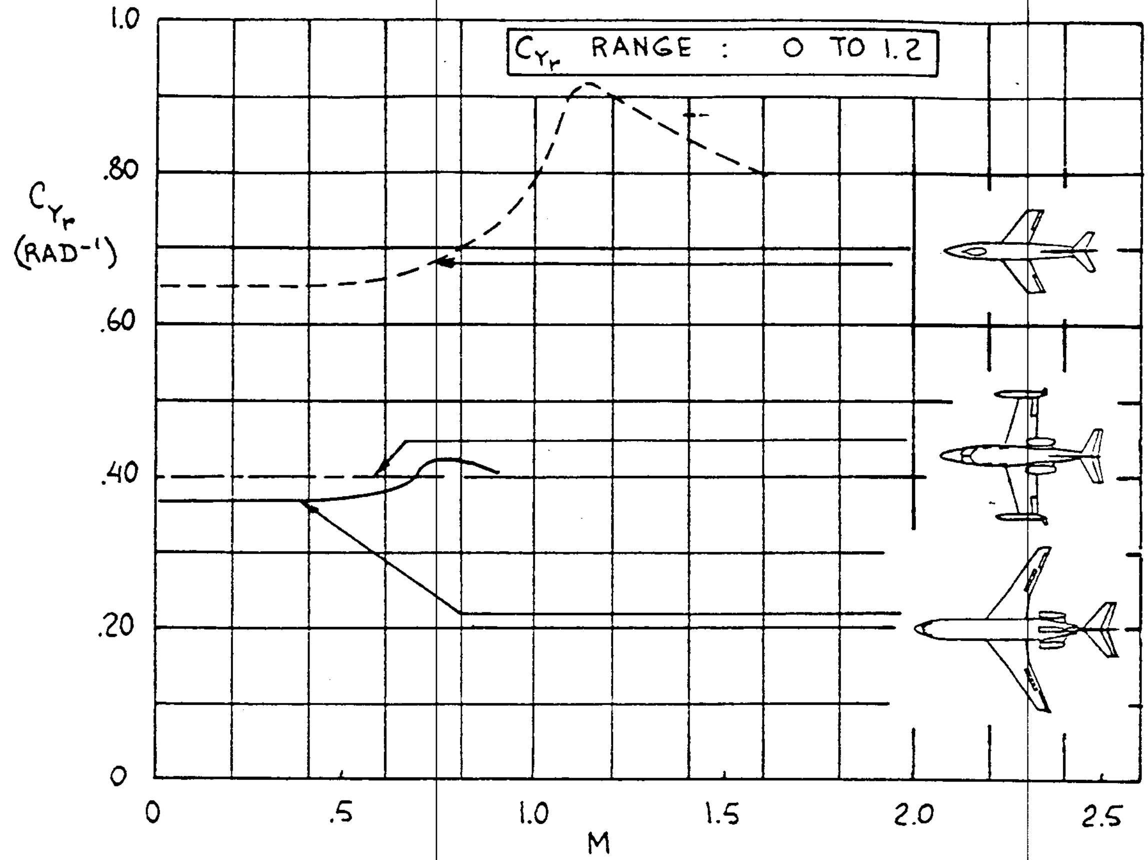
Note: This rolling moment is proportional to C_L

Wing-Body Contribution to C_{ℓ_r}

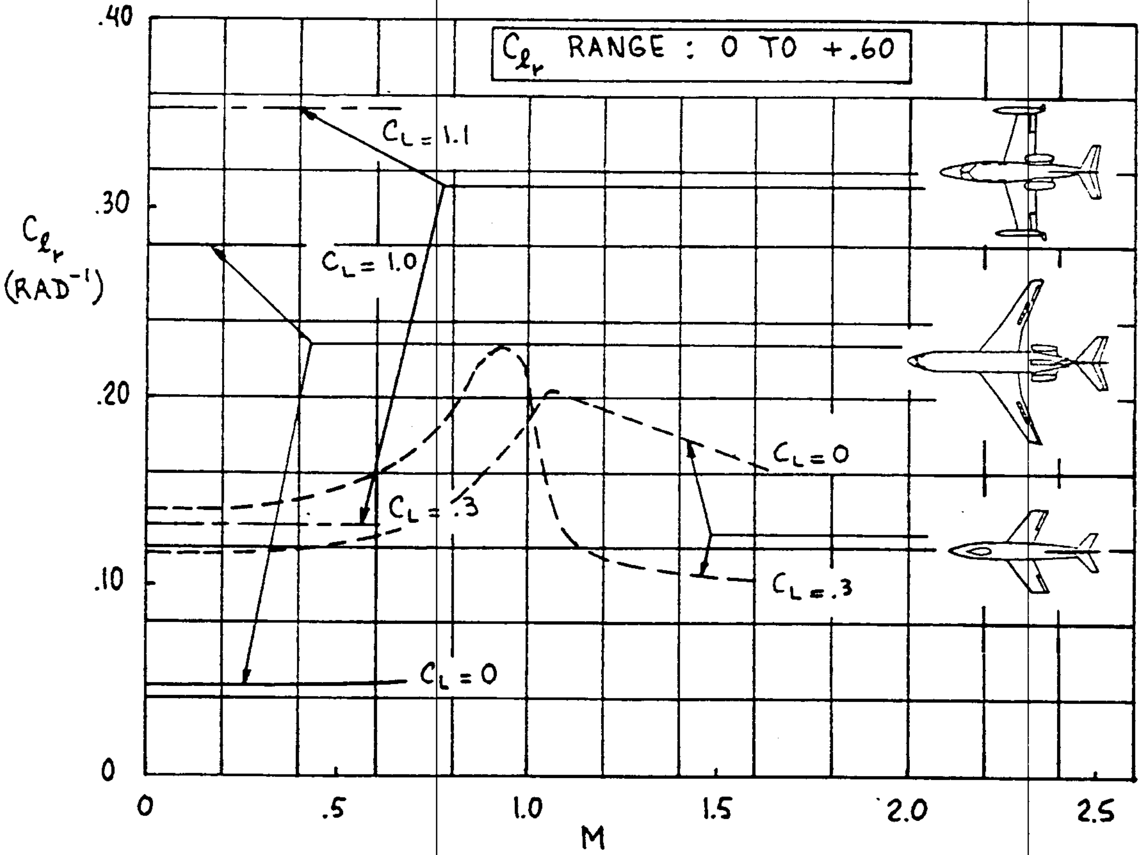


Vertical Tail Contribution to C_{ℓ_r}

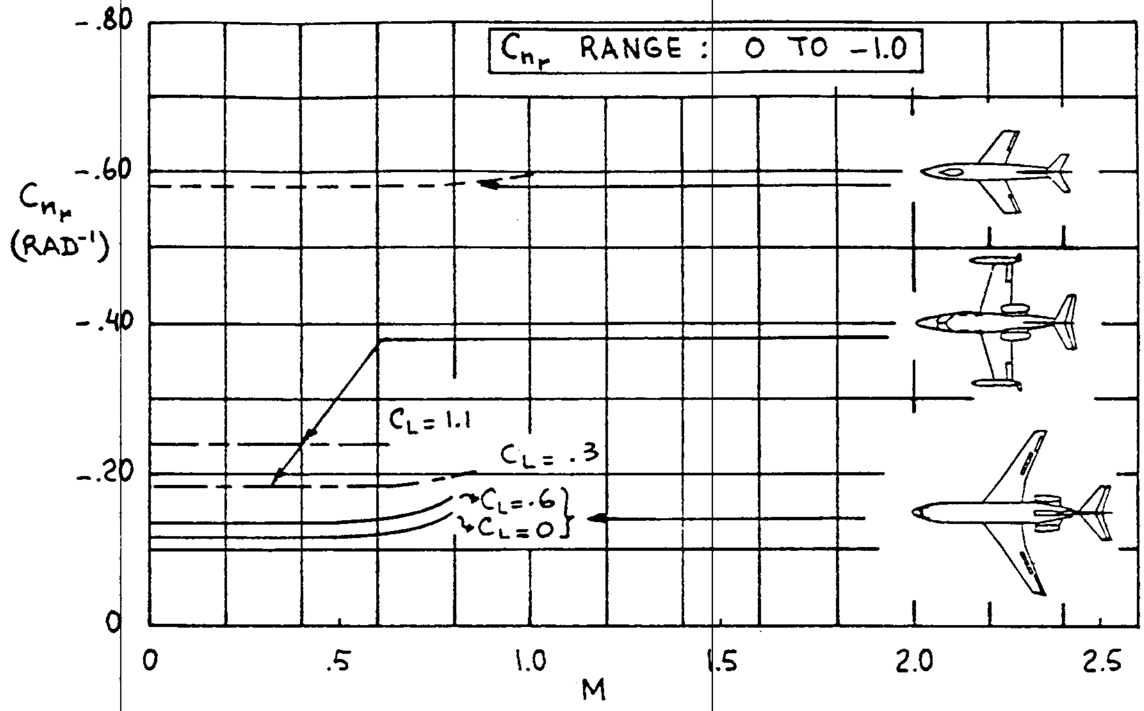
Physical Explanation for Wing-Body and Vertical Tail Contributions to Rolling Moment due to Yaw Rate



Variation of C_{y_r} with Mach Number for Typical Jet Aircraft



Variation of C_{ℓ_r} with Mach Number for Typical Jet Aircraft



Variation of C_{n_r} with Mach Number for Typical Jet Aircraft

$$\left.\begin{array}{l}\text{Nondimensional}\\ \text{Motion}\\ \text{Variables}\end{array}\right\} \quad \frac{u}{U_1} \qquad \alpha \qquad \frac{\dot{\alpha}\bar{c}}{2U_1} \qquad \frac{q\bar{c}}{2U_1} \qquad \delta_E$$

Forces and Moments

$$\underset{(3\text{x}1)}{\begin{Bmatrix} \dfrac{f_{A_x}}{\bar{q}S} \\[2ex] \dfrac{f_{A_z}}{\bar{q}S} \\[2ex] \dfrac{m_A}{\bar{q}S} \end{Bmatrix}} = \underset{(3\text{x}5)}{\begin{bmatrix} -(C_{D_u}+2C_{D_1}) & (-C_{D_\alpha}+C_{L_1}) & -C_{D_{\dot{\alpha}}} & -C_{D_q} & -C_{D_{\delta_E}} \\ -(C_{L_u}+2C_{L_1}) & (-C_{L_\alpha}-C_{D_1}) & -C_{L_{\dot{\alpha}}} & -C_{L_q} & -C_{L_{\delta_E}} \\ (C_{m_u}+2C_{m_1}) & C_{m_\alpha} & C_{m_{\dot{\alpha}}} & C_{m_q} & C_{m_{\delta_E}} \end{bmatrix}} \underset{(5\text{x}1)}{\begin{Bmatrix} \dfrac{u}{U_1} \\ \alpha \\ \dfrac{\dot{\alpha}\bar{c}}{2U_1} \\ \dfrac{q\bar{c}}{2U_1} \\ \delta_E \end{Bmatrix}}$$

Summary of Perturbed Aerodynamic Forces and Moments Using Stability and Control Derivatives

Summary of Perturbed Aerodynamic Forces and Moments Using Stability and Control Derivatives

Nondimensional Motion Variables $\left.\right\}$ $\beta \quad \frac{\dot{\beta} b}{2U_1} \quad \frac{pb}{2U_1} \quad \frac{rb}{2U_1} \quad \delta_A \quad \delta_R$

Forces and Moments

$$
\begin{Bmatrix} \frac{f_{A_y}}{\bar{q}S} \\ \frac{\ell_A}{\bar{q}Sb} \\ \frac{n_A}{\bar{q}Sb} \end{Bmatrix}
=
\begin{bmatrix}
C_{y_\beta} & C_{y_{\dot{\beta}}} & C_{y_p} & C_{y_r} & C_{y_{\delta_A}} & C_{y_{\delta_R}} \\
C_{\ell_\beta} & C_{\ell_{\dot{\beta}}} & C_{\ell_p} & C_{\ell_r} & C_{\ell_{\delta_A}} & C_{\ell_{\delta_R}} \\
C_{n_\beta} & C_{n_{\dot{\beta}}} & C_{n_p} & C_{n_r} & C_{n_{\delta_A}} & C_{n_{\delta_R}}
\end{bmatrix}
\begin{Bmatrix} \beta \\ \frac{\dot{\beta} b}{2U_1} \\ \frac{pb}{2U_1} \\ \frac{rb}{2U_1} \\ \delta_A \\ \delta_R \end{Bmatrix}
$$

(3x1) (3x6) (6x1)

The Longitudinal Characteristic Equation

$$\frac{u(s)}{\delta_E(s)} = \frac{\begin{vmatrix} X_{\delta_E} & -X_\alpha & g\cos\Theta_1 \\ Z_{\delta_E} & \{s(U_1 - Z_{\dot\alpha}) - Z_\alpha\} & \{-(Z_q + U_1)s + g\sin\Theta_1\} \\ M_{\delta_E} & -\{M_{\dot\alpha}s + M_\alpha + M_{T_\alpha}\} & (s^2 - M_q s) \end{vmatrix}}{\begin{vmatrix} s - X_u - X_{T_u} & -X_\alpha & g\cos\Theta_1 \\ -Z_u & \{s(U_1 - Z_{\dot\alpha}) - Z_\alpha\} & \{-(Z_q + U_1)s + g\sin\Theta_1\} \\ -(M_u + M_{T_u}) & -\{(M_{\dot\alpha}s + M_\alpha + M_{T_\alpha}\} & (s^2 - M_q s) \end{vmatrix}} = \frac{N_u}{D_1}$$

$$\boxed{D_1 = As^4 + Bs^3 + Cs^2 + Ds + E}$$

$$A = U_1 - Z_{\dot\alpha}$$

$$B = -(U_1 - Z_{\dot\alpha})\{X_u + X_{T_u} + M_q\} - Z_\alpha - M_{\dot\alpha}(U_1 + Z_q)$$

$$C = (X_u + X_{T_u})[M_q(U_1 - Z_{\dot\alpha}) + Z_\alpha + M_{\dot\alpha}(U_1 + Z_q)] + M_q Z_\alpha - Z_u X_\alpha + + M_{\dot\alpha} g\sin\Theta_1 - (M_\alpha + M_{T_\alpha})(U_1 + Z_q)$$

$$D = g\sin\Theta_1[M_\alpha + M_{T_\alpha} - M_{\dot\alpha}(X_u + X_{T_u})] + g\cos\Theta_1[Z_u M_{\dot\alpha} + (M_u + M_{T_u})(U_1 - Z_{\dot\alpha})] + (M_u + M_{T_u})[-X_\alpha(U_1 + Z_q)] + Z_u X_\alpha M_q + (X_u + X_{T_u})[(M_\alpha + M_{T_\alpha})(U_1 + Z_q) + - M_q Z_\alpha]$$

$$E = g\cos\Theta_1\,[(M_\alpha + M_{T_\alpha})Z_u - Z_\alpha(M_u + M_{T_u})] + g\sin\Theta_1[(M_u + M_{T_u})X_\alpha + - (X_u + X_{T_u})(M_\alpha + M_{T_\alpha})]$$

- D_1 is a common denominator for $\frac{u(s)}{\delta_E(s)}, \frac{\alpha(s)}{\delta_E(s)}, \frac{\theta(s)}{\delta_E(s)}$

-- The form of the characteristic equation leads to the conclusion (for typical aircraft coefficient values) that there are two oscillatory longitudinal modes:

1. Short period - highly damped, high frequency, basically constant speed

2. Phugoid - lightly damped, low frequency, basically constant angle of attack

Approximated by:

$$\omega_{n_{SP}} = \sqrt{\frac{Z_\alpha M_q}{U_1} - M_\alpha}$$

$$\zeta_{SP} = \frac{-\left(M_q + \frac{Z_\alpha}{U_1} + M_{\dot\alpha}\right)}{2\,\omega_{n_{SP}}} \doteq \sqrt{-M_\alpha} = \sqrt{\frac{-C_{m_\alpha}\bar{q}_1 S \bar{c}}{I_{yy}}}$$

Divergence of short period when

$$\frac{dC_m}{dC_L} = -\frac{C_{m_q}\rho S \bar{c} g}{4W}$$

or $\quad \bar{X}_{cg} = \bar{X}_{ac} - \dfrac{C_{m_q}\rho S\bar{c} g}{4W}$

this is the maneuver point!

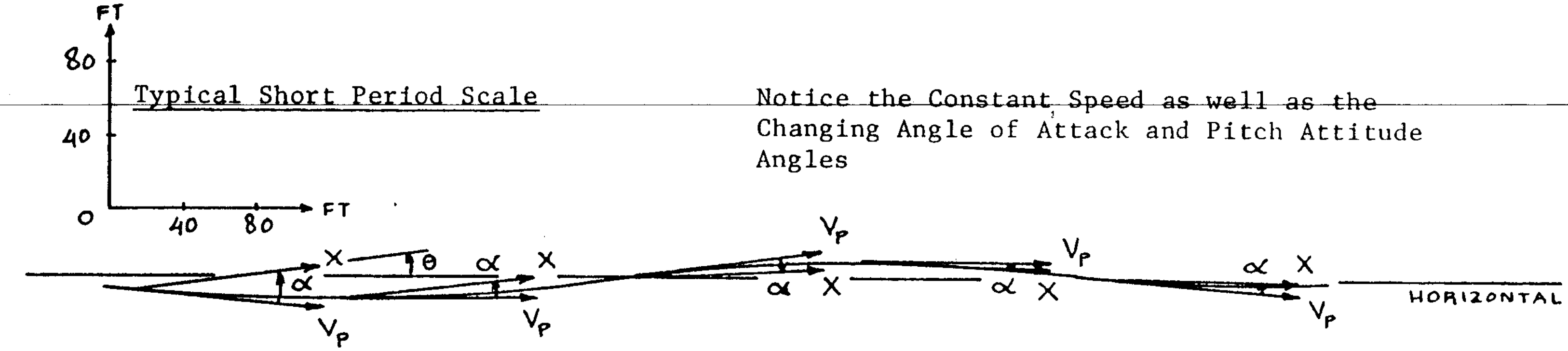
Phugoid $\quad \omega_{n_P} = \sqrt{\dfrac{-Z_u g}{U_1}}$

$$\zeta_P = \frac{-X_u}{2\,\omega_{n_P}} = \frac{\bar{q}_1 S}{2mU_1\omega_{n_P}}\left(C_{D_u} + 2C_{D_1} - C_{T_{x_u}} - 2C_{T_{x_1}}\right)$$

- This is a poor approximation for damping ratio
- Note dependence on propulsion characteristics
- Undamped natural frequency is inversely proportional to forward velocity

Typical Short Period Scale

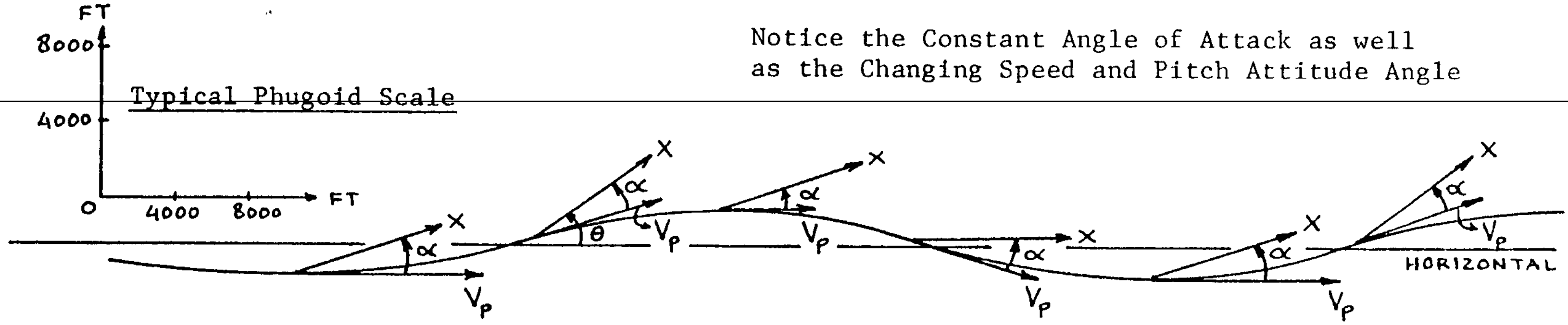
Notice the Constant Speed as well as the Changing Angle of Attack and Pitch Attitude Angles



Short Period as Seen by an Outside Observer

Typical Phugoid Scale

Notice the Constant Angle of Attack as well as the Changing Speed and Pitch Attitude Angle



Phugoid as Seen by an Outside Observer

The Lateral-Directional Characteristic Equation

$$\frac{\beta(s)}{\delta(s)} = \frac{\begin{vmatrix} Y_\delta & -(sY_p + g\cos\Theta_1) & s(U_1 - Y_r) \\ L_\delta & s^2 - L_p s & -(s^2 A_1 + sL_r) \\ N_\delta & -(s^2 B_1 + N_p s) & (s^2 - sN_r) \end{vmatrix}}{\begin{vmatrix} (sU_1 - Y_\beta) & -(sY_p + g\cos\Theta_1) & s(U_1 - Y_r) \\ -L_\beta & s^2 - L_p s & -(s^2 A_1 + sL_r) \\ -N_\beta - N_{T_\beta} & -(s^2 B_1 + N_p s) & s^2 - sN_r \end{vmatrix}} = \frac{N_\beta}{D_2}$$

$$\boxed{D_2 = s(As^4 + Bs^3 + Cs^2 + Ds + E)}$$

$$A = U_1(1 - A_1 B_1)$$

$$B = -Y_\beta(1 - A_1B_1) - U_1(L_p + N_r + A_1N_p + B_1L_r)$$

$$C = U_1(L_pN_r - L_rN_p) + Y_\beta(N_r + L_p + A_1N_p + B_1L_r) - Y_p(L_\beta + N_\beta A_1 + N_{T_\beta}A_1) + \\ + U_1(L_\beta B_1 + N_\beta + N_{T_\beta}) - Y_r(L_\beta B_1 + N_\beta + N_{T_\beta})$$

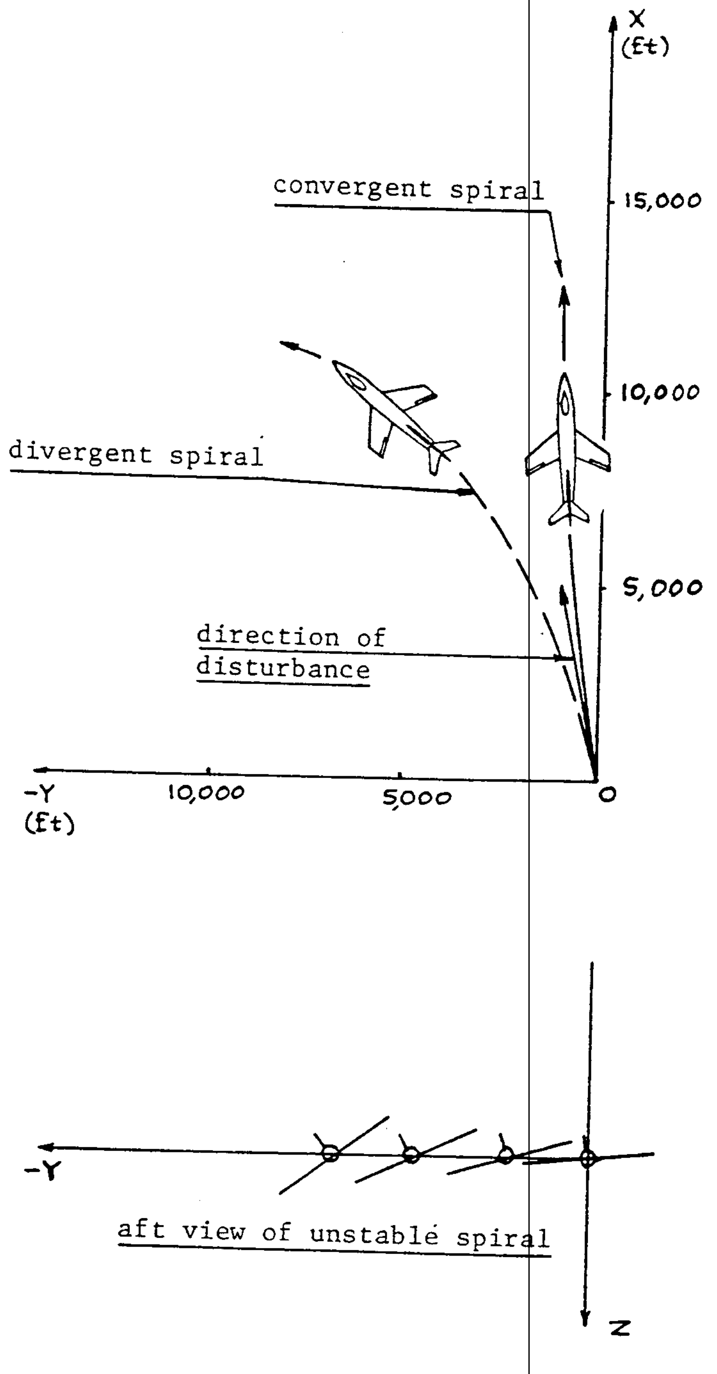
$$D = -Y_\beta(L_pN_r - L_rN_p) + Y_p(L_\beta N_r - N_\beta L_r - N_{T_\beta}L_r) - g\cos\theta_1(L_\beta + N_\beta A_1 + N_{T_\beta}A_1) + \\ + U_1(L_\beta N_p - N_\beta L_p - N_{T_\beta}L_p) - Y_r(L_\beta N_p - N_\beta L_p - N_{T_\beta}L_p)$$

$$E = g\cos\Theta_1(L_\beta N_r - N_\beta L_r - N_{T_\beta}L_r)$$

- D_2 is a common denominator for $\frac{\beta(s)}{\delta(s)}, \frac{\phi(s)}{\delta(s)}, \frac{\psi(s)}{\delta(s)}$

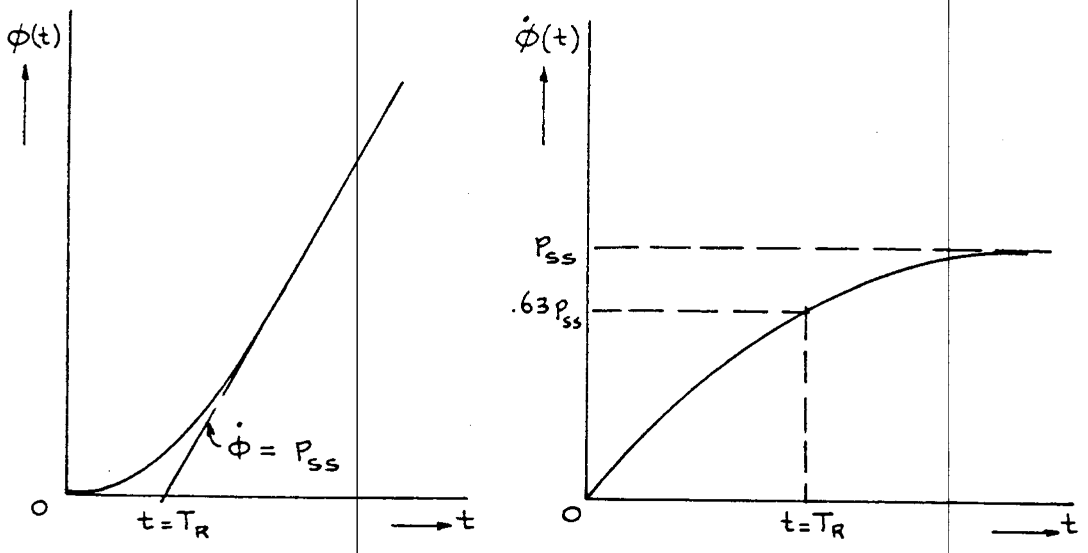
Three modes of interest:

1. Spiral mode
2. 1 DOF roll response
3. Dutch roll

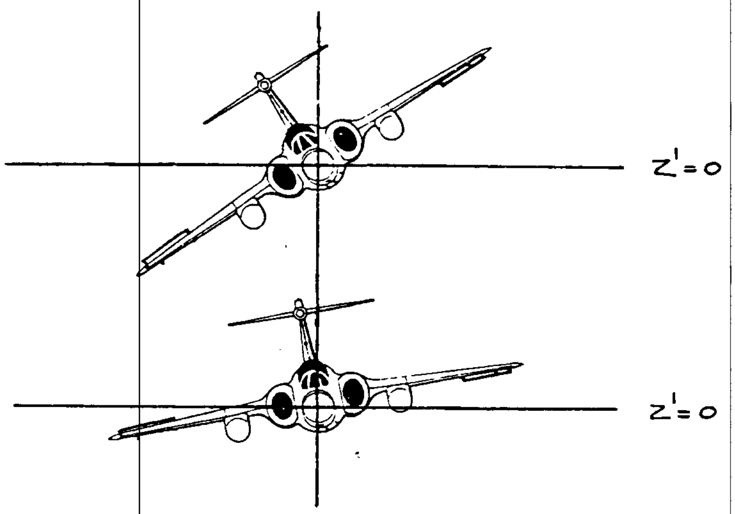


Spiral Mode as Seen by an Outside Observer

- Spiral mode is best analyzed by full equations



Typical Example of Rolling Mode Time Histories



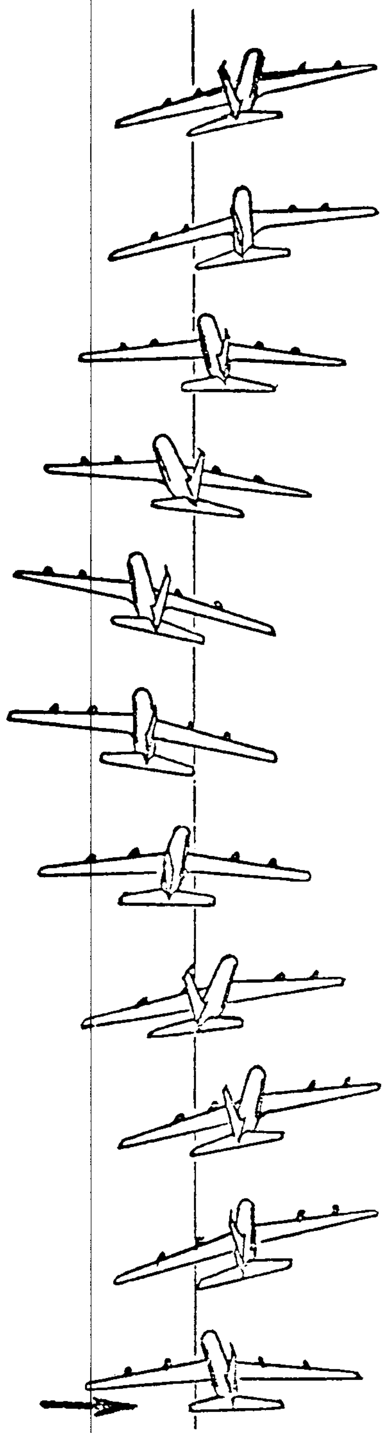
Rolling Mode as Seen by an Outside Observer

1 DOF Approximation

$$p_{ss} = \frac{-L_{\delta_A}\delta_A}{L_p} \qquad T_R = -\frac{1}{L_p}$$

$$\phi(t) = -\frac{L_{\delta_A}\delta_A}{L_p} + \frac{L_{\delta_A}\delta_A}{L_p^2}\left(e^{L_p t} - 1\right)$$

step response from $\phi = \text{constant} = 0$



Dutch Roll Mode as Seen by an Outside Observer

Dutch Roll Approximation

$$\omega_{n_D} = \sqrt{\frac{1}{U_1}\left(Y_\beta N_r + N_\beta U_1 - N_\beta Y_r\right)}$$

$$\zeta_D = \frac{-1}{2\omega_{n_D}}\left(N_r + \frac{Y_\beta}{U_1}\right)$$

- Dutch roll damping poorly estimated
- Swept wing aircraft (large C_{l_β}) and sizable I_{xz} ~~invalid~~ both invalidate application of the Dutch Roll approximation.
- Dutch roll characteristics must be obtained from full six-DOF model (a good 3-DOF model may be OK)

Certification of Gulfstream Class Aircraft

- FAR Part 25 and JAR 25 applies
- Attached are excerpts:
  - FAR Part 25
  - JAR 25
  - FAA Advisory Circular AC-25-7, "Flight Test Guide for Certification of Transport Category Airplanes"

- Regulatory environment includes other FAA paper, including Special Conditions, recent changes and updates, plus requirements levied for foriegn certification.

(3) The w   t used when V_S is being used as a f   r to determine compliance with a required performance standard; and

(4) The most unfavorable center of gravity allowable.

(b) The stalling speed V_S is the minimum speed obtained as follows:

(1) Trim the airplane for straight flight at any speed not less than 1.2 V_S or more than 1.4 V_S. At a speed sufficiently above the stall speed to ensure steady conditions, apply the elevator control at a rate so that the airplane speed reduction does not exceed one knot per second.

(2) Meet the flight characteristics provisions of § 25.203.

§ 25.105 Takeoff.

(a) The takeoff speeds described in § 25.107, the accelerate-stop distance described in § 25.109, the takeoff path described in § 25.111, and the takeoff distance and takeoff run described in § 25.113, must be determined—

(1) At each weight, altitude, and ambient temperature within the operational limits selected by the applicant; and

(2) In the selected configuration for takeoff.

(b) No takeoff made to determine the data required by this section may require exceptional piloting skill or alertness.

(c) The takeoff data must be based on—

(1) A smooth, dry, hard-surfaced runway, in the case of land planes and amphibians;

(2) Smooth water, in the case of seaplanes and amphibians; and

(3) Smooth, dry snow, in the case of skiplanes.

(d) The takeoff data must include, within the established operational limits of the airplane, the following operational correction factors:

(1) Not more than 50 percent of nominal wind components along the takeoff path opposite to the direction of takeoff, and not less than 150 percent of nominal wind components along the takeoff path in the direction of takeoff.

(2) Effective runway gradients.

§ 25.107 Takeoff speeds.

(a) V_1 must be established in relation to V_{EF} as follows:

(1) V_{EF} is the calibrated airspeed at which the critical engine is assumed to fail. V_{EF} must be selected by the applicant, but may not be less than V_{MCG} determined under § 25.149(e).

(2) V_1, in terms of calibrated airspeed, is the takeoff decision speed selected by the applicant; however, V_1 may not be less than V_{EF} plus the speed gained with the critical engine inoperative during the time interval between the instant at which the critical engine is failed, and the instant at which the pilot recognizes and reacts to the engine failure, as indicated by the pilot's application of the first retarding means during accelerate-stop tests.

(b) V_{2MIN}, in terms of calibrated airspeed, may not be less than—

(1) 1.2 V_S for—

(i) Two-engine and three-engine turbopropeller and reciprocating engine powered airplanes; and

(ii) Turbojet powered airplanes without provisions for obtaining a significant reduction in the one-engine-inoperative power-on stalling speed;

(2) 1.15 V_S for—

(i) Turbopropeller and reciprocating engine powered airplanes with more than three engines; and

(ii) Turbojet powered airplanes with provisions for obtaining a significant reduction in the one-engine-inoperative power-on stalling speed; and

(3) 1.10 times V_{MC} established under § 25.149.

(c) V_2, in terms of calibrated airspeed, must be selected by the applicant to provide at least the gradient of climb required by § 25.121(b) but may not be less than—

(1) V_{2MIN}, and

(2) V_R plus the speed increment attained (in accordance with § 25.111 (c)(2)) before reaching a height of 35 feet above the takeoff surface.

(d) V_{MU} is the calibrated airspeed at and above which the airplane can safely lift off the ground, and continue the takeoff. V_{MU} speeds must be selected by the applicant throughout the range of thrust-to-weight ratios to be certificated. These speeds may be

(1) The pressures on the wheel braking systems may not exceed those specified by the brake manufacturer;

(2) The brakes may not be used so as to cause excessive wear of brakes or tires; and

(3) Means other than wheel brakes may be used if that means—

(i) Is safe and reliable;

(ii) Is used so that consistent results can be expected in service; and

(iii) Is such that exceptional skill is not required to control the airplane.

(c) For seaplanes and amphibians, the landing distance on water must be determined on smooth water.

(d) For skiplanes, the landing distance on snow must be determined on smooth, dry, snow.

(e) The landing distance data must include correction factors for not more than 50 percent of the nominal wind components along the landing path opposite to the direction of landing, and not less than 150 percent of the nominal wind components along the landing path in the direction of landing.

(f) If any device is used that depends on the operation of any engine, and if the landing distance would be noticeably increased when a landing is made with that engine inoperative, the landing distance must be determined with that engine inoperative unless the use of compensating means will result in a landing distance not more than that with each engine operating.

CONTROLLABILITY AND MANEUVERABILITY

§ 25.143 General.

(a) The airplane must be safely controllable and maneuverable during—

(1) Takeoff;

(2) Climb;

(3) Level flight;

(4) Descent; and

(5) Landing.

(b) It must be possible to make a smooth transition from one flight condition to any other flight condition without exceptional piloting skill, alertness, or strength, and without danger of exceeding the airplane limit-load factor under any probable operating conditions, including—

(1) The sudde   lure of the critical engine;

(2) For airplanes with three or more engines, the sudden failure of the second critical engine when the airplane is in the en route, approach, or landing configuration and is trimmed with the critical engine inoperative; and

(3) Configuration changes, including deployment or retraction of deceleration devices.

(c) If, during the testing required by paragraphs (a) and (b) of this section, marginal conditions exist with regard to required pilot strength, the "strength of pilots" limits may not exceed the limits prescribed in the following table:

Values in pound of force as applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
For temporary application	75	60	150
For prolonged application	10	5	20

(d) In showing the temporary control force limitations of paragraph (c) of this section, approved operating procedures or conventional operating practices must be followed (including being as nearly trimmed as possible at the next preceding steady flight condition, except that, in the case of takeoff, the airplane must be trimmed in accordance with approved operating procedures).

(e) For the purpose of complying with the prolonged control force limitations of paragraph (c) of this section, the airplane must be as nearly trimmed as possible.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

§ 25.145 Longitudinal control.

(a) It must be possible at any speed between the trim speed prescribed in § 25.49(c)(2)(i) and V_{S1} (for reciprocating engine powered airplanes), or at any speed between the trim speed prescribed in § 25.103(b)(1) and V_S (for turbine engine powered airplanes), to pitch the nose downward so that the acceleration to this selected trim speed is prompt with—

(1) The airplane trimmed at the trim speed prescribed in § 25.49(c)(2)(i) (for reciprocating engine powered airplanes), or in § 25.103(b)(1) (for turbine engine powered airplanes);
(2) The landing gear extended;
(3) The wing flaps (i) retracted and (ii) extended; and
(4) Power (i) off and (ii) at maximum continuous power on the engines.

(b) With the landing gear extended, no change in trim control, or exertion of more than 50 pounds control force (representative of the maximum temporary force that readily can be applied by one hand) may be required for the following maneuvers:

(1) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{S1}, extend the flaps as rapidly as possible while maintaining the airspeed at approximately 40 percent above the stalling speed existing at each instant throughout the maneuver.
(2) Repeat paragraph (b)(1) except initially extend the flaps and then retract them as rapidly as possible.
(3) Repeat paragraph (b)(2) except with takeoff power.
(4) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{S1}, apply takeoff power rapidly while maintaining the same airspeed.
(5) Repeat paragraph (b)(4) except with flaps extended.
(6) With power off, flaps extended, and the airplane trimmed at 1.4 V_{S1}, obtain and maintain airspeeds between 1.1 V_{S1}, and either 1.7 V_{S1}, or V_{FE}, whichever is lower.

(c) Is must be possible, without exceptional piloting skill, to prevent loss of altitude when complete retraction of the high lift devices from any position is begun during steady, straight, level flight at 1.1 V_{S1} for propeller powered airplanes, or 1.2 V_{S1} for turbojet powered airplanes, with—

(1) Simultaneous application of not more than takeoff power taking into account the critical engine operating conditions;
(2) The landing gear extended; and
(3) The critical combinations of landing weights and altitudes.

If gated high-lift device control positions are provided, retraction must be shown from any position from the maximum landing position to the first gated position, between gated positions, and from the last gated position to the full retraction position. In addition, the first gated control position from the landing position must correspond with the high-lift devices configuration used to establish the go-around procedure from the landing configuration. Each gated control position must require a separate and distinct motion of the control to pass through the gated position and must have features to prevent inadvertent movement of the control through the gated position.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.147 Directional and lateral control.

(a) *Directional control; general.* It must be possible, while holding the wings approximately level, to safely make reasonably sudden changes in heading in both directions. This must be shown at 1.4 V_{S1} for heading changes up to 15° (except that the heading change at which the rudder pedal force is 150 pounds need not be exceeded), and with—

(1) The critical engine inoperative and its propeller in the minimum drag position;
(2) The power required for level flight at 1.4 V_{S1}, but not more than maximum continuous power;
(3) The most unfavorable center of gravity;
(4) Landing gear retracted;
(5) Flaps in the approach position; and
(6) Maximum landing weight.

(b) *Directional control; airplanes with four or more engines.* Airplanes with four or more engines must meet the requirements of paragraph (a) of this section except that—

(1) The two critical engines must be inoperative with their propellers (if applicable) in the minimum drag position;
(2) The center of gravity must be in the most forward position; and
(3) The flaps must be in the most favorable climb position.

(c) *Lateral control; general.* It must be possible to make 20° banked turns, with and against the inoperative engine, from steady flight at a speed equal to 1.4 V_{S1}, with—

(1) The critical engine inoperative and its propeller (if applicable) in the minimum drag position;
(2) The remaining engines at maximum continuous power;
(3) The most unfavorable center of gravity;
(4) Landing gear (i) retracted and (ii) extended;
(5) Flaps in the most favorable climb position; and
(6) Maximum takeoff weight.

(d) *Lateral control; airplanes with four or more engines.* Airplanes with four or more engines must be able to make 20° banked turns, with and against the inoperative engines, from steady flight at a speed equal to 1.4 V_{S1}, with maximum continuous power, and with the airplane in the configuration prescribed by paragraph (b) of this section.

(e) *Lateral control; all engines operating.* With the engines operating, roll response must allow normal maneuvers (such as recovery from upsets produced by gusts and the initiation of evasive maneuvers). There must be enough excess lateral control in sideslips (up to sideslip angles that might be required in normal operation), to allow a limited amount of maneuvering and to correct for gusts. Lateral control must be enough at any speed up to V_{FC}/M_{FC} to provide a peak roll rate necessary for safety, without excessive control forces or travel.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

§ 25.149 Minimum control speed.

(a) In establishing the minimum control speeds required by this section, the method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability expected in service.

(b) V_{MC} is the calibrated airspeed, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than five degrees.

(c) V_{MC} may not exceed 1.2 V_S with—

(1) Maximum available takeoff power or thrust on the engines;
(2) The most unfavorable center of gravity;
(3) The airplane trimmed for takeoff;
(4) The maximum sea level takeoff weight (or any lesser weight necessary to show V_{MC});
(5) The airplane in the most critical takeoff configuration existing along the flight path after the airplane becomes airborne, except with the landing gear retracted;
(6) The airplane airborne and the ground effect negligible; and
(7) If applicable, the propeller of the inoperative engine—
(i) Windmilling;
(ii) In the most probable position for the specific design of the propeller control; or
(iii) Feathered, if the airplane has an automatic feathering device acceptable for showing compliance with the climb requirements of § 25.121.

(d) The rudder forces required to maintain control at V_{MC} may not exceed 150 pounds nor may it be necessary to reduce power or thrust of the operative engines. During recovery, the airplane may not assume any dangerous attitude or require exceptional piloting skill, alertness, or strength to prevent a heading change of more than 20 degrees.

(e) V_{MCG}, the minimum control speed on the ground, is the calibrated airspeed during the takeoff run, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with the use of primary aerodynamic controls alone (without the use of nosewheel steering) to enable the takeoff to be safely continued using normal piloting skill and rudder control forces not exceeding 150 pounds. In the determination of V_{MCG}, assuming that the path of the airplane accelerating with all engines operating is along the centerline of the runway, its path from the point at which the critical engine is made inoperative to the point at which recovery to a direction parallel to the centerline is completed may not deviate more than 30 feet laterally from the centerline at any point. V_{MCG} must be established with—

(1) The airplane in each takeoff configuration or, at the option of the applicant, in the most critical takeoff configuration;

(2) Maximum available takeoff power or thrust on the operating engines;

(3) The most unfavorable center of gravity;

(4) The airplane trimmed for takeoff; and

(5) The most unfavorable weight in the range of takeoff weights.

(f) V_{MCL}, the minimum control speed during landing approach with all engines operating, is the calibrated airspeed at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL} must be established with—

(1) The airplane in the most critical configuration for approach with all engines operating;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with all engines operating;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL}); and

(5) Maximum available takeoff power or thrust on the operating engines.

(g) For airplanes with three or more engines, V_{MCL-2}, the minimum control speed during landing approach with one critical engine inoperative, is the calibrated airspeed at which, when a second critical engine is suddenly made inoperative, it is possible to recover control of the airplane with both engines still inoperative and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL-2} must be established with—

(1) The airplane in the most critical configuration for approach with the critical engine inoperative;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with the critical engine inoperative;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL-2});

(5) The power or thrust on the operating engines required to maintain an approach path angle of 3 degrees when one critical engine is inoperative; and

(6) The power or thrust on the operating engines rapidly changed, immediately after the second critical engine is made inoperative, from the power or thrust prescribed in paragraph (g)(5) of this section to—

(i) Minimum available power or thrust; and

(ii) Maximum available takeoff power or thrust.

(h) The rudder control forces required to maintain control at V_{MCL} and V_{MCL-2} may not exceed 150 pounds, nor may it be necessary to reduce the power or thrust of the operating engines. In addition, the airplane may not assume any dangerous attitudes or require exceptional piloting skill, alertness, or strength to prevent a divergence in the approach flight path that would jeopardize continued safe approach when—

(1) The critical engine is suddenly made inoperative; and

(2) For the determination of V_{MCL-2}, the power or thrust on the operating engines is changed in accordance with paragraph (g)(6) of this section.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

TRIM

§ 25.161 Trim.

(a) *General.* Each airplane must meet the trim requirements of this section after being trimmed, and without further pressure upon, or movement of, either the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) *Lateral and directional trim.* The airplane must maintain lateral and directional trim with the most adverse lateral displacement of the center of gravity within the relevant operating limitations, during normally expected conditions of operation (including operation at any speed from 1.4 V_{S1} to V_{MO}/M_{MO}).

(c) *Longitudinal trim.* The airplane must maintain longitudinal trim during—

(1) A climb with maximum continuous power at a speed not more than 1.4 V_{S1}, with the landing gear retracted, and the flaps (i) retracted and (ii) in the takeoff position;

(2) A glide with power off at a speed not more than 1.4 V_{S1}, with the landing gear extended, the wing flaps (i) retracted and (ii) extended, the most unfavorable center of gravity position approved for landing with the maximum landing weight, and with the most unfavorable center of gravity position approved for landing regardless of weight; and

(3) Level flight at any speed from 1.4 V_{S1}, to V_{MO}/M_{MO}, with the landing gear and flaps retracted, and from 1.4 V_{S1} to V_{LE} with the landing gear extended.

(d) *Longitudinal, directional, and lateral trim.* The airplane must maintain longitudinal, directional, and lateral trim (and for the lateral trim, the angle of bank may not exceed five degrees) at 1.4 V_{S1} during climbing flight with—

(1) The critical engine inoperative;

(2) The remaining engines at maximum continuous power; and

(3) The landing gear and flaps retracted.

(e) *Airplanes with four or more engines.* Each airplane with four or more engines must maintain trim in rectilinear flight—

(1) At the climb speed, configuration, and power required by § 25.123(a) for the purpose of establishing the rate of climb;

(2) With the most unfavorable center of gravity position; and

(3) At the weight at which the two-engine-inoperative climb is equal to at least 0.013 V_{S0}^2 at an altitude of 5,000 feet.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-38, 41 FR 55466, Dec. 20, 1976]

§ 25.171 General.

The airplane must be longitudinally, directionally, and laterally stable in accordance with the provisions of §§ 25.173 through 25.177. In addition, suitable stability and control feel (static stability) is required in any condition normally encountered in service, if flight tests show it is necessary for safe operation.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

§ 25.173 Static longitudinal stability.

Under the conditions specified in § 25.175, the characteristics of the elevator control forces (including friction) must be as follows:

(a) A pull must be required to obtain and maintain speeds below the specified trim speed, and a push must be required to obtain and maintain speeds above the specified trim speed. This must be shown at any speed that can be obtained except speeds higher than the landing gear or wing flap operating limit speeds or V_{FC}/M_{FC}, whichever is appropriate, or lower than the minimum speed for steady unstalled flight.

(b) The airspeed must return to within 10 percent of the original trim speed for the climb, approach, and landing conditions specified in § 25.175 (a), (c), and (d), and must return to within 7.5 percent of the original trim speed for the cruising condition specified in § 25.175(b), when the control force is slowly released from any speed within the range specified in paragraph (a) of this section.

(c) The average gradient of the stable slope of the stick force versus speed curve may not be less than 1 pound for each 6 knots.

(d) Within the free return speed range specified in paragraph (b) of this section, it is permissible for the airplane, without control forces, to stabilize on speeds above or below the desired trim speeds if exceptional attention on the part of the pilot is not required to return to and maintain the desired trim speed and altitude.

[Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

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§ 25.173 Demonstration of static longitudinal stability.

Static longitudinal stability must be shown as follows:

(a) *Climb.* The stick force curve must have a stable slope at speeds between 85 and 115 percent of the speed at which the airplane—

(1) Is trimmed, with—

(i) Wing flaps retracted;

(ii) Landing gear retracted;

(iii) Maximum takeoff weight; and

(iv) 75 percent of maximum continuous power for reciprocating engines or the maximum power or thrust selected by the applicant as an operating limitation for use during climb for turbine engines; and

(2) Is trimmed at the speed for best rate-of-climb except that the speed need not be less than 1.4 V_{S1}.

(b) *Cruise.* Static longitudinal stability must be shown in the cruise condition as follows:

(1) With the landing gear retracted at high speed, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than V_{FC}/M_{FC}, nor speeds that require a stick force of more than 50 pounds), with—

(i) The wing flaps retracted;

(ii) The center of gravity in the most adverse position (see § 25.27);

(iii) The most critical weight between the maximum takeoff and maximum landing weights;

(iv) 75 percent of maximum continuous power for reciprocating engines or for turbine engines, the maximum cruising power selected by the applicant as an operating limitation (see § 25.1521), except that the power need not exceed that required at V_{MO}/M_{MO}; and

(v) The airplane trimmed for level flight with the power required in paragraph (b)(1)(iv) of this section.

(2) With the landing gear retracted at low speed, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than the minimum speed of the applicable speed range prescribed in paragraph (b)(1), nor speeds that require a stick force of more than 50 pounds), with—

(i) Wing flaps, center of gravity position, and weight as specified in paragraph (b)(1) of this section;

(ii) Power required for level flight at a speed equal to $V_{MO} + 1.4\ V_{S1}/2$; and

(iii) The airplane trimmed for level flight with the power required in paragraph (b)(2)(ii) of this section.

(3) With the landing gear extended, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than V_{LE}, nor speeds that require a stick force of more than 50 pounds), with—

(i) Wing flap, center of gravity position, and weight as specified in paragraph (b)(1) of this section;

(ii) 75 percent of maximum continuous power for reciprocating engines or, for turbine engines, the maximum cruising power selected by the applicant as an operating limitation, except that the power need not exceed that required for level flight at V_{LE}; and

(iii) The aircraft trimmed for level flight with the power required in paragraph (b)(3)(ii) of this section.

(c) *Approach.* The stick force curve must have a stable slope at speeds between 1.1 V_{S1} and 1.8 V_{S1}, with—

(1) Wing flaps in the approach position;

(2) Landing gear retracted;

(3) Maximum landing weight; and

(4) The airplane trimmed at 1.4 V_{S1} with enough power to maintain level flight at this speed.

(d) *Landing.* The stick force curve must have a stable slope, and the stick force may not exceed 80 pounds, at speeds between 1.1 V_{S0} and 1.3 V_{S0} with—

(1) Wing flaps in the landing position;

(2) Landing gear extended;

(3) Maximum landing weight;

(4) Power or thrust off on the engines; and

(5) The airplane trimmed at 1.4 V_{S0} with power or thrust off.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

§ 25.177 Static directional and lateral stability.

(a) The static directional stability (as shown by the tendency to recover from a skid with the rudder free) must be positive for any landing gear and flap position and symmetrical power condition, at speeds from 1.2 V_{S1} up to V_{FE}, V_{LE}, or V_{FC}/M_{FC} (as appropriate).

(b) The static lateral stability (as shown by the tendency to raise the low wing in a sideslip with the aileron controls free and for any landing gear and flap position and symmetrical power condition) may not be negative at any airspeed (except speeds higher than V_{FE} or V_{LE}, when appropriate) in the following airspeed ranges:

(1) From 1.2 V_{S1} to V_{MO}/M_{MO}.

(2) From V_{MO}/M_{MO} to V_{FC}/M_{FC} unless the Administrator finds that the divergence is—

(i) Gradual;

(ii) Easily recognizable by the pilot; and

(iii) Easily controllable by the pilot.

(c) In straight, steady, sideslips (unaccelerated forward slips) the aileron and rudder control movements and forces must be substantially proportional to the angle of sideslip, and the factor of proportionality must lie between limits found necessary for safe operation throughout the range of sideslip angles appropriate to the operation of the airplane. At greater angles, up to the angle at which full rudder control is used or a rudder pedal force of 180 pounds is obtained, the rudder pedal forces may not reverse and increased rudder deflection must produce increased angles of sideslip. Unless the airplane has a yaw indicator, there must be enough bank accompanying sideslipping to clearly indicate any departure from steady unyawed flight.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.181 Dynamic stability.

(a) Any short period oscillation, not including combined lateral-directional oscillations, occurring between stalling speed and maximum allowable speed appropriate to the configuration of the airplane must be heavily damped with the primary controls—

(1) Free; and

(2) In a fixed position.

(b) Any combined lateral-directional oscillations ("Dutch roll") occurring between stalling speed and maximum allowable speed appropriate to the configuration of the airplane must be positively damped with controls free, and must be controllable with normal use of the primary controls without requiring exceptional pilot skill.

[Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

STALLS

§ 25.201 Stall demonstration.

(a) Stalls must be shown in straight flight and in 30 degree banked turns with—

(1) Power off; and

(2) The power necessary to maintain level flight at 1.6 V_{S1} (where V_{S1} corresponds to the stalling speed with flaps in the approach position, the landing gear retracted, and maximum landing weight).

(b) In either condition required by paragraph (a) of this section, it must be possible to meet the applicable requirements of § 25.203 with—

(1) Flaps and landing gear in any likely combination of positions;

(2) Representative weights within the range for which certification is requested; and

(3) The most adverse center of gravity for recovery.

(c) The following procedure must be used to show compliance with § 25.203:

(1) With the airplane trimmed for straight flight at the speed prescribed in § 25.103(b)(1), reduce the speed with the elevator control until it is steady at slightly above stalling speed. Apply elevator control so that the speed reduction does not exceed one knot per

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second unt[illegible] the airplane is stalled, or (ii) the c[illegible]ol reaches the stop.

(2) As soon as the airplane is stalled, recover by normal recovery techniques.

(d) Occurrence of stall is defined as follows:

(1) The airplane may be considered stalled when, at an angle of attack measurably greater than that for maximum lift, the inherent flight characteristics give a clear and distinctive indication to the pilot that the airplane is stalled. Typical indications of a stall, occurring either individually or in combination, are—

(i) A nose-down pitch that cannot be readily arrested;

(ii) A roll that cannot be readily arrested; or

(iii) If clear enough, a loss of control effectiveness, an abrupt change in control force or motion, or a distinctive shaking of the pilot's controls.

(2) For any configuration in which the airplane demonstrates an unmistakable inherent aerodynamic warning of a magnitude and severity that is a strong and effective deterrent to further speed reduction, the airplane may be considered stalled when it reaches the speed at which the effective deterrent is clearly manifested.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-38, 41 FR 55466, Dec. 20, 1976; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.203 Stall characteristics.

(a) It must be possible to produce and to correct roll and yaw by unreversed use of the aileron and rudder controls, up to the time the airplane is stalled. No abnormal nose-up pitching may occur. The longitudinal control force must be positive up to and throughout the stall. In addition, it must be possible to promptly prevent stalling and to recover from a stall by normal use of the controls.

(b) For level wing stalls, the roll occurring between the stall and the completion of the recovery may not exceed approximately 20 degrees.

(c) For turning flight stalls, the action of the airplane after the stall may not be so violent or extreme as to make it difficult, with normal piloting skill, to effect a prompt recovery and to regain control of the airplane.

§ 25.205 Stalls: Critical engine inoperative.

(a) It must be possible to safely recover from a stall with the critical engine inoperative—

(1) Without applying power to the inoperative engine;

(2) With flaps and landing gear retracted; and

(3) With the remaining engines at up to 75 percent of maximum continuous power, or up to the power at which the wings can be held level with the use of maximum control travel, whichever is less.

(b) The operating engines may be throttled back during stall recovery from stalls with the critical engine inoperative.

§ 25.207 Stall warning.

(a) Stall warning with sufficient margin to prevent inadvertent stalling with the flaps and landing gear in any normal position must be clear and distinctive to the pilot in straight and turning flight.

(b) The warning may be furnished either through the inherent aerodynamic qualities of the airplane or by a device that will give clearly distinguishable indications under expected conditions of flight. However, a visual stall warning device that requires the attention of the crew within the cockpit is not acceptable by itself. If a warning device is used, it must provide a warning in each of the airplane configuations prescribed in paragraph (a) of this section at the speed prescribed in paragraph (c) of this section.

(c) The stall warning must begin at a speed exceeding the stalling speed (i.e., the speed at which the airplane stalls or the minimum speed demonstrated, whichever is applicable under the provisions of § 25.201(d)) by seven percent or at any lesser margin if the stall warning has enough clarity, duration, distinctiveness, or similar properties.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13118, Oct. 15, 1965; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

Ground and Water Handling Characteristics

§ 25.231 Longitudinal stability and control.

(a) Landplanes may have no uncontrollable tendency to nose over in any reasonably expected operating condition or when rebound occurs during landing or takeoff. In addition—

(1) Wheel brakes must operate smoothly and may not cause any undue tendency to nose over; and

(2) If a tail-wheel landing gear is used, it must be possible, during the takeoff ground run on concrete, to maintain any altitude up to thrust line level, at 80 percent of V_{S1}.

(b) For seaplanes and amphibians, the most adverse water conditions safe for takeoff, taxiing, and landing, must be established.

§ 25.233 Directional stability and control.

(a) There may be no uncontrollable ground-looping tendency in 90° cross winds, up to a wind velocity of 20 knots or 0.2 V_{S0}, whichever is greater, except that the wind velocity need not exceed 25 knots. At any speed at which the airplane may be expected to be operated on the ground. This may be shown while establishing the 90° cross component of wind velocity required by § 25.237.

(b) Landplanes must be satisfactorily controllable, without exceptional piloting skill or alertness, in power-off landings at normal landing speed, without using brakes or engine power to maintain a straight path. This may be shown during power-off landings made in conjunction with other tests.

(c) The airplane must have adequate directional control during taxiing. This may be shown during taxiing prior to takeoffs made in conjunction with other tests.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.235 Taxiing condition.

The shock absorbing mechanism may not damage the structure of the airplane when the airplane is taxied on the roughest ground that may reasonably be expect[illegible] [illegible] normal operation.

§ 25.237 Wind velocities.

(a) For landplanes and amphibians, a 90-degree cross component of wind velocity, demonstrated to be safe for takeoff and landing, must be established for dry runways and must be at least 20 knots or 0.2 V_{S0}, whichever is greater, except that it need not exceed 25 knots.

(b) For seaplanes and amphibians, the following applies:

(1) A 90-degree cross component of wind velocity, up to which takeoff and landing is safe under all water conditions that may reasonably be expected in normal operation, must be established and must be at least 20 knots or 0.2 V_{S0}, whichever is greater, except that it need not exceed 25 knots.

(2) A wind velocity, for which taxiing is safe in any direction under all water conditions that may reasonably be expected in normal operation, must be established and must be at least 20 knots or 0.2 V_{S0}, whichever is greater, except that it need not exceed 25 knots.

[Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.239 Spray characteristics, control, and stability on water.

(a) For seaplanes and amphibians, during takeoff, taxiing, and landing, and in the conditions set forth in paragraph (b) of this section, there may be no—

(1) Spray characteristics that would impair the pilot's view, cause damage, or result in the taking in of an undue quantity of water;

(2) Dangerously uncontrollable porpoising, bounding, or swinging tendency; or

(3) Immersion of auxiliary floats or sponsons, wing tips, propeller blades, or other parts not designed to withstand the resulting water loads.

(b) Compliance with the requirements of paragraph (a) of this section must be shown—

(1) In water conditions, from smooth to the most adverse condition established in accordance with § 25.231;

(2) In wind and cross-wind velocities, water currents, and associated waves

and swells that may reasonably be expected in operation on water;

(3) At speeds that may reasonably be expected in operation on water;

(4) With sudden failure of the critical engine at any time while on water; and

(5) At each weight and center of gravity position, relevant to each operating condition, within the range of loading conditions for which certification is requested.

(c) In the water conditions of paragraph (b) of this section, and in the corresponding wind conditions, the seaplane or amphibian must be able to drift for five minutes with engines inoperative, aided, if necessary, by a sea anchor.

MISCELLANEOUS FLIGHT REQUIREMENTS

§ 25.251 Vibration and buffeting.

(a) The airplane must be designed to withstand any vibration and buffeting that might occur in any likely operating condition. This must be shown by calculations, resonance tests, or other tests found necessary by the Administrator.

(b) Each part of the airplane must be shown in flight to be free from excessive vibration, under any appropriate speed and power conditions up to at least the minimum value of V_D allowed in § 25.335. The maximum speeds shown must be used in establishing the operating limitations of the airplane in accordance with § 25.1505. In addition, it must be shown by analysis or tests, that the airplane is free from such vibration that would prevent safe flight under the conditions in § 25.629(d).

(c) Except as provided in paragraph (d) of this section, there may be no buffeting condition, in normal flight, including configuration changes during cruise, severe enough to interfere with the control of the airplane, to cause excessive fatigue to the crew, or to cause structural damage. Stall warning buffeting within these limits is allowable.

(d) There may be no perceptible buffeting condition in the cruise configuration in straight flight at any speed up to V_{MO}/M_{MO}, except that stall warning buffeting is allowable.

(e) With the airplane in the cruise configuration, the positive maneuvering load factors at which the onset of perceptible buffeting occurs must be determined for the ranges of airspeed or Mach Number, weight, and altitude for which the airplane is to be certificated. The envelopes of load factor, speed, altitude, and weight must provide a sufficient range of speeds and load factors for normal operations. Probable inadvertent excursions beyond the boundaries of the buffet onset envelopes may not result in unsafe conditions.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.253 High-speed characteristics.

(a) *Speed increase and recovery characteristics.* The following speed increase and recovery characteristics must be met:

(1) Operating conditions and characteristics likely to cause inadvertent speed increases (including upsets in pitch and roll) must be simulated with the airplane trimmed at any likely cruise speed up to V_{MO}/M_{MO}. These conditions and characteristics include gust upsets, inadvertent control movements, low stick force gradient in relation to control friction, passenger movement, leveling off from climb, and descent from Mach to airspeed limit altitudes.

(2) Allowing for pilot reaction time after effective inherent or artificial speed warning occurs, it must be shown that the airplane can be recovered to a normal attitude and its speed reduced to V_{MO}/M_{MO}, without—

(i) Exceptional piloting strength or skill;

(ii) Exceeding V_D/M_D, V_{DF}/M_{DF}, or the structural limitations; and

(iii) Buffeting that would impair the pilot's ability to read the instruments or control the airplane for recovery.

(3) There may be no control reversal about any axis at any speed up to V_{DF}/M_{DF}. Any reversal of elevator control force or tendency of the airplane to pitch, roll, or yaw must be mild and readily controllable, using normal piloting techniques.

(b) *Maximum speed for stability characteristics, V_{FC}/M_{FC}.* V_{FC}/M_{FC} is the maximum speed at which the requirements of §§ 25.147(e), 25.175(b)(1), 25.177, and 25.181 must be met with flaps and landing gear retracted. It may not be less than a speed midway between V_{MO}/M_{MO} and V_{DF}/M_{DF}, except that, for altitudes where Mach number is the limiting factor, M_{FC} need not exceed the Mach number at which effective speed warning occurs.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-54, 45 FR 60172, Sept. 11, 1980]

§ 25.255 Out-of-trim characteristics.

(a) From an initial condition with the airplane trimmed at cruise speeds up to V_{MO}/M_{MO}, the airplane must have satisfactory maneuvering stability and controllability with the degree of out-of-trim in both the airplane nose-up and nose-down directions, which results from the greater of—

(1) A three-second movement of the longitudinal trim system at its normal rate for the particular flight condition with no aerodynamic load (or an equivalent degree of trim for airplanes that do not have a power-operated trim system), except as limited by stops in the trim system, including those required by § 25.655(b) for adjustable stabilizers; or

(2) The maximum mistrim that can be sustained by the autopilot while maintaining level flight in the high speed cruising condition.

(b) In the out-of-trim condition specified in paragraph (a) of this section, when the normal acceleration is varied from +1 g to the positive and negative values specified in paragraph (c) of this section—

(1) The stick force vs. g curve must have a positive slope at any speed up to and including V_{FC}/M_{FC}; and

(2) At speeds between V_{FC}/M_{FC} and V_{DF}/M_{DF} the direction of the primary longitudinal control force may not reverse.

(c) Except as provided in paragraphs (d) and (e) of this section, compliance with the provisions of paragraph (a) of this section must be demonstrated in flight over the acceleration range—

(1) −1 g to +2.5 g; or

(2) 0 g to 2.0 g, and extrapolating by an acceptable method to −1 g and +2.5 g.

(d) If the procedure set forth in paragraph (c)(2) of this section is used to demonstrate compliance and marginal conditions exist during flight test with regard to reversal of primary longitudinal control force, flight tests must be accomplished from the normal acceleration at which a marginal condition is found to exist to the applicable limit specified in paragraph (b)(1) of this section.

(e) During flight tests required by paragraph (a) of this section, the limit maneuvering load factors prescribed in §§ 25.333(b) and 25.337, and the maneuvering load factors associated with probable inadvertent excursions beyond the boundaries of the buffet onset envelopes determined under § 25.251(e), need not be exceeded. In addition, the entry speeds for flight test demonstrations at normal acceleration values less than 1 g must be limited to the extent necessary to accomplish a recovery without exceeding V_{DF}/M_{DF}.

(f) In the out-of-trim condition specified in paragraph (a) of this section, it must be possible from an overspeed condition at V_{DF}/M_{DF} to produce at least 1.5 g for recovery by applying not more than 125 pounds of longitudinal control force using either the primary longitudinal control alone or the primary longitudinal control and the longitudinal trim system. If the longitudinal trim is used to assist in producing the required load factor, it must be shown at V_{DF}/M_{DF} that the longitudinal trim can be actuated in the airplane nose-up direction with the primary surface loaded to correspond to the least of the following airplane nose-up control forces:

(1) The maximum control forces expected in service as specified in §§ 25.301 and 25.397.

(2) The control force required to produce 1.5 g.

(3) The control force corresponding to buffeting or other phenomena of such intensity that it is a strong deterrent to further application of primary longitudinal control force.

[Amdt. No. 25-42, 43 FR 2322, Jan. 16, 1978]

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PILOT CONTROL FORCE LIMITS (SECONDARY CONTROLS)—Continued

Control	Limit pilot forces
Push-pull	To be chosen by applicant.

[1] Limited to flap, tab, stabilizer, spoiler, and landing gear operation controls.

§ 25.407 Trim tab effects.

The effects of trim tabs on the control surface design conditions must be accounted for only where the surface loads are limited by maximum pilot effort. In these cases, the tabs are considered to be deflected in the direction that would assist the pilot, and the deflections are—

(a) For elevator trim tabs, those required to trim the airplane at any point within the positive portion of the pertinent flight envelope in § 25.333(b), except as limited by the stops; and

(b) For aileron and rudder trim tabs, those required to trim the airplane in the critical unsymmetrical power and loading conditions, with appropriate allowance for rigging tolerances.

§ 25.409 Tabs.

(a) *Trim tabs.* Trim tabs must be designed to withstand loads arising from all likely combinations of tab setting, primary control position, and airplane speed (obtainable without exceeding the flight load conditions prescribed for the airplane as a whole), when the effect of the tab is opposed by pilot effort forces up to those specified in § 25.397(b).

(b) *Balancing tabs.* Balancing tabs must be designed for deflections consistent with the primary control surface loading conditions.

(c) *Servo tabs.* Servo tabs must be designed for deflections consistent with the primary control surface loading conditions obtainable within the pilot maneuvering effort, considering possible opposition from the trim tabs.

§ 25.415 Ground gust conditions.

(a) The control system must be designed as follows for control surface loads due to ground gusts and taxiing downwind:

(1) The control system between the stops nearest the surfaces and the cockpit controls must be designed for loads corresponding to the limit hinge moments H of paragraph (a)(2) of this section. These loads need not exceed—

(i) The loads corresponding to the maximum pilot loads in § 25.397(c) for each pilot alone; or

(ii) 0.75 times these maximum loads for each pilot when the pilot forces are applied in the same direction.

(2) The control system stops nearest the surfaces, the control system locks, and the parts of the systems (if any) between these stops and locks and the control surface horns, must be designed for limit hinge moments H obtained from the formula, $H=KcSq$, where—

H=limit hinge moment (ft. lbs.);
c=mean chord of the control surface aft of the hinge line (ft.);
S=area of the control surface aft of the hinge line (sq. ft.);
q=dynamic pressure (p.s.f.) based on a design speed not less than $14.6\sqrt{W/S}+14.6$ (f.p.s.), except that the design speed need not exceed 88 f.p.s.; and
K=limit hinge moment factor for ground gusts derived in paragraph (b) of this section.

(b) The limit hinge moment factor K for ground gusts must be derived as follows:

Surface	K	Position of controls
(a) Aileron	0.75	Control column locked or lashed in mid-position.
(b) ......do	[1] ±0.50	Ailerons at full throw.
(c) Elevator	[1] ±0.75	(c) Elevator full down.
(d) ......do	[1] ±0.75	(d) Elevator full up.
(e) Rudder	0.75	(e) Rudder in neutral.
(f) ......do	0.75	(f) Rudder at full throw.

[1] A positive value of K indicates a moment tending to depress the surface, while a negative value of K indicates a moment tending to raise the surface.

§ 25.427 Unsymmetrical loads.

(a) Horizontal tail surfaces and their supporting structure must be designed for unsymmetrical loads arising from yawing and slipstream effects, in combination with the prescribed flight conditions.

(b) In the absence of more rational data, the following apply:

tected location of control system elements or protective devices such as splitter plates or energy absorbing material is acceptable. Where compliance is shown by analysis, tests, or both, use of data on airplanes having similar structural design is acceptable.

[Amdt. 25-23, 35 FR 5674, Apr. 8, 1970]

CONTROL SURFACES

§ 25.651 Proof of strength.

(a) Limit load tests of control surfaces are required. These tests must include the horn or fitting to which the control system is attached.

(b) Compliance with the special factors requirements of §§ 25.619 through 25.625 and 25.657 for control surface hinges must be shown by analysis or individual load tests.

§ 25.655 Installation.

(a) Movable tail surfaces must be installed so that there is no interference between any surfaces when one is held in its extreme position and the others are operated through their full angular movement.

(b) If an adjustable stabilizer is used, it must have stops that will limit its range of travel to the maximum for which the airplane is shown to meet the trim requirements of § 25.161.

§ 25.657 Hinges.

(a) For control surface hinges, including ball, roller, and self-lubricated bearing hinges, the approved rating of the bearing may not be exceeded. For nonstandard bearing hinge configurations, the rating must be established on the basis of experience or tests and, in the absence of a rational investigation, a factor of safety of not less than 6.67 must be used with respect to the ultimate bearing strength of the softest material used as a bearing.

(b) Hinges must have enough strength and rigidity for loads parallel to the hinge line.

[Amdt. 25-23, 35 FR 5674, Apr. 8, 1970]

CONTROL SYSTEMS

§ 25.671 General.

(a) Each control and control system must operate with the ease, smoothness, and positiveness appropriate to its function.

(b) Each element of each flight control system must be designed, or distinctively and permanently marked, to minimize the probability of incorrect assembly that could result in the malfunctioning of the system.

(c) The airplane must be shown by analysis, tests, or both, to be capable of continued safe flight and landing after any of the following failures or jamming in the flight control system and surfaces (including trim, lift, drag, and feel systems), within the normal flight envelope, without requiring exceptional piloting skill or strength. Probable malfunctions must have only minor effects on control system operation and must be capable of being readily counteracted by the pilot.

(1) Any single failure, excluding jamming (for example, disconnection or failure of mechanical elements, or structural failure of hydraulic components, such as actuators, control spool housing, and valves).

(2) Any combination of failures not shown to be extremely improbable, excluding jamming (for example, dual electrical or hydraulic system failures, or any single failure in combination with any probable hydraulic or electrical failure).

(3) Any jam in a control position normally encountered during takeoff, climb, cruise, normal turns, descent, and landing unless the jam is shown to be extremely improbable, or can be alleviated. A runaway of a flight control to an adverse position and jam must be accounted for if such runaway and subsequent jamming is not extremely improbable.

(d) The airplane must be designed so that it is controllable if all engines fail. Compliance with this requirement may be shown by analysis where that method has been shown to be reliable.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5674, Apr. 8, 1970]

§ 25.672 Stability augmentation and automatic and power-operated systems.

If the functioning of stability augmentation or other automatic or power-operated systems is necessary to

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Section 3. CONTROLLABILITY AND MANEUVERABILITY

20. GENERAL - § 25.143.

a. Explanation. The purpose of § 25.143 is to verify that any operational maneuvers conducted within the operational envelope can be accomplished smoothly with average piloting skill and without exceeding any airplane structural limits. Control forces should not be so high that the pilot cannot safely maneuver the airplane. Also, the forces should not be so light it would take exceptional skill to maneuver the airplane without overstressing it or losing control. The airplane response to any control input should be predictable to the pilot.

b. The applicable regulation is § 25.143.

c. Procedures. Compliance with § 25.143 is primarily a qualitative determination by the pilot during the course of the flight test program. The control forces required and airplane response should be evaluated during changes from one flight condition to another and during maneuvering flight. The forces required should be compatible for each flight condition evaluated. For example, during an approach for landing, the forces should be light and the airplane responsive in order that adjustments in the flight path can be accomplished with a minimum of workload. In cruise flight, forces and airplane response should be such that inadvertent control input does not result in exceeding limits or in undesirable maneuvers. Longitudinal control forces should be evaluated during accelerated flight to ensure a positive stick force with increasing normal acceleration. Forces should be heavy enough at the limit load factor to prevent inadvertent excursions beyond design limit. Sudden engine failures should be investigated during any flight condition or in any configuration considered critical, if not covered by another Section of Part 25. Control forces considered excessive should be measured to show compliance with § 25.143(c), "strength of pilots" limits. Allowance should be made for delays in the initiation of recovery action appropriate to the situation.

21. LONGITUDINAL CONTROL - § 25.145

a. Explanation.

(1) Section 25.145(a) requires that there be adequate longitudinal control to promptly pitch the airplane nose down from, at, or near the stall to return to the original trim speed. The intent is to insure sufficient pitch control if inadvertently slowed to the point of stall.

(2) Section 25.145(b) requires changes to be made in flap position, power, and speed without undue effort when retrimming is not practical. The purpose is to insure that any of these changes are possible assuming that the pilot finds it necessary to devote at least one hand to the initiation of the desired operation without being overpowered by the primary airplane controls. The objective is that no excessive change in trim will result from the application or removal of power or the extension or retraction of wing flaps. Compliance with its terms also requires that the relation of control force to speed be such that reasonable changes in speed may be made without encountering very high control forces.

(3) Section 25.145(c) is concerned with the eventuality of going around during an approach for landing in which event it is desirable to be able to retract the wing flaps and leading edge slats, if applicable, quickly at such a rate that there will be no loss of altitude if power is applied simultaneously with the initiation of flap/slat retraction. The design feature involved in this requirement is the rate of flap/slat retraction. Several changes to § 25.145(c) were made as a result of Amendment 25-23, which became effective May 8, 1970.

(i) The use of maximum continuous power was changed to takeoff power because there is no need to reserve additional power (thrust) between maximum continuous power and takeoff power for contingencies since compliance at critical altitudes and weights is required. Further, takeoff power from a controllability standpoint could be more critical (i.e. pitch up). Some airplanes have had to limit their go-around thrust at certain weights because of this condition. The critical engine operating conditions within the approved airplane operating envelope must also be considered.

(ii) Partial flap retraction to a gated position (design feature to prevent inadvertent operation beyond this position) is permitted. The gate design requirements are in the rule.

(iii) The initial speed was changed from $1.1V_{S1}$ to a speed of $1.2V_{S1}$ for turbojet airplanes, which is intended to assure that the minimum inflight go-around speed is related to realistic landing touchdown speeds.

b. The applicable regulations are §§ 25.145(a),(b), and (c) of the FAR.

c. Procedures. The following test procedures outline an acceptable means for demonstrating compliance with § 25.145. These tests may be conducted at an optional altitude in accordance with § 25.21(c). Where applicable, the conditions should be maintained on the engines throughout the maneuver.

(1) Longitudinal control recovery, § 25.145(a):

(i) Configuration:

(A) Maximum weight or a lighter weight if considered more critical.

(B) Aft c.g. position.

(C) Landing gear extended.

(D) Wing flaps retracted and extended to the maximum landing position.

(E) Engine power at idle and maximum continuous.

(ii) Test procedure: The airplane should be trimmed at the speed for each configuration as prescribed in § 25.103(b)(1). The nose should be pitched downward from any speed between V trim and the stall. In past programs the most critical point has been at the stall when in stall buffet. The rate of speed increase should be adequate to promptly return to the trim point. Data from the stall characteristics test could be used to evaluate this condition at the stall.

(2) Longitudinal control, flap extension, § 25.145(b)(1).

(i) Configuration:

(A) Maximum landing weight.

(B) Critical c.g. position.

(C) Wing flaps retracted.

(D) Landing gear extended.

(E) Engine power at flight idle.

(ii) Test procedure: The airplane should be trimmed at a speed of $1.4V_S$. The flaps should be extended to the maximum landing position as rapidly as possible while maintaining approximately $1.4V_S$ for the flap position existing at each instant throughout the maneuver. The control forces should not exceed 50 lbs. (the maximum temporary forces that can be applied readily by one hand) throughout the maneuver without changing the trim control.

(3) Longitudinal control, flap retraction, §§ 25.145(b)(2) & (3).

(i) Configuration:

(A) Maximum landing weight.

(B) Critical c.g. position.

(C) Wing flaps extended to maximum landing position.

(D) Landing gear extended.

(E) Engine power at flight idle and takeoff.

(ii) With the airplane trimmed at $1.4V_S$, the flaps should be retracted to the full up position while maintaining approximately $1.4V_S$ for the flap position existing at each instant throughout the maneuver. The longitudinal control force should not exceed 50 lbs. throughout the maneuver without changing the trim control.

(4) Longitudinal control, power application, § 25.145(b)(4) & (5).

(i) Configuration:

(A) Maximum landing weight.

(B) Critical c.g. position.

(C) Wing flaps retracted and extended to the maximum landing position.

(D) Landing gear extended.

(E) Engine power at flight idle.

(ii) Test procedure: The airplane should be trimmed at a speed of $1.4V_S$. Takeoff power should be applied quickly while maintaining the speed of $1.4V_S$. The longitudinal control force should not exceed 50 lbs. throughout the maneuver without changing the trim control.

(5) Longitudinal control, airspeed variation, § 25.145(b)(6).

(i) Configuration:

(A) Maximum landing weight.

(B) Most forward c.g. position.

(C) Wing flaps extended to the maximum landing position.

(D) Landing gear extended.

(E) Engine power at flight idle.

(ii) Test Procedure: The airplane should be trimmed at a speed of $1.4V_S$. The speed should then be reduced to $1.1V_S$ and then increased to $1.7V_S$, or the flap placard speed, V_{FE}, whichever is lower. The longitudinal control force should not be greater than 50 lbs. Data from the static longitudinal stability tests in the landing configuration at forward c.g., § 25.175(d), may be used to show compliance with this requirement.

(6) Longitudinal control, flap retraction and power application, § 25.145(c).

(i) Configuration:

(A) Critical combinations of maximum landing weights and altitudes.

(B) Critical c.g. position.

(C) Wing flaps extended to the maximum landing position and gated position, if applicable.

(D) Landing gear extended.

(E) Engine power for level flight at a speed of $1.1V_S$ for propeller driven airplanes, or $1.2V_S$ for turbojet powered airplanes.

(ii) Test procedure: With the airplane stable in level flight at a speed of $1.1V_S$ for propeller driven airplanes, or $1.2V_S$ for turbojet powered airplanes, the flaps should be retracted to the full up position, or next gated position, while simultaneously applying inflight takeoff power. The power used should be critical with respect to controllability or performance. It should be possible to prevent any loss of altitude without exceptional piloting skill. Trimming throughout this maneuver is permissible. If gates are provided, this test should be conducted from the maximum landing flap position to the first gate, from gate to gate, and from the last gate to the fully retracted position. The gate design requirement is specified in the rule. The landing gear should remain extended throughout the test.

22. DIRECTIONAL AND LATERAL CONTROL - § 25.147.

a. Explanation.

(1) Sections 25.147(a) and (b) are intended to be investigated for dangerous characteristics such as rudder lock or loss of directional control with one or two critical engines inoperative. Sudden heading changes of up to 15 degrees are required unless the rudder force limit of 150 lbs. (180 lbs. prior to Amendment 25-42) is reached. If the rudder reaches full travel without attaining 150 lbs. force limit or a 15-degree heading change, satisfactory controllability must be demonstrated with this configuration for expected service operations. After full rudder is reached, heading changes using lateral control are permissible provided that no more than a 5-degree bank angle is required.

(2) Sections 25.147(a) and (b) are written to show an airplane will still be under control if yawed suddenly toward and against inoperative engine(s). Paragraphs (c) and (d) require an airplane to be easily controllable with critical inoperative engine(s). Roll response, § 25.147(e), should be satisfactory for takeoff, approach, landing, and high speed configurations. Any permissible configuration which could affect roll response should be evaluated.

b. Procedures.

(1) Directional Control - General, § 25.147(a).

(i) Configuration:

(A) Maximum landing weight.

(B) Most aft c.g. position.

(C) Wing flaps extended to the approach position.

(D) Landing gear retracted.

(E) Yaw SAS on, and off if applicable.

(F) Operating engine(s) at the power for level flight at $1.4V_S$, but not more than maximum continuous power.

(G) Inoperative engine that would be most critical for controllability, with propeller feathered, if applicable.

(ii) Test Procedure: The airplane should be trimmed in level flight at the most critical altitude in accordance with § 25.21(c). Reasonably sudden changes in heading to the left and right, using ailerons to maintain approximately wings level flight, should be made demonstrating a change up to 15 degrees or at which 150 lbs. rudder force is required. The airplane should be controllable and free from any hazardous characteristics during this maneuver.

(2) Directional Control - Four or More Engines, § 25.147(b).

(i) Configuration:

(A) Maximum landing weight.

(B) Most forward c.g. position.

(C) Wing flaps in the most favorable climb position (normally retracted).

(D) Landing gear retracted.

(E) Yaw SAS on, and off if applicable.

(F) Operating engines at the power required for level flight at $1.4V_{S1}$, but not more than maximum continuous power.

(G) Two inoperative engines that would be more critical for controllability with (if applicable) propellers feathered.

(ii) Test Procedure: The procedure outlined in subparagraph b(1)(ii) above is applicable to this test.

(3) Lateral Control - General, § 25.147(c).

(i) Configuration:

(A) Maximum takeoff weight.

(B) Most aft c.g. position.

(C) Wing flaps in the most favorable climb position.

(D) Landing gear retracted and extended.

(E) Yaw SAS on, and off if applicable.

(F) Operating engine(s) at maximum continuous power.

(G) The inoperative engine that would be most critical for controllability, with the propeller feathered, if applicable.

(ii) Test Procedure: With the airplane trimmed at $1.4V_S$, turns with a bank angle of 20 degrees should be demonstrated with and against the inoperative engine from a steady climb at $1.4V_{S1}$. It should not take exceptional piloting skill to make smooth, predictable turns.

(4) Lateral Control - Four or More Engines, § 25.147(d).

(i) Configuration:

(A) Maximum takeoff weight.

(B) Most aft c.g. position.

(C) Wing flaps in the most favorable climb position.

(D) Landing gear retracted and extended.

(E) Yaw SAS on, and off if applicable.

(F) Operating engines at maximum continuous power.

(G) Two inoperative engines most critical for controllability, with propellers feathered, if applicable.

(ii) Test Procedure: The procedure outlined in subparagraph b(3)(ii) is applicable to this test.

(5) Lateral Control - All Engines Operating, § 25.147(e).

(i) Configuration: All configurations within the flight envelope for normal operation.

(ii) Test Procedure: This is primarily a qualitative evaluation which should be conducted throughout the test program. Roll performance should be investigated throughout the flight envelope, including speeds to V_{FC}/M_{FC}, to ensure adequate peak roll rates for safety, considering the flight condition, without excessive control force or travel. Roll response during sideslips expected in service should provide enough maneuvering capabilities adequate to recover from such conditions. Approach and landing configurations should be carefully evaluated to ensure adequate control to compensate for gusts and wake turbulence while in close proximity to the ground.

23. MINIMUM CONTROL SPEED - § 25.149.

a. Explanation. Section 25.149 defines requirements for minimum control speeds during takeoff climb (V_{MC}), during takeoff ground roll (V_{MCG}), and during landing approach (V_{MCL}). The V_{MC} (commonly referred to as V_{MCA}) requirements are specified in §§ 25.149(a), (b), (c) and (d); the V_{MCG} requirements are described in § 25.149(e); and the V_{MCL} requirements are covered in §§ 25.149(f), (g) and (h). Section 25.149(a) states that the method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability in service. That is, the thrust loss from the inoperative engine must be at the rate that would occur if an engine suddenly became inoperative in service. Prior to Amendment 25-42 to § 25.149, the regulation required that rudder control forces must not exceed 180 lbs. Amendment 25-42 limits rudder control forces to 150 lbs. The relationship between V_{EF}, V_1, and V_{MCG}, including the requirements applicable prior to Amendment 25-42, is discussed in paragraph 10, Takeoff and Takeoff Speeds, and paragraph 11, Accelerate-Stop Distance.

b. Procedures.

(1) Minimum Control Speeds - Air (V_{MCA}).

(i) To comply with the V_{MCA} requirements, the following two conditions must be satisfied: (Separate tests are usually conducted to show compliance with these two requirements.)

(A) The dynamic condition in which control is maintained without exceeding a heading change of 20 degrees.

(B) The stabilized (static) condition where constant heading is maintained without exceeding a 5-degree bank angle.

(ii) Static Test Procedure and Required Data. After establishing the critical inoperative engine, the tests for establishing the minimum control speed may be conducted. Using the configuration specified in § 25.149 with the critical engine inoperative, the remaining engine(s) will be adjusted to maximum takeoff power and/or thrust; the airspeed will be decreased until one of the limiting factors specified in §§ 25.149(b), (c) or (d) is experienced. For airplanes with more than two engines, the inboard engine(s) may be throttled, provided the appropriate yawing moment coefficient (C_N) is maintained. If the maximum power and/or thrust within the approved airplane operating envelope was maintained to the minimum test speed, this speed may be used as the V_{MCA} for the airplane. If, at the option of the applicant, V_{MCA} is to vary with altitude and temperature, the minimum test speed and corresponding thrust may be reduced to an equivalent C_N. From this C_N, V_{MCA} may be calculated to vary with takeoff thrust. If maximum takeoff thrust could not be achieved during this test, the C_N can be used to calculate the V_{MCA} for maximum takeoff thrust. It has been acceptable to extend the thrust 5 percent beyond the test thrust. If V_{MCA} is near or less than V_S for the test airplane, consideration may be given to conducting the test at a more extended flap position. It should be noted, however, that a more extended flap position may produce unconservative results. In the event V_{MCA} is less than stall speed at all usable operational gross weights, demonstration that shows compliance with the V_{MCA} requirements may be shown as follows:

(A) Conduct static V_{MCA} tests using partial rudder deflections to achieve a variation in C_N with rudder deflection.

(B) Plot the asymmetric thrust yawing moment (C_N) versus control surface deflection (lateral and directional). These plots should be faired and then extrapolated to full control surface deflections. Plot C_N versus rudder pedal force. This plot should be faired to 150 lbs. (180 lbs. prior to Amendment 25-42). Whichever condition of the three is most limiting determines the maximum C_N from which V_{MCA} can be calculated.

(C) The extrapolation should be limited to 5 percent of the yawing moment coefficient unless a rigorous analysis is made to account for all of the stability and control terms.

(D) Compute the stalling speed (V_S) at the airplane Operational Weight Empty (OWE) for the maximum takeoff flap position and compute V_{MCA} from C_N using the maximum asymmetric takeoff thrust. If the computed V_{MCA} is less than V_S, then the airplane is stall limited and V_{MCA} is not a factor.

(iii) Dynamic Test Procedure and Required Data. In addition to the static test procedure, dynamic demonstrations should be made to provide adequate proof that the speed(s) determined also meet the dynamic requirements. The dynamic demonstration is conducted by applying the maximum rated power and/or thrust to all engines and suddenly cutting the critical engine. It should be possible to recover to a constant heading without exceeding the requirements of § 25.149(d). If the thrust/weight for the dynamic demonstration produces an extreme nose-high attitude, normally more than 20 degrees, another method should be used such as conducting dynamic demonstrations using a minimum required rudder and aileron control at reduced thrust and comparing control deflection and force required between the dynamic demonstration and static demonstration at several reduced thrust conditions.

(iv) If V_{MCA} has been shown to be less than V_S by the static method, the dynamic demonstration may be conducted at speeds such as $1.1V_S$ and evaluated in accordance with paragraph (iii) above.

(v) Normally, V_{MCA} and V_{MCG} will be determined by rendering the engine inoperative and allowing the propeller to autofeather; however, on some airplanes a more critical drag condition can be produced during a partial power condition. Some engine propeller combinations might be subject to this type of failure. One example is some turbopropeller installations can have a fuel control failure that causes the engine to go to flight idle, resulting in a higher asymmetric drag than that obtained from an inoperative engine. In such a case, the test must be conducted in the most critical condition.

(vi) There may be some difference between right and left engine inoperative V_{MCA} due to propeller slip stream rotation reducing rudder effectiveness to maintain the airplane on its original heading. The critical engine should be determined and the V_{MCA} for that configuration should be used.

(vii) V_{MCA} and V_{MCG} should be based on the maximum net thrust reasonably expected for a production engine. These speeds should not be based on specification thrust since this thrust represents the minimum thrust as guaranteed by the engine manufacturer, and the resulting speeds could be too slow. The thrust used for scheduled V_{MCA} and V_{MCG} speeds should represent the high side of the tolerance band and may be determined by analysis instead of tests.

(2) Minimum Control Speed - Ground (V_{MCG}) - § 25.149(e).

(i) It must be demonstrated that, when the critical engine is suddenly made inoperative at V_{MCG} during the takeoff ground roll, the airplane is safely controllable to continue the takeoff. During the demonstration, the airplane must not deviate more than 30 ft. (25 ft. prior to Amendment 25-42) from the pre-engine-cut projected ground track. The critical engine is determined by the methods as described above under § 25.149(c).

(ii) Tests may be conducted by abruptly retarding the engine to idle to establish the target V_{MCG}. At least one fuel cut should be made at each maximum asymmetric thrust level desired to be certificated to investigate the more rapid thrust decay associated with this type of engine failure. At the applicant's option, in crosswind conditions, the runs may be made on reciprocal headings or an analytical correction may be applied to determine the zero crosswind value of V_{MCG}.

(iii) During determination of V_{MCG}, engine failure recognition should be by pilot sensation or outside reference only, unless an engine failure warning device is installed.

(iv) Control of the airplane should be accomplished by use of the rudder only. All other controls, like ailerons and spoilers, should only be used to correct any alterations in the airplane attitude and to maintain a wings level condition. Use of those controls to supplement the rudder effectiveness should not be allowed.

(v) The V_{MCG} should be considered at the heaviest weight where V_{MCG} may impact the AFM V_1 schedule.

(vi) The test should be conducted at aft c.g. and with the nose wheel free to caster, to minimize the stabilizing effect of the nose gear.

(vii) For airplanes with certification basis prior to Amendment 25-42, V_{MCG} values may be demonstrated with nose wheel rudder pedal steering operative for dispatch on wet runways. The test should be conducted on an actual wet runway. The test(s) should include engine failure at or near a minimum V_{EF} associated with minimum V_R to demonstrate adequate controllability during during rotation, liftoff, and the initial climbout. The V_{MCG} values obtained by this method are applicable for wet or dry runways only, not for icy runways.

(3) Minimum Control Speed During Landing Approach V_{MCL} - § 25.149(f).

(i) This section is intended to cover the controllability aspects of an engine failure during landing approach. Section 25.149(f) requires that minimum control speeds during landing approach with all engines operating at maximum available inflight power or thrust be determined when the critical engine is suddenly made inoperative. V_{MCL} is defined as that speed where it is possible to recover control of the airplane with the engine still inoperative and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL} is the minimum control speed for the situation where an engine fails after power or thrust has been increased to make a go-around from an approach with all engines operating.

(ii) The initial power condition at the time of engine failure should be the maximum available inflight takeoff thrust. The procedures given in paragraph (1) for V_{MCA} may be applied in determining V_{MCL}, except flap settings and trim settings should be appropriate to the maximum approach flap used to show compliance with § 25.121(d).

(4) <u>Minimum Control Speed with Two Inoperative Engines During Landing Approach (V_{MCL-2}) - § 25.149(g)</u>.

(i) For airplanes with three or more engines, V_{MCL-2} is the minimum speed for maintaining safe control during the power or thrust changes that are likely to be made following the failure of a second engine during an approach initiated with one engine inoperative.

(ii) This test should be conducted in the most critical one-engine-inoperative approach or landing configuration (from the AFM), usually the minimum flap deflection. Two demonstrations are required to determine V_{MCL-2}.

(A) With power on the operating engines set to maintain a -3 degree glideslope, with one critical engine inoperative, the second critical engine is made inoperative and the remaining operating engine(s) advanced to maximum available inflight takeoff power. The V_{MCL-2} speed is established by the procedures presented in paragraphs (1)(i) and (ii) for V_{MCA}.

(B) With power on the operating engines set to maintain a -3 degree glideslope, with one critical engine inoperative:

(1) Set the airspeed at the value determined above in step (A) and, with zero bank angle, maintain a constant heading using trim to reduce the control force to zero. If full trim is insufficient to reduce the control force to zero, full trim should be used plus control deflection as required; and

(2) Make the second critical engine inoperative and retard the remaining operating engine(s) to minimum available power without changing the directional trim. The V_{MCL-2} determined in paragraph (A) is acceptable if constant heading can be maintained without exceeding a 5-degree bank angle and the limiting conditions of § 25.149(h).

FAR REQUIREMENT

Controllability and Maneuverability

§ 25.143 **General.**

(a) The airplane must be safely controllable and maneuverable during—
(1) Takeoff;
(2) Climb;
(3) Level flight;
(4) Descent; and
(5) Landing.

(b) It must be possible to make a smooth transition from one flight condition to any other flight condition without exceptional piloting skill, alertness, or strength, and without danger of exceeding the airplane limit-load factor under any probable operating conditions, including—

(1) The sudden failure of the critical engine;

(2) For airplanes with three or more engines, the sudden failure of the second critical engine when the airplane is in the en route, approach, or landing configuration and is trimmed with the critical engine inoperative; and

(3) Configuration changes, including deployment or retraction of deceleration devices.

(c) If, during the testing required by paragraphs (a) and (b) of this section, marginal conditions exist with regard to required pilot strength, the "strength of pilots" limits may not exceed the limits prescribed in the following table:

Values in pound of force as applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
For temporary application	75	60	150
For prolonged application	10	5	20

JAR REQUIREMENT

(c) If, during the testing required by sub-paragraphs (a) and (b) of this paragraph marginal conditions exist with regard to required pilot strength, the 'strength of pilots' limits for conventional wheel type controls may not exceed the limits prescribed in the following table. (See ACJ 25.143 (c)):

Values in pounds of force as applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
For temporary application for pitch and roll control — two hands available for control	75	50	—
For temporary application for pitch and roll control — one hand available for control	50	25	—
For temporary application for yaw control	—	—	150
For prolonged application	10	5	20

(f) When manoeuvring at a constant air speed or Mach number (up to V_{MO}/M_{MO}) the stick forces and the gradient of the curve of stick force versus manoeuvring load factor must lie within satisfactory limits. The stick forces must not be so great as to make excessive demands on the pilot's strength [when manoeuvring the aeroplane (see ACJ No.1 to JAR 25.143 (f)), and must not be so low that the] aeroplane can easily be overstressed inadvertently. Changes of gradient which occur with changes of load factor must not cause undue difficulty in maintaining control of the aeroplane, and local gradients must not be so low as to result in a danger [of over-controlling. (See ACJ No. 2 to JAR 25.143 (f).)]

FAR REQUIREMENT

§ 25.145 Longitudinal control.

(a) It must be possible at any speed between the trim speed prescribed in § 25.49(c)(2)(i) and V_{S_1} (for reciprocating engine powered airplanes), or at any speed between the trim speed prescribed in § 25.103(b)(1) and V_S (for turbine engine powered airplanes), to pitch the nose downward so that the acceleration to this selected trim speed is prompt with—

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JAR REQUIREMENT

JAR 25.145 Longitudinal control

(a) It must be possible at any speed between the trim speed prescribed in JAR 25.103 (a) (5) and Vs, to pitch the nose downward so that the acceleration to this selected trim speed is prompt with —

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(d) During the climb after take-off with one engine inoperative and landing gear retracted, it must be possible in conditions of moderate atmospheric turbulence, to manoeuvre the aeroplane in a manner appropriate to the phase of flight without encountering natural or artificial stall warning. (See ACJ 25.145 (d).)

FAR REQUIREMENT

§ 25.147 Directional and lateral control.

(a) *Directional control; general.* It must be possible, while holding the wings approximately level, to safely make reasonably sudden changes in heading in both directions. This must be shown at 1.4 V_{S1} for heading changes up to 15° (except that the heading change at which the rudder pedal force is 150 pounds need not be exceeded), and with—

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JAR REQUIREMENT

JAR 25.147 Directional and lateral control

(a) *Directional control; general.* It must be possible, while holding the wings approximately level, to safely make reasonably sudden changes in heading in both directions. This must be shown at 1·4 V_{S1} for heading changes up to 15° (except that the heading change at which the rudder pedal force is 150 pounds or the rudder pedal reaches its stops, need not be exceeded), and with —

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(2) With the critical engine inoperative, roll response must allow normal manoeuvres. Lateral control must be sufficient, at the speeds likely to be used with one engine inoperative for climb, cruise, descent and landing approach, to provide a peak roll rate necessary for safety without excessive control forces or travel. (See ACJ 25.147 (c) (2).)

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(e) *Lateral control; all engines operating.* With the engines operating, roll response must allow normal manoeuvres (such as recovery from upsets produced by gusts and the initiation of evasive manoeuvres). There must be enough excess lateral control in sideslips (up to sideslip angles that might be required in normal operation), to allow a limited amount of manoeuvring and to correct for gusts. Lateral control must be enough at any speed up to V_{FC}/M_{FC} to provide a peak roll rate necessary for safety, without excessive control forces or travel. (See ACJ 25.147 (e).)

ACJ 25.147(c)(2)
Lateral Control: One Engine Inoperative (Interpretative Material)
See JAR 25.147(c)(2)

An acceptable method of demonstrating compliance with JAR 25.147(c)(2) is as follows:

It should be possible in the conditions specified below to roll the aeroplane from a steady 30° banked turn through an angle of 60° so as to reverse the direction of the turn in not more than 11 seconds. In this demonstration the rudder may be used to the extent necessary to minimise sideslip. The demonstration should be made rolling the aeroplane in either direction, and the manoeuvre may be unchecked.

Conditions: Airspeed. V_2.

Wing-flaps. In each take-off position.

Landing Gear. Retracted.

Power. The critical engine inoperative and its propeller (if applicable) in the minimum drag condition; the remaining engines operating at maximum take-off power.

Trim. The aeroplane should be in trim, or as nearly as possible in trim, for straight flight in these conditions, and the trimming controls should not be moved during the manoeuvre.

ACJ 25.147(e)
Lateral Control: All Engines Operating (Interpretative Material)
See JAR 25.147(e)

An acceptable method of demonstrating that roll response and peak roll rates are adequate for compliance with JAR 25.147(e) is as follows:

It should be possible in the conditions specified below to roll the aeroplane from a steady 30° banked turn through an angle of 60° so as to reverse the direction of the turn in not more than 7 seconds. In these demonstrations the rudder may be used to the extent necessary to minimise sideslip. The demonstrations should be made rolling the aeroplane in either direction, and the manoeuvres may be unchecked.

Conditions:

(a) En-route: Airspeed. All speeds between the minimum value of the scheduled all-engines-operating climb speed and VMO/MMO.

Wing-flaps. En-route position(s).

Air Brakes. All permitted settings from Retracted to Extended.

Landing Gear. Retracted.

Power. All engines operating at all powers from flight idle up to maximum continuous power.

Trim. The aeroplane should be in trim from straight flight in these conditions, and the trimming controls should not be moved during the manoeuvre.

(b) Approach: Airspeed. Either the speed maintained down to the 50 ft height in compliance with JAR 25.125(a)(2), or the target threshold speed determined in accordance with JAR 25.125(c)(2)(i) as appropriate to the method of landing distance determination used.

Wing-flaps. In each landing position.

Air Brakes. In the maximum permitted extended setting.

Landing Gear. Extended.

Power. All engines operating at the power required to give a gradient of descent of 5·0%.

Trim. The aeroplane should be in trim for straight flight in these conditions, and the trimming controls should not be moved during the manoeuvre.

FAR REQUIREMENT	JAR REQUIREMENT
§ 25.149 Minimum control speed.	**JAR 25.149 Minimum control speed** (See ACJ 25.149)
(a) In establishing the minimum control speeds required by this section, the method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability expected in service.	(a) In establishing the minimum control speeds required by this paragraph, the method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability expected in service.
(b) V_{MC} is the calibrated airspeed, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than five degrees.	(b) VMC is the calibrated airspeed, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the aeroplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5°.
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(e) V_{MCG}, the minimum control speed on the ground, is the calibrated airspeed during the takeoff run, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with the use of primary aerodynamic controls alone (without the use of nosewheel steering) to enable the takeoff to be safely continued using normal piloting skill and rudder control forces not exceeding 150 pounds. In the determination of V_{MCG}, assuming that the path of the airplane accelerating with all engines operating is along the centerline of the runway, its path from the point at which the critical engine is made inoperative to the point at which recovery to a direction parallel to the centerline is completed may not deviate more than 30 feet laterally from the centerline at any point. V_{MCG} must be established with—	(e) VMCG, the minimum control speed on the ground, is the calibrated airspeed during the take-off run, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the aeroplane with the use of primary aerodynamic controls alone (without the use of nose-wheel steering) to enable the take-off to be safely continued using normal piloting skill and rudder control forces not exceeding 150 pounds. In the determination of VMCG, assuming that the path of the aeroplane accelerating with all engines operating is along the centreline of the runway, its path from the point at which the critical engine is made inoperative to the point at which recovery to a direction parallel to the centreline is completed may not deviate more than 30 ft laterally from the centreline at any point. VMCG must be established with —
o o o	o o o

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FAR REQUIREMENT

(f) V_{MCL}, the minimum control speed during landing approach with all engines operating, is the calibrated airspeed at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL} must be established with—

(1) The airplane in the most critical configuration for approach with all engines operating;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with all engines operating;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL}); and

(5) Maximum available takeoff power or thrust on the operating engines.

JAR REQUIREMENT

(f) V_{MCL}, the minimum control speed during landing approach with all engines operating, must be established with the aeroplane in the most critical configuration for approach and landing with all engines operating and the aeroplane trimmed for approach and landing with all engines operating. V_{MCL} may not be lower than the calibrated airspeed at which it has been demonstrated that it is possible—

(1) With the critical engine inoperative and the operating engines developing maximum take-off power or thrust—

(i) To maintain straight flight with an angle of bank of not more than 5°; and

(ii) Starting from this condition of steady straight flight to roll the aeroplane through an angle of 20°, in the direction necessary to initiate a turn away from the inoperative engine in not more than 5 seconds; and

(2) With the critical engine inoperative and the operating engines developing the power or thrust necessary to maintain a gradient of descent of 5% and starting from a condition of steady flight, to roll the aeroplane through an angle of 20° in the direction necessary to initiate a turn away from the inoperative engine, in not more than 3·5 seconds; and

(3) With all engines initially operating at maximum take-off power and the critical engine suddenly made inoperative, to prevent excessive changes of bank angle or heading and to recover control of the aeroplane without reducing power of the remaining engines.

FAR REQUIREMENT

(g) For airplanes with three or more engines, V_{MCL-2}, the minimum control speed during landing approach with one critical engine inoperative, is the calibrated airspeed at which, when a second critical engine is suddenly made inoperative, it is possible to recover control of the airplane with both engines still inoperative and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL-2}, must be established with—

(1) The airplane in the most critical configuration for approach with the critical engine inoperative;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with the critical engine inoperative;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL-2});

(5) The power or thrust on the operating engines required to maintain an approach path angle of 3 degrees when one critical engine is inoperative; and

(6) The power or thrust on the operating engines rapidly changed, immediately after the second critical engine is made inoperative, from the power or thrust prescribed in paragraph (g)(5) of this section to—

(i) Minimum available power or thrust; and

(ii) Maximum available takeoff power or thrust.

JAR REQUIREMENT

(g) V_{MCL-1}, the minimum control speed during landing approach with one engine inoperative, must be established with the aeroplane in the most critical configuration for approach and landing with one engine inoperative and the aeroplane trimmed as recommended for approach and landing with the critical engine inoperative. V_{MCL-1} may not be lower than the calibrated airspeed for which it has been demonstrated that it is possible —

(1) With the critical engine inoperative and the operating engines developing maximum take-off power or thrust:—

(i) To maintain straight flight with an angle of bank of not more than 5°; and

(ii) Starting from this condition of steady straight flight to roll the aeroplane through an angle of 20°, in the direction necessary to initiate a turn away from the inoperative engine, in not more than 5 seconds; and

(2) With the critical engine inoperative and the operating engines developing the power or thrust necessary to maintain a gradient of descent of 5%, and starting from a condition of steady straight flight, to roll the aeroplane through an angle of 20°, in the direction necessary to initiate a turn away from the inoperative engine, in not more than 3·5 seconds.

FAR REQUIREMENT

(h) The rudder control forces required to maintain control at V_{MCL} and V_{MCL-2} may not exceed 150 pounds, nor may it be necessary to reduce the power or thrust of the operating engines. In addition, the airplane may not assume any dangerous attitudes or require exceptional piloting skill, alertness, or strength to prevent a divergence in the approach flight path that would jeopardize continued safe approach when—

(1) The critical engine is suddenly made inoperative; and

(2) For the determination of V_{MCL-2}, the power or thrust on the operating engines is changed in accordance with paragraph (g)(6) of this section.

JAR REQUIREMENT

(h) For aeroplanes with three or more engines, V_{MCL-2}, the minimum control speed during landing approach with two critical engines inoperative, must be established with the aeroplane in the recommended configuration for approach with two engines inoperative and the aeroplane trimmed as recommended for approach with two engines inoperative. V_{MCL-2} may not be lower than the lowest calibrated airspeed at which it has been demonstrated that it is possible —

(1) With the critical two engines inoperative and the operating engines developing maximum take-off power or thrust —

(i) To maintain straight flight with an angle of bank of not more than 5°; and

(ii) Starting from this condition of steady straight flight to roll the aeroplane through an angle of 20°, in the direction necessary to initiate a turn away from the inoperative engines, in not more than 5 seconds; and

(2) With the two critical engines inoperative and the operating engines developing the power or thrust necessary to maintain a gradient of descent of 5%, and starting from a condition of steady straight flight, to roll the aeroplane through an angle of 20° in the direction necessary to initiate a turn away from the inoperative engines, in not more than 3·5 seconds.

(i) In demonstrations of the speeds V_{MCL}, V_{MCL-1} and V_{MCL-2} —

(1) The rudder force may not exceed 150 lb and the lateral control force may not be excessive;

(2) The control of the aeroplane may not require exceptional piloting skill, alertness or strength;

(3) The aeroplane must be loaded to the most unfavourable centre of gravity; and

(4) The aeroplane must be loaded to the most unfavourable weight, or at the option of the applicant, the speeds may be determined and scheduled as a function of weight.

FAR REQUIREMENT

§ 25.161 Trim.

(a) *General.* Each airplane must meet the trim requirements of this section after being trimmed, and without further pressure upon, or movement of, either the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) *Lateral and directional trim.* The airplane must maintain lateral and directional trim with the most adverse lateral displacement of the center of gravity within the relevant operating limitations, during normally expected conditions of operation (including operation at any speed from 1.4 V_{S1} to V_{MO}/M_{MO}).

(c) *Longitudinal trim.* The airplane must maintain longitudinal trim during—

(1) A climb with maximum continuous power at a speed not more than 1.4 V_{S1}, with the landing gear retracted, and the flaps (i) retracted and (ii) in the takeoff position;

(2) A glide with power off at a speed not more than 1.4 V_{S1}, with the landing gear extended, the wing flaps (i) retracted and (ii) extended, the most unfavorable center of gravity position approved for landing with the maximum landing weight, and with the most unfavorable center of gravity position approved for landing regardless of weight; and

JAR REQUIREMENT

JAR 25.161 Trim

(a) *General.* Each aeroplane must meet the trim requirements of this paragraph after being trimmed, and without further pressure upon, or movement of, either the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) *Lateral and directional trim.* The aeroplane must maintain lateral and directional trim with the most adverse lateral displacement of the centre of gravity within the relevant operating limitations, during normally expected conditions of operation (including operation at any speed from 1·4 V_{S1} to V_{MO}/M_{MO}).

(c) *Longitudinal trim.* The aeroplane must maintain longitudinal trim during —

(1) A climb with maximum continuous power at a speed not more than 1·4 V_{S1}, with the landing gear retracted, and the wing-flaps (i) retracted and (ii) in the take-off position;

(2) Either a glide with power off at a speed not more than 1·4 V_{S1}, or an approach within the normal range of approach speeds appropriate to the weight and configuration with power settings corresponding to a 3° glidepath, whichever is the most severe, with the landing gear extended, the wing-flaps (i) retracted and (ii) extended, the most unfavourable centre of gravity position ~~approved for landing with the~~ maximum landing weight, and with the most unfavourable centre of gravity position approved for landing regardless of weight; and

FAR REQUIREMENT

§ 25.253 High-speed characteristics.

(a) *Speed increase and recovery characteristics.* The following speed increase and recovery characteristics must be met:

(1) ~~Operating conditions and charac~~teristics likely to cause inadvertent speed increases (including upsets in pitch and roll) must be simulated with the airplane trimmed at any likely cruise speed up to V_{MO}/M_{MO}. These conditions and characteristics include gust upsets, inadvertent control movements, low stick force gradient in relation to control friction, passenger movement, leveling off from climb, and descent from Mach to airspeed limit altitudes.

(2) Allowing for pilot reaction time after effective inherent or artificial speed warning occurs, it must be shown that the airplane can be recovered to a normal attitude and its speed reduced to V_{MO}/M_{MO}, without-

(i) Exceptional piloting strength or skill;

(ii) Exceeding V_D/M_D, V_{DF}/M_{DF}, or the structural limitations; and

(iii) Buffeting that would impair the ~~pilot's ability to read the instruments~~ or control the airplane for recovery.

(3) There may be no control reversal about any axis at any speed up to V_{DF}/M_{DF}. Any reversal of elevator control force or tendency of the airplane to pitch, roll, or yaw must be mild and readily controllable, using normal piloting techniques.

JAR REQUIREMENT

JAR 25.253 High-speed characteristics

(a) *Speed increase and recovery characteristics.* The following speed increase and recovery characteristics must be met:

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(3) There may be no control reversal about any axis at any speed up to VDF/MDF. Any reversal of elevator control force or tendency of the aeroplane to pitch, roll, or yaw must be mild and readily controllable, using normal piloting techniques. Adequate roll capability to assure a prompt recovery from a laterally upset condition must be available. (See ACJ 25.253 (a) (3).)

(4) [Reserved.]

(5) *Trim change due to airbrake selection.* With the aeroplane trimmed at VMO/MMO, extension of the airbrakes at speeds above VMO/MMO, over the available range of movements of the pilots control must not result in an excessive positive load factor with the stick free, and any nose-down pitching moment must be small. (See ACJ 25.253 (a) (5).)

(6) When the aeroplane is trimmed at VMO/MMO —

(i) The slope of the stick force versus speed curve need not be positive at all points but there must be a push force at all speeds up to VDF/MDF.

(ii) There must be no sudden or excessive reduction of stick force as speed approaches VDF/MDF.

ACJ 25.253(a)(3)
Lateral Control at Speeds in Excess of V_{MO}/M_{MO} (Interpretative Material)
See JAR 25.253(a)(3)

An acceptable method of demonstrating that roll capability is adequate to assure prompt recovery from a laterally upset condition is as follows:

It should be possible using lateral control alone to roll the aeroplane from a steady 20° banked turn through an angle of 40° so as to reverse the direction of the turn in not more than 8 seconds. The demonstration should be made rolling the aeroplane in either direction in the conditions specified below. The manoeuvres may be unchecked.

Conditions: Air Speed. All speeds from V_{MO}/M_{MO} to a speed close to V_{DF}/M_{DF} but limited to the extent necessary to accomplish the manoeuvre and recovery without exceeding V_{DF}/M_{DF}.

Wing-flaps. En-route position(s).

Air Brakes. All permitted settings from retracted to extended.

Landing Gear. Retracted.

Power (i) All engines operating at the power required to maintain level flight at V_{MO}/M_{MO}, except that maximum continuous power need not be exceeded; and

(ii) If the effect of power is significant, with the throttles closed.

Trim. The aeroplane trimmed for straight flight at V_{MO}/M_{MO}. The trimming controls should not be moved during the manoeuvre.

In addition it should be demonstrated that use of rudder in the conventional sense, if it results in an adverse effect on roll rate, will not result in a dangerous reduction in lateral control capability.

ACJ 25.253(a)(5)
High Speed Characteristics (Interpretative Material)
See JAR 25.253(a)(5)

1 *Trim Change due to Airbrake Selection*

1.1 A load factor should be regarded as excessive if it exceeds 2·0 with stick free.

1.2 A nose-down pitching moment should be regarded as small if it necessitates a stick-force of less than 20 lb to maintain 1 g flight.

1.3 Compliance with the requirements of JAR 25.253(a)(5) should be demonstrated at speeds up to a speed close to V_{DF}/M_{DF} but limited to the extent necessary to accomplish the manoeuvre and recovery without exceeding V_{DF}/M_{DF}.

Session VII (last)

AEROELASTICITY

1. AEROELASTICITY

Large high speed airplanes are significantly flexible - Aerodynamic performance and stability and control affected.

We'll consider: o Wing o Aft body o Horizontal tail o Vertical tail

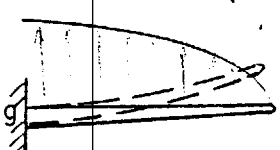
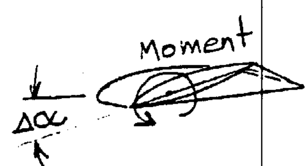
Elastic deformation comes from: o aerodynamic loads
o inertia loads

Important factors: Aerodynamic shape - AR, Λ, λ, t/c, fineness ratio
Structural stiffness - EI, GJ
Mass distribution

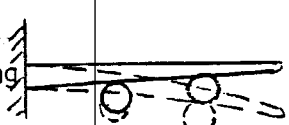
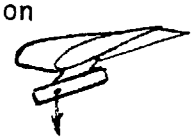
Key points:
1) Aero loads and structural deflections are <u>interdependent</u>
2) Rigid body and structural modes usually well separated in <u>frequency</u> - lowest structural frequency order of 1 or 2 cps - .5 to 1 sec period - highest rigid body mode order of 8 - 10 seconds long. short period - This leads to expressing aeroelastic corrections as quasi-static.
3) Common form of aeroelastic correction is an ELASTIC/RIGID ratio

<u>Development of Concepts</u>

o Aerodynamic loads: (zero sweep first)

Bending  Torsion 

o Inertia loads:

Bending  Torsion 

Applying simple beam theory:

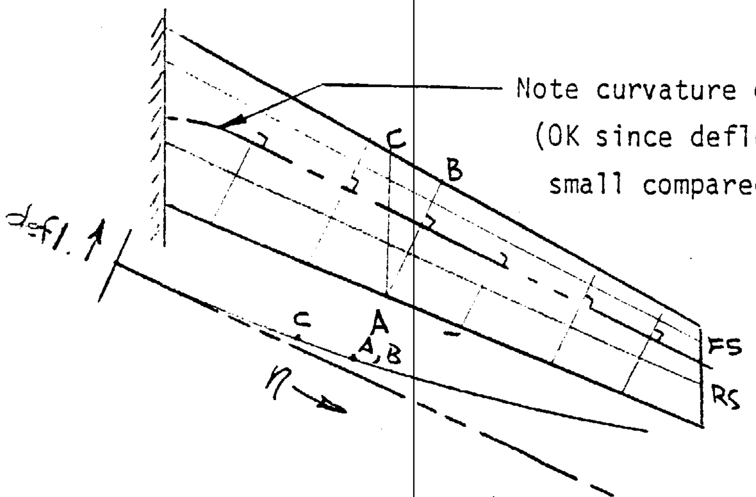
- Concept of elastic axis allows "decoupling" of bending and torsion
- Chordwise deformations that change camber are usually small

(Simple beam theory not valid for low AR, high Λ like SST's)

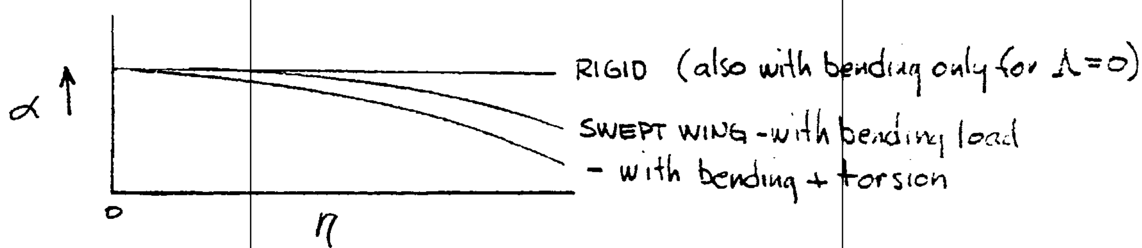
Side track for concept of elastic axis:

- Good for high AR, moderate sweep
- Bending & torsion vectors mutually $\perp$ (along & normal to EA)
- Bending moment results in translation of sections normal to EA
- Torsional moment results in pure rotation of sections normal to EA

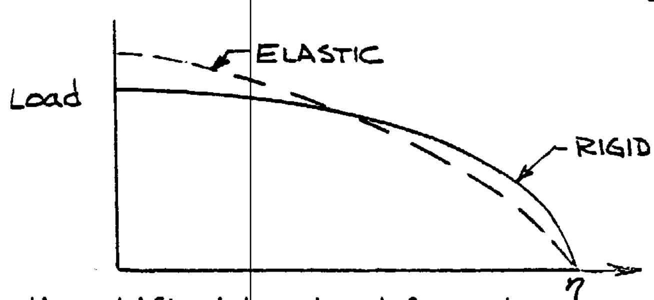
Consider a moderate sweep, high $\mathcal{R}$ wing of conventional planform:



Since deflection at C < deflection at A and B, wing "sees" spanwise twist, wing washes out and unloads at the tip.



What does this do to the load? (Remember interdependence)



Loading shifts inboard and forward.

But what about inertia load? Tends to be relieving -

Because of this relief, wings with engines tend to be more flexible.

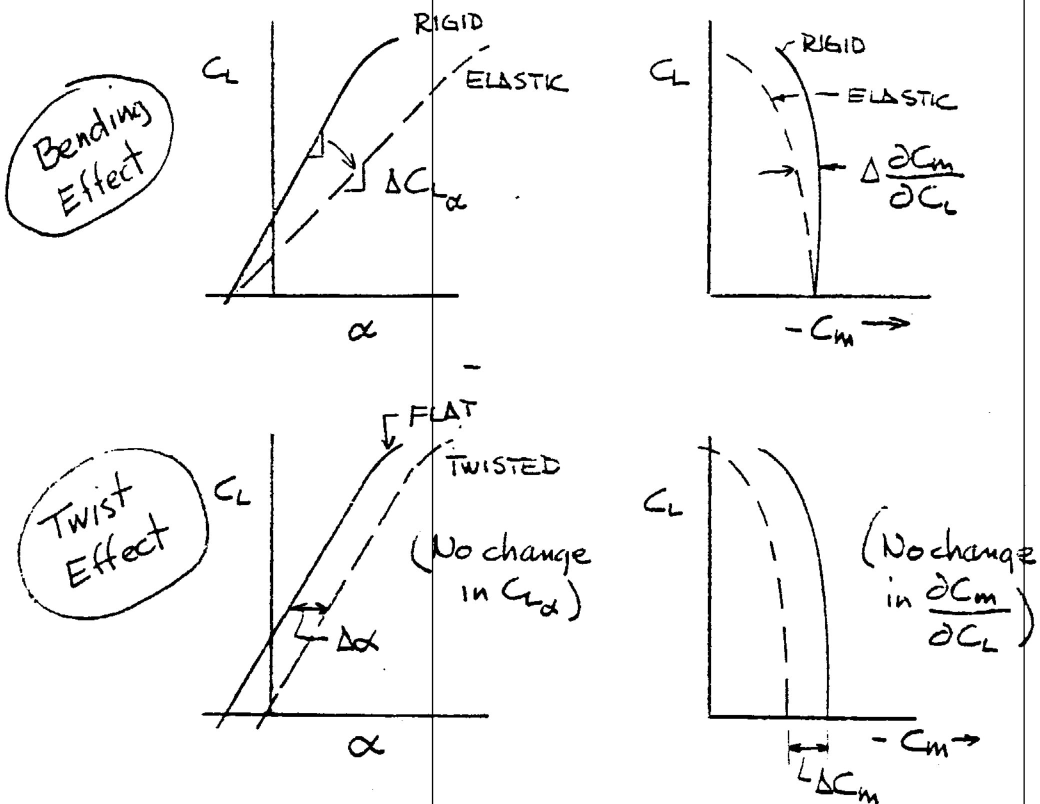
Fuel in wings also beneficial as is any payload in wings.

Wing cruise shape can be preselected via jig twist - primarily "1 g".
Shape that results from maneuvers ($n_z \neq 1.0$) we have to live with.

#9

REV SYM B

What is overall effect for the wing?



Note: Wind tunnel models are currently constructed with average cruise "1 g" twist. Putting it all together for wing body:

$$C_{L_{\alpha_{FLEX}}} = C_{L_{\alpha_{RIGID}}} \left(\frac{L_E}{L_R}\right)_{WW}$$

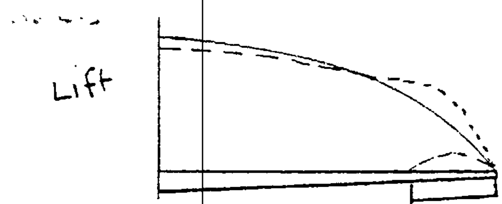
$$C_{m_{\alpha_{FLEX}}} = \left[\left(\frac{\partial C_m}{\partial C_L}\right)_{RIGID} + \Delta \frac{\partial C_m}{\partial C_L}\right] C_{L_{\alpha_{RIGID}}} \left(\frac{L_E}{L_R}\right)_{WW}$$

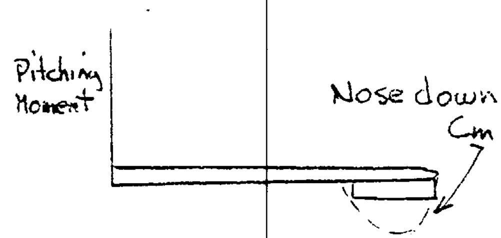
Inertia effects are usually accounted for by CL_{n_Z} and $C_{m_{n_Z}}$. This is done to keep track of "1 g" characteristics like for trim and speed stability and flexibility increments in maneuvering flight for $n_Z \neq 1.0$ - man. stability.

DEU SYM B

Next, let's consider control surfaces on the wing-like ailerons.

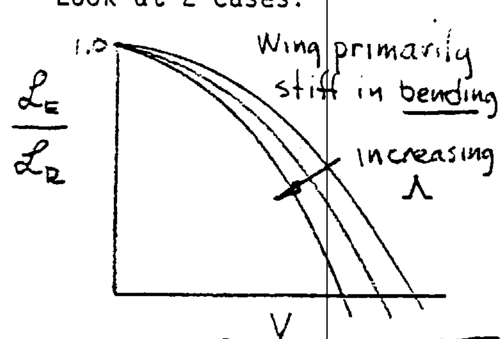
Flap-type surfaces generate local lift: and local pitching moment:

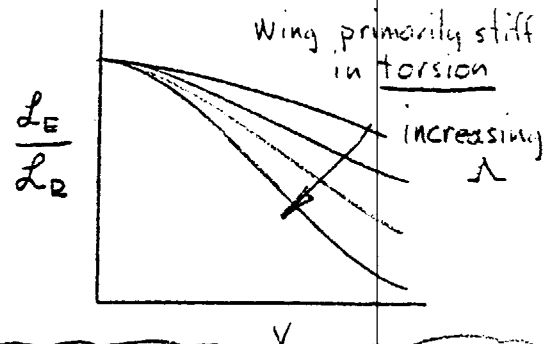




How the wing reacts is very dependent upon its IE & GJ.

Look at 2 cases:





Side track to discuss why aeroelastic affects are expressed as functions of q.

- We assume coefficients good for all speeds
- Loads increase as function of V^2 or q.
- Most analyses center around some fixed M.
- Inertia loads "don't care" about q.

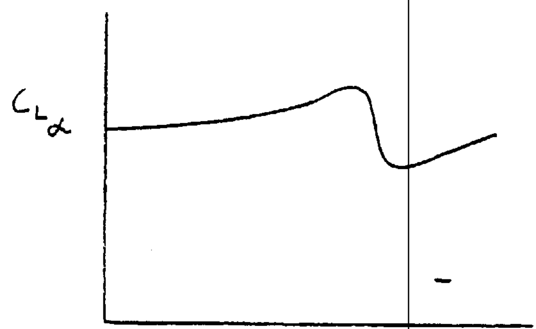
Aileron (control surface) Reversed

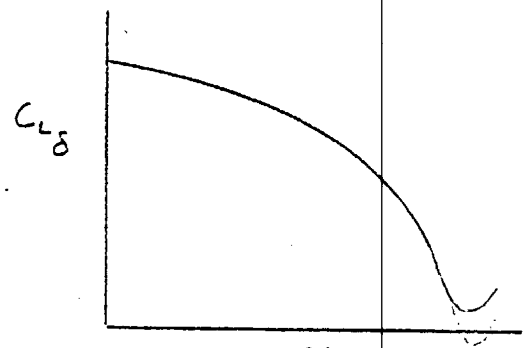
There exists a speed for most wings of the transport class where the combination of reduced effectiveness (due to bending) plus torsion (due to C_{m_δ}) will cause the wing to distort in such a manner that the elastic loading is reversed compared to the rigid.

E.g. 727 outboard aileron reversal test

- Original prediction not too close
- Nearest reversal at M = .86 (compressibility effects relieving for M > .86)

Consider Mach number effects for trailing edge type controls:

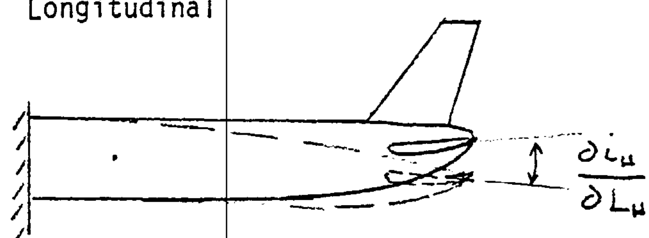




In final analysis, Mach number effects must be considered since a small aeroelastic effect can drive the effectiveness to zero when rigid effectiveness is low.

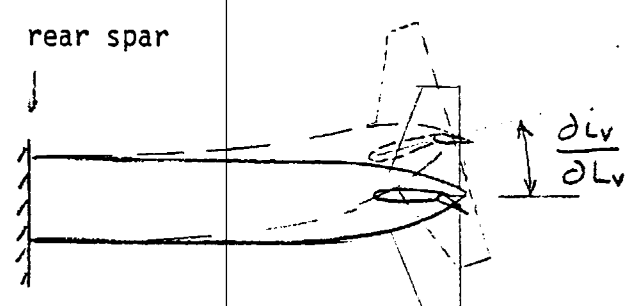
On to <u>aft body</u> flexibility :

Longitudinal



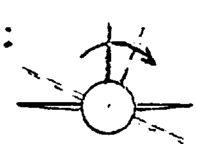
Aero load due to α_H, δ_e
Inertia load due to n_z

rear spar



Aero load due to β , δ_R
Inertia load less important since lateral accelerations are generally small.

Key points -
- Loads due to α_H, β are primarily bending loads.
- Loads due to δ_e, δ_R are bending + torsion
- Torsional loads that twist the body : are not too significant from S&C standpoint.



Also note for T-tails - bending occurs in aft body and fin due to horizontal tail.

Horizontal and Vertical Tail felxibility (See handout)

Just like small wings with full span plain flap

$$C_{L_{\alpha}}{}_{H_{FLEX}} = C_{L_{\alpha}}{}_{H_{RIGID}} \left(\frac{L_E}{L_R}\right)_H$$ tail and aft body flexibility

$$\left(\frac{L_E}{L_R}\right)_H = \frac{\left(\frac{L_E}{L_R}\right)_{RIGID\ BODY}}{1 - \left(\frac{L_E}{L_R}\right)_{H_{RIG.BOD.}} \eta\, C_{L_{\alpha_H}} \frac{\partial i_H}{\partial L_H}\, q\, S_H}$$

Usual form of elastic factors on the following pages.

Note that C_{l_p} and $C_{l_{\dot{p}}}$ can be important factors in roll performance.

6-8243-20-015

AEROELASTIC TERM	DESCRIPTION	ELASTIC STRUCTURE	UNITS	VARIABLES
$\left(\frac{L_E}{L_R}\right)_{WW}$	RATIO OF ELASTIC TO RIGID WING LIFT DUE TO ANGLE OF ATTACK (WEIGHTLESS WING)	WING	—	V_e, M
$\Delta\left(\frac{dC_{m_{.25\bar{c}_W}}}{dC_L}\right)_{WW}$	SHIFT IN WING AERODYNAMIC CENTER DUE TO WING FLEXIBILITY (WEIGHTLESS WING)	WING	FRACTION $\bar{c}_W$	V_e, M
$C_{L_{n_{z_W}}}$	CHANGE IN WING LIFT DUE TO LOAD FACTOR	WING	1/g	V_e, M, AND FUEL LOADING
$C_{m_{n_{z_W}}}$	CHANGE IN WING PITCHING MOMENT DUE TO LOAD FACTOR	WING	1/g	V_e, M, AND FUEL LOADING
$\left(\frac{L_E}{L_R}\right)_{H_{RIGID\ BODY}}$	RATIO OF ELASTIC TO RIGID HORIZONTAL TAIL LIFT DUE TO A LOAD ON HORIZONTAL TAIL (RIGID BODY)	HORIZONTAL TAIL	—	V_e, M
$\frac{\partial i_H}{\partial L_H}$	CHANGE IN INCIDENCE OF HORIZONTAL TAIL DUE TO A LOAD ON HORIZONTAL TAIL (FLEX. BODY)	AFT BODY	DEG/LB	—
$\left(\frac{L_E}{L_R}\right)_{\delta_E\ RIGID\ BODY}$	RATIO OF ELASTIC TO RIGID LIFT DUE TO ELEVATOR DEFLECTION (FLEX. HORIZONTAL)	HORIZONTAL TAIL	—	V_e, M
$\frac{\partial i_H}{\partial L_E}$	CHANGE IN INCIDENCE OF HORIZONTAL TAIL DUE TO LOAD ON THE ELEVATOR	AFT BODY	DEG/LB	—
$\frac{\partial i_H}{\partial n_z}$	CHANGE IN INCIDENCE OF HORIZONTAL TAIL DUE TO LOAD FACTOR	AFT BODY	DEG/g	PAYLOAD FUEL LOADING
$\left(\frac{M_E}{M_{R_{.25\bar{c}}}}\right)_{\delta_E\ RIGID\ BODY}$	RATIO OF ELASTIC TO RIGID PITCHING MOMENT DUE TO ELEVATOR DEFLECTION (FLEX. HORIZONTAL)	HORIZONTAL TAIL	—	V_e, M

			REVISED	DATE	
ENGR.	MULALLY	2-29-72			LONGITUDINAL AEROELASTIC CORRECTIONS
CHECK					NTP
APR					TABLE I
APR					
	STROUS	2-29-72			

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LIFT

$$\left(C_{L_\alpha}\right)_{\substack{\text{FLEX}\\1g}} = C_{L_{\alpha_{WB}}\,\text{RIGID}} \cdot \left(\frac{L_E}{L_R}\right)_{WW} + \eta C_{L_{\alpha_H}\,\text{RIGID}} \cdot \left(\frac{L_E}{L_R}\right)_{H\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}} \cdot \frac{S_H}{S_W} \cdot \left(1-\frac{\partial\epsilon}{\partial\alpha}\right)$$

WHERE:

$$\left(\frac{L_E}{L_R}\right)_{H\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}} = \frac{\left(\frac{L_E}{L_R}\right)_{H\,\text{RIGID BODY}}}{1-\left(\frac{L_E}{L_R}\right)_{H\,\substack{\text{RIGID}\\\text{BODY}}} \cdot \eta C_{L_{\alpha_H}\,\text{RIGID}} \cdot \frac{\partial i_H}{\partial L_H} \cdot \bar{q} \cdot S_H}$$

$$\left(C_{L_{\delta_e}}\right)_{\text{FLEX}} = \left(\frac{L_E}{L_R}\right)_{\delta_e\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}} \cdot C_{L_{\delta_e}\,\text{RIGID}}$$

WHERE:

$$\left(\frac{L_E}{L_R}\right)_{\delta_e\,\text{FLEX. BODY}} = \frac{\left(\frac{L_E}{L_R}\right)_{\delta_e\,\text{RIGID BODY}}}{1-\left(\frac{L_E}{L_R}\right)_{H\,\substack{\text{RIGID}\\\text{BODY}}} \cdot \eta C_{L_{\alpha_H}\,\text{RIGID}} \cdot \frac{\partial i_H}{\partial L_E} \cdot \bar{q} \cdot S_H}$$

$$C_{L_{n_z}} = C_{L_{n_{z_W}}} + C_{L_{n_z}\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}}$$

WHERE:

$$C_{L_{n_z}\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}} = \eta C_{L_{\alpha_H}\,\text{RIGID}} \cdot \left(\frac{L_E}{L_R}\right)_{H\,\substack{\text{FLEX. BODY}\\+\text{HORIZONTAL}}} \cdot \frac{S_H}{S_W} \cdot \frac{\partial i_H}{\partial n_z}$$

ENGR.	MULALLY	2-25-72	REVISED	DATE	LONGITUDINAL ELASTIC EQUATIONS		NTP
CHECK							TABLE II
APR							
APR							
	STROUS	2-25-72					

PITCHING MOMENT

$$\left(C_{m_{\alpha_{.25\bar{c}_w}}}\right)_{FLEX.} = \left[\left(\frac{\partial C_m}{\partial C_L}\right)_{\substack{WB \\ RIGID}} + \Delta\left(\frac{\partial C_m}{\partial C_L}\right)_{WW}\right]\left(C_{L_\alpha}\right)_{\substack{WB \\ RIGID}}\left(\frac{L_E}{L_R}\right)_{WW}$$

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$$- \eta\, C_{L_{\alpha_H}}{}_{RIGID} \cdot \left(\frac{L_E}{L_R}\right)_{\substack{H \\ FLEX.BODY \\ +HORIZONTAL}} \cdot \bar{V}_H \cdot \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)$$

$$\left(C_{m_{\delta_e}}\right)_{FLEX.} = \left(\frac{M_E}{M_{R_{.25\bar{c}_w}}}\right)_{\substack{\delta_e \\ FLEX.\ BODY \\ +\ HORIZONTAL}} \cdot C_{m_{\delta_{e.25\bar{c}_w}}}{}_{RIGID}$$

$$\text{WHERE: } \left(\frac{M_E}{M_{R_{.25\bar{c}_w}}}\right)_{\substack{\delta_e \\ FLEX.BODY \\ +HORIZONTAL}} = \frac{\left(\frac{M_E}{M_{R_{.25\bar{c}_w}}}\right)_{\delta_e\, RIGID\ BODY}}{1 - \left(\frac{L_E}{L_R}\right)_{\substack{H \\ RIGID \\ BODY}} \cdot \eta\, C_{L_{\alpha_H}}{}_{RIGID} \cdot \frac{\partial i_H}{\partial L_E} \cdot \bar{q} \cdot S_H}$$

$$C_{m_{n_{z.25\bar{c}_w}}} = C_{m_{n_{z_W}}} - \left(C_{L_{n_z}}\right)_{\substack{FLEX\ BODY \\ +HORIZONTAL}} \cdot \frac{\ell_H}{\bar{c}_W}$$

ENGR.	MULALLY	2-29-72	REVISED	DATE	LONGITUDINAL ELASTIC EQUATIONS (CONTINUED)	ATP
CHECK						TABLE II CONT.
APR						
APR						9
	STROUS	2-29-72				

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AEROELASTIC TERM	DESCRIPTION	ELASTIC STRUCTURE	UNITS	VARIABLES
$\left(\frac{L_E}{L_R}\right)_{V_{RIGID\ BODY}}$	RATIO OF ELASTIC TO RIGID VERTICAL TAIL LIFT DUE TO SIDESLIP (RIGID BODY)	VERTICAL TAIL	—	V_e, M
$\frac{\partial i_V}{\partial L_V}$	CHANGE IN INCIDENCE OF VERTICAL TAIL DUE TO β LOAD ON VERTICAL TAIL	AFT-BODY	DEG/LB	—
$\left(\frac{L_E}{L_R}\right)_{\delta_{R\ RIGID\ BODY}}$	RATIO OF ELASTIC TO RIGID LIFT DUE TO RUDDER DEFLECTION (FLEX. VERTICAL)	VERTICAL TAIL	—	V_e, M
$\frac{\partial i_V}{\partial L_{\delta_R}}$	CHANGE IN INCIDENCE OF VERTICAL TAIL DUE TO LOAD ON VERTICAL TAIL DUE TO RUDDER DEFLECTION	AFT BODY	DEG/LB	—
$\left(\frac{M_E}{M_R}\right)_{\delta_{R\ RIGID\ BODY}}$	RATIO OF ELASTIC TO RIGID YAWING MOMENT DUE TO RUDDER DEFLECTION (FLEX. VERTICAL)	VERTICAL TAIL	—	V_e, M
$\left(\frac{R_E}{R_R}\right)_p$	RATIO OF ELASTIC TO RIGID ROLLING MOMENT DUE TO ROLL RATE	WING	—	V_e, M
$\left(\frac{R_E}{R_R}\right)_{\delta_s}$	RATIO OF ELASTIC TO RIGID ROLLING MOMENT DUE TO SPOILER DEFLECTION	WING	—	V_e, M
$\left(\frac{R_E}{R_R}\right)_{\delta_a}$	RATIO OF ELASTIC TO RIGID ROLLING MOMENT DUE TO AILERON DEFLECTION	WING	—	V_e, M
$\left(\Delta C_{\ell_\beta}\right)_{W_{"1g"}}$	EFFECT OF WING FLEXIBILITY ON C_{ℓ_β} ("1 g" FLIGHT)	WING	1/DEG	V_e, M

ENGR.	MULALLY	2-29-72	REVISED	DATE	LATERAL-DIRECTIONAL AEROELASTIC CORRECTIONS
CHECK					NTP
APR					TABLE III
APR					
	STROUD	2-29-72			

$$\left(C_{n_\beta}\right)_{FLEX.} = C_{n_{\beta}}{}_{WB_{RIGID}} + \left(\frac{L_E}{L_R}\right)_{V_{\substack{FLEX.BODY\\+VERTICAL}}} \cdot C_{n_{\beta_V}}{}_{RIGID}$$

$$\left(C_{Y_\beta}\right)_{FLEX} = C_{Y_{\beta}}{}_{WB_{RIGID}} + \left(\frac{L_E}{L_R}\right)_{V_{\substack{FLEX.BODY\\+VERTICAL}}} \cdot C_{Y_{\beta_V}}{}_{RIGID}$$

$$\left(C_{\ell_\beta}\right)_{FLEX} = C_{\ell_{\beta}}{}_{WB_{RIGID}} + \Delta C_{\ell_{\beta_W}}{}_{2J} + \left(\frac{L_E}{L_R}\right)_{V_{\substack{FLEX\ BODY\\+VERTICAL}}} \cdot C_{\ell_{\beta_V}}{}_{RIGID}$$

WHERE: $$\left(\frac{L_E}{L_R}\right)_{V_{\substack{FLEX.BODY\\+VERTICAL}}} = \frac{\frac{L_E}{L_R}{}_{V_{RIGID BODY}}}{1 - \left(\frac{L_E}{L_R}\right)_{V_{\substack{RIGID\\BODY}}} \cdot \eta C_{L_{\alpha_V}} \cdot \frac{\partial C_V}{\partial L_V} \cdot \bar{q} \cdot S_V}$$

$$\left(C_{n_{\delta_R}}\right)_{FLEX} = \left(\frac{M_E}{M_{R_{.25\bar{c}_W}}}\right)_{\delta_{R_{\substack{FLEX\ BODY\\+VERTICAL}}}} \cdot C_{n_{\delta_R}}{}_{RIGID}$$

$$\left(C_{Y_{\delta_R}}\right)_{FLEX} = \left(\frac{L_E}{L_R}\right)_{\delta_{R_{\substack{FLEX.\ BODY\\+VERTICAL}}}} \cdot C_{Y_{\delta_R}}{}_{RIGID}$$

$$\left(C_{\ell_{\delta_R}}\right)_{FLEX} = \left(\frac{L_E}{L_R}\right)_{\delta_{R_{\substack{FLEX.BODY\\+VERTICAL}}}} \cdot C_{\ell_{\delta_R}}{}_{RIGID}$$

ENGR.	MULALLY	2-29-72	REVISED	DATE	LATERAL-DIRECTIONAL ELASTIC EQUATIONS	NTP
CHECK						TABLE IV
APR						
APR						11
	STROUS	2-29-72				

WHERE:

$$\left(\frac{L_E}{L_R}\right)_{\substack{\delta_R \\ \text{FLEX. BODY} \\ +\text{VERTICAL}}} = \frac{\left(\frac{L_E}{L_R}\right)_{\delta_R\ \text{RIGID BODY}}}{1-\left(\frac{L_E}{L_R}\right)_{V_{\substack{\text{RIGID}\\ \text{BODY}}}} \cdot \eta C_{L_{\alpha_{V}}}{}_{\text{RIGID}} \cdot \frac{\partial \alpha_V}{\partial \delta_R} \cdot \bar{q} \cdot S_V}$$

$$\left(\frac{M_E}{M_{R,182W}}\right)_{\substack{\delta_R \\ \text{FLEX. BODY} \\ +\text{VERTICAL}}} = \frac{\left(\frac{M_E}{M_R}\right)_{\delta_R\ \text{RIGID BODY}}}{1-\left(\frac{L_E}{L_R}\right)_{V_{\substack{\text{RIGID}\\ \text{BODY}}}} \cdot \eta C_{L_{\alpha_{V\,\text{RIGID}}}} \cdot \frac{\partial \alpha_V}{\partial \delta_R} \cdot \bar{q} \cdot S_V}$$

$$\left(C_{l_{\delta_a}}\right)_{\text{FLEX}} = \left(\frac{R_E}{R_R}\right)_{\delta_a} \cdot C_{l_{\delta_a}\ \text{RIGID}}$$

$$\left(C_{l_{\delta_s}}\right)_{\text{FLEX}} = \left(\frac{R_E}{R_R}\right)_{\delta_s} \cdot C_{l_{\delta_s}\ \text{RIGID}}$$

$$\left(C_{l_{\hat{p}_W}}\right)_{\text{FLEX}} = \left(\frac{R_E}{R_R}\right)_{p} \cdot C_{l_{\hat{p}_W}\ \text{RIGID}}$$

$$\begin{Bmatrix} \left(C_{n_{\hat{p}_V}}\right)_{\text{FLEX}} \\ \left(C_{Y_{\hat{p}_V}}\right)_{\text{FLEX}} \\ \left(C_{l_{\hat{p}_V}}\right)_{\text{FLEX}} \\ \left(C_{n_{\hat{r}_V}}\right)_{\text{FLEX}} \\ \left(C_{Y_{\hat{r}_V}}\right)_{\text{FLEX}} \\ \left(C_{l_{\hat{r}_V}}\right)_{\text{FLEX}} \end{Bmatrix} = \left(\frac{L_E}{L_R}\right)_{V_{\substack{\text{FLEX BODY}\\ +\text{VERTICAL}}}} \cdot \begin{Bmatrix} \left(C_{n_{\hat{p}_V}}\right)_{\text{RIGID}} \\ \left(C_{Y_{\hat{p}_V}}\right)_{\text{RIGID}} \\ \left(C_{l_{\hat{p}_V}}\right)_{\text{RIGID}} \\ \left(C_{n_{\hat{r}_V}}\right)_{\text{RIGID}} \\ \left(C_{Y_{\hat{r}_V}}\right)_{\text{RIGID}} \\ \left(C_{l_{\hat{r}_V}}\right)_{\text{RIGID}} \end{Bmatrix}$$

ENGR.	MULALLY	2-29-72	REVISED	DATE	LATERAL-DIRECTIONAL ELASTIC EQUATIONS (CONTINUED)	NTP
CHECK						TABLE IV (CONT)
APR						
APR						12
	STROUS	2-29-72				

2. SIMULATION

A. Basic Considerations

The modern aircraft is a complex flying machine/system. The greater complexity makes the mathematical description difficult. Accurate and complete mathematical description is required to predict flight characteristics - especially dynamic behavior. Even for simple configurations the differential equations of motion are usually too complex for direct analytical solution. For most aircraft the aerodynamic and propulsion forces and moments are non-linear and functions of many variables. The control system tends to be very complex if system limits and failure states are to be investigated.

"Simulation" in its broad definition covers all techniques and facilities used to represent the characteristics of a system. The simplest aircraft simulation uses linearized equations of motion, linearized aerodynamic and propulsion data and rudimentary control system. The pilot may even be represented analytically. At the other end of the spectrum lies the comprehensive flight training simulator. The pilot sits in a cab of extremely detailed fidelity. A high-speed digital computer "solves" the equations of motion using non-linear data and detailed math models of the systems. The engineering flight simulation is situated somewhere in between. In flight dynamics we are interested in the total man-machine system; therefore, the piloted flight simulation is of major importance.

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2. A. Basic Considerations (Continued)

TYPES OF SIMULATION

PILOTLESS	GROUND-BASED PILOTED SIMULATION	IN-FLIGHT PILOTED SIMULATION
• MIMIC	• Simple "Chair & Stick"	• TIFS - Calspan/USAF
• Linear Analyses	• Fixed-Base Cab, T.V., Sophisticated Math Models	• Boeing 367-80
• Non-Real Time Digital		• Boeing/NASA 737
• Iron-Bird Rigs (May Use Pilot)	• Motion Base, T.V., Complete Cab, High-Fidelity Math Model	

PILOTED SIMULATOR USE IN DESIGN CYCLE

PHASE	RESEARCH	PRELIMINARY DESIGN	DETAILED DEVELOPMENT	FLIGHT TEST	OPERATIONAL USE
TASK	Critera Rule Making Human Factors	Design Reqmts. Data on Specific Problems Control Laws	Design Reqmts. Design Validation Equipment Development Pilot Training	Pilot Training Pilot Proficiency Dangerous Regimes Resolution of Problems	Crew Training Procedural Training Navigation All-Weather Operation
TYPE	Simple Fixed Base Sim. Moving Cockpit Sim. In-Flight	Fixed Base	Fixed Base Motion Base Iron Bird	Fixed or Motion Base Motion Base In-Flight	Motion Base In-Flight

Increasing Detail →

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SIMULATOR ADVANTAGES

- Design Tool for Complex, Non-Linear Problems
  - * Control System Development (with/without pilot)
  - * Safety/Failure Modes
  - * Engineering Insight into the problem
- Integrated Performance & Handling Qualities
  - * "Dynamic", Transient Characteristics (i.e. Eng-Out)
  - * Validates Total System Operation
  - * Assess Design Criteria/Requirements
- Pilot-In-The-Loop Contribution
  - * Pilot Impact Design
  - * Training/Familiarity/Confidence
  - * "Test" and "Line" Pilot Evaluation
- Cheaper (and Safer) Than Actual Flying

DATA OUTPUT

- Pilot Opinion & Comments
- Time Histories of Parameters
- Auxiliary Computations (Trims, Touchdown Dispersion, etc .)

THE "FIDELITY" SPECTRUM

- Linear EQ of Motion
- Simple Aero/Prop
- "Point" Flight Condition
- Fixed Base
- Simple Cockpit/Controllers
- No Visual Display

⇨

- Rigorous EQ. of Motion
- "Complete" Aero/Prop
- Full Flight Envelope
- Motion Base
- "Complete" Cockpit
- Visual/Aural Cues

Increased fidelity may be the only way to obtain credible predicted flight characteristics. The risk of problems or "surprises" in flight test goes down. Flight test time may be reduced. The pilot will develop confidence in the design. Detailed design questions can be answered.

The price of increased fidelity: more and better wind tunnel data, propulsion information, detailed software specification, more manpower, increased check-out, longer flow time, more money. For training simulators or post-design engineering development simulators, additional flight test data (at high cost) may be required to achieve proper fidelity.

The Key Point:

> Flight simulation is generally necessary in the design process; however, be sure to balance simulation detail/fidelity against the benefits to be derived.

B. Simulation Computer Fundamentals

The heart of the simulation is the computer which "solves" the equations of motion. Consider the following simple example:

$$\boxed{F = ma \text{ for Mass-Spring-Damper}}$$

$$F(t) - Kx - C\dot{x} = m\ddot{x}$$

where F (t), K, and C are in themselves variables of other quantities, such as V_e, Mach No., flap setting, etc.

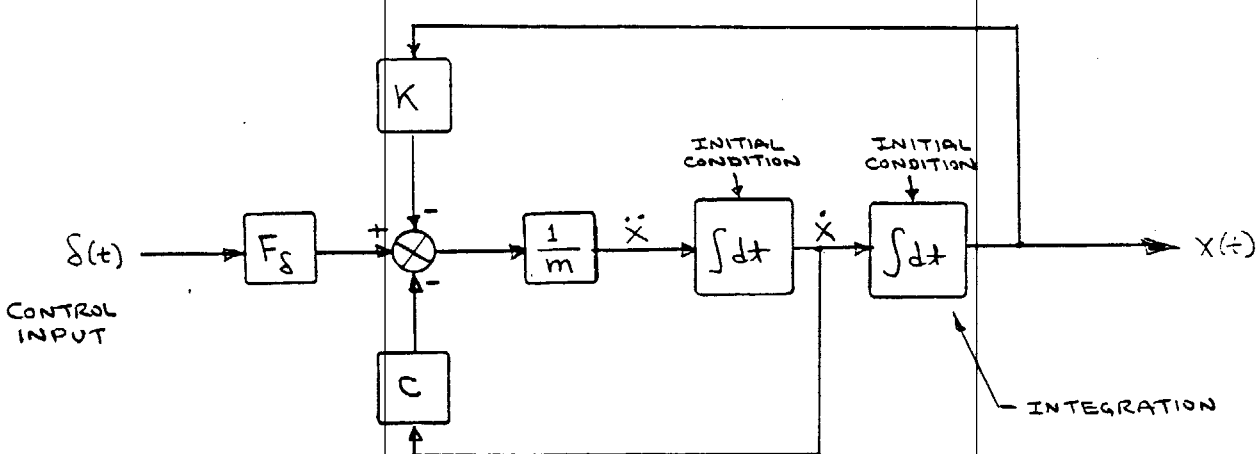
The equation of motion may be written as

$$\ddot{x} = \frac{1}{m}\left[F_\delta \cdot \delta(t) - Kx - C\dot{x}\right]$$

$$\dot{x} = \int \ddot{x}\,dt$$

$$x = \int \dot{x}\,dt$$

BLOCK DIAGRAM FORM :



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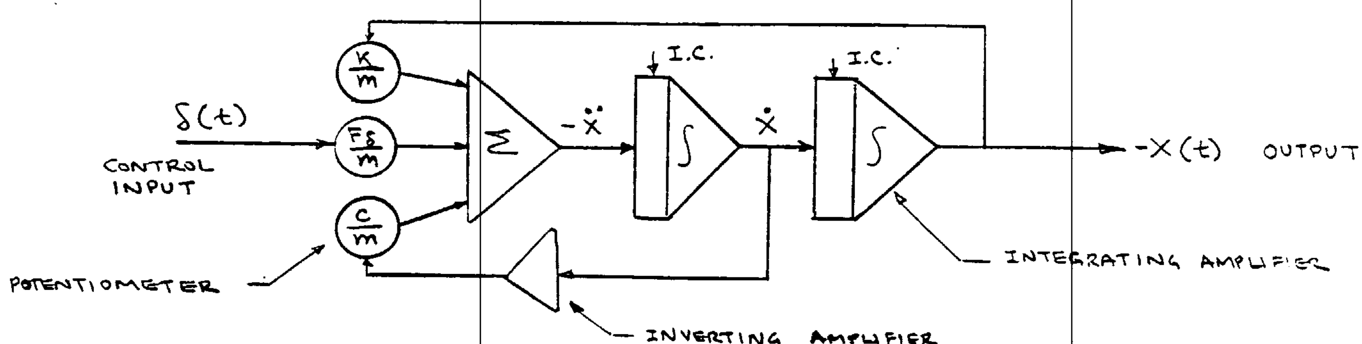
B. Simulation Computer Fundamentals (Continued)

- If the computation can be performed in "real time", then a human pilot can interact with the computation to "fly" the simulation.

Three types of computers are used in flight simulation: Analog, Digital, Hybrid. The analog computer uses operational amplifiers to sum and integrate electrical signals (voltage) representing analogous physical quantities. The digital computer breaks the problem down into "software" coding and algebraic manipulation. The solution to the problem is done in small time frames analogous to motion pictures. The hybrid computer is a combination of the digital and analog computer.

ANALOG SIMULATION

The mass-spring-damper problem:



Analog Advantages

- Often simple/cheap to use
- Excellent frequency response
- Able to run "faster" than "real time"
- Control system filters are easily simulated.

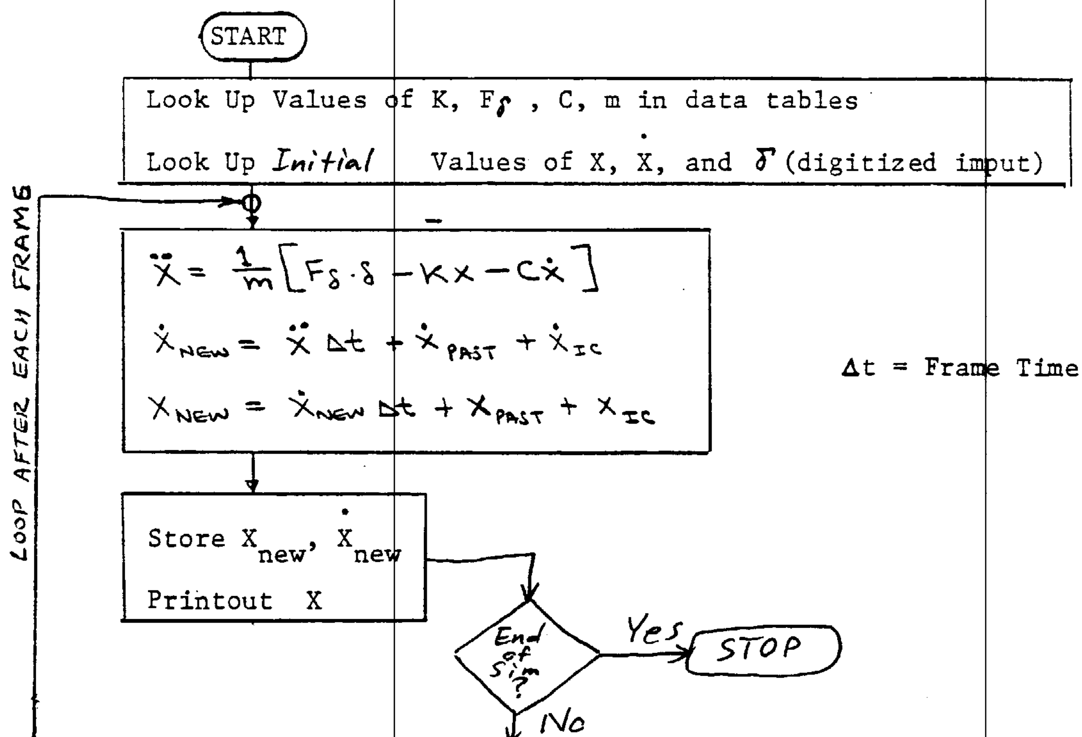
Disadvantages

- Amplifier overload forces "scaling"
- Noise, drift, accuracy
- Not good for complex data functions
- Trig. functions, etc. must be handled by special equipment
- Extensive setup & checkout time required daily if problem is large.

B. Simulation Computer Fundamentals (Continued)

DIGITAL SIMULATION

The mass-spring-damper problem is software.



To maintain "real-time" fidelity at airplane/pilot frequencies of interest (up to ω = 10 RAD/SEC disregarding structural modes, etc.) the solution frame time must be less than approx. 40 milliseconds (i.e. 25 frames/sec.). Modern digital computers perform their arithmetic very quickly, thus very detailed and complex problems can be handled. Most flight simulation today is digital.

Digital Advantages

- Able to store complex data tables
- "No" drift, signal-to-noise, etc.
- Very accurate (little scaling req'd.)
- Handles complex trig. functions, axes transformations, etc.
- Repeatable, "easily" checked
- FMS is digital, FCS may be digital

~~Disadvantages~~ Circa 1975!

- ~~Expensive, requires specialized personnel~~
- ~~Limited to lower frequencies (not good for system analyses)~~

HYBRID SIMULATION

The hybrid simulation uses the digital computer for data storage (table lookup, etc.) signal conditioning, etc. Integration is then accomplished using the analog operational amplifiers, which increases frequency response. (Some frequency response limitations may be imposed by the digital-analog data exchange rate--depending on the configuration of the math model).

System filters, limits, etc., can be readily "patched" on the analog. In addition, it is possible to monitor any signal in the control system/augmentation system - on demand. (This capability must frequently be set up in advance with a digital program.)

⇒ Hybrid has numerous cost and design problems

C. Typical Piloted Simulation (Digital)

Figure 1 sketches the overall simulation with the "pilot in the loop". The pilot must respond to realistic cues, i.e., the higher the degree of realism the less the pilot will have to extrapolate to real life. Figure 2 describes a typical developed engineering simulation. Note that the software is structured into modules to facilitate coding, checkout, etc. Software management in a sophisticated simulation is a major concern.

Figures 3 and 4 present typical solution flow charts for solving the equations of motion. It can be seen that the digital computer provides the only practical way to get through the maze.

Typical examples of problems addressed on the simulator:

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C. Typical Piloted Simulation (Digital) (Continued)

- 737 PROGRAM

  1. Spoiler Pitch-Up (F40 Turn Entry)
  2. Roll Response at Landing Approach
  3. V_{MC_g} - Engine-out Control Speeds
  4. Yaw Damper/Autopilot Development
  5. Motion Requirements for Training Simulator
  6. Thrust Reverser Failure
  7. "Deep Sideslip", Fin Stall at Full Rudder
  8. High-Speed Longitudinal Stability
  9. Training Simulator Check-Out
  10. Mach Trim/"Tuck" Effects

- MOD C-8A, AUGMENTOR WING BUFFALO

  1. Lateral Control Characteristics
  2. SAS Design
  3. Engine-Out Control Techniques
  4. Basic A/P Characteristics
  5. Engine & Nozzle Characteristics (Rates, etc.)
  6. Placard Speeds/Structural Flight Conditions
  7. Surface Rates/Hydraulic System Requirements
  8. Operating Procedures

D. Simulation Papers/Articles

- General development

* AGARD Conference Proceedings No. 79-70 on Simulation, NATO, January 1971

* Westbrook, C.B.; "Background of Piloted Simulator Development", Conf. on Simulation in Space Technology, Virginia Polytechnic Institute, Aug. 1964, Proceedings #A65-29088.

D. <u>Simulation Papers/Articles</u> (Continued)

- Examples of Engineering Use

  * AIAA Paper 72-764, "Use of Fixed and Moving Base Flight Simulators for the Aerodynamic Design and Development of the S-3A Airplane", C. F. Anderson and B. T. Averett, Lockheed California Co., Aug. 1972

  * AIAA Paper 72-762, "Use of the Flight Simulator in the Design of a STOL Research Aircraft", R. E. Spitzer and P. C. Rumsey - The Boeing Co., and H. C. Quigley - NASA Ames, Aug. 1972.

- The Use of Training Simulators

  * "The Pilot's Future Role is Scrutinized", Aviation Week, Oct. 22, 1973, PP 108, 109.

- NASA-Ames Flight Simulator for Advanced Aircraft

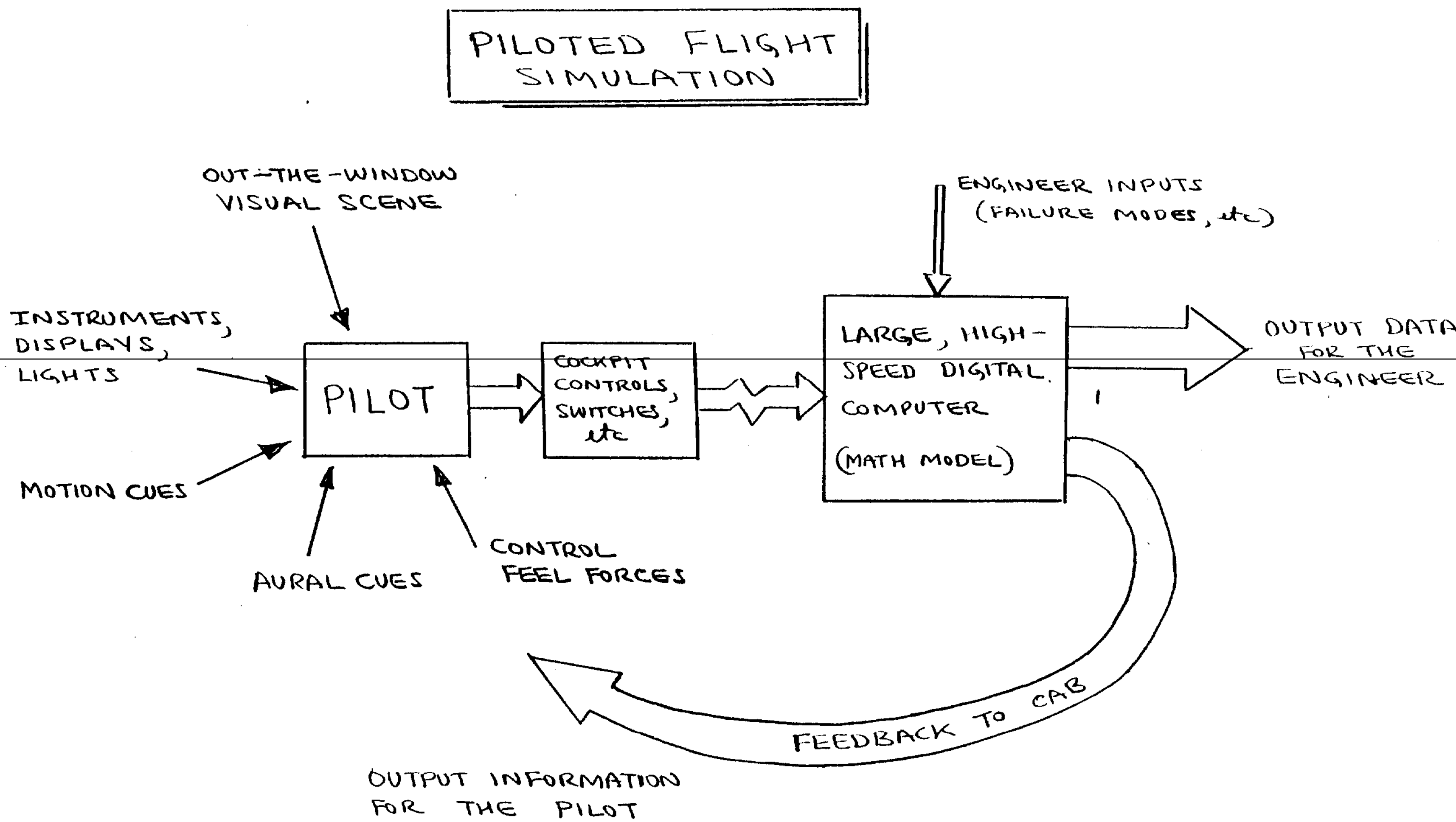
  * "The Simulator to Match the Transports to Come," M. D. White and J. C. Dusterberry of NASA-Ames, Aeronautics/Astronautics, Sept. 1969.

- Simulation Math Models (VTOL, STOL, Conventional)

  * Boeing Doc. D6-42106, "Simulator Specification for the R984-33 Lift-Fan V/STOL Research Aircraft", A. A. Lambregts, W. M. Eldridge, R. E. Spitzer, Oct. 1972

  * Boeing Doc. D6-26065TN, "Simulator Model Specification for the Modified C-8A", P. C. Rumsey and R. E. Spitzer, Dec. 1971.

  * Boeing Doc. D6-41217, "Description of a General Engineering Flight Simulator Model", R. A. Curnutt, June 1974.



PILOT RECEIVES CUES & STIMULI SUFFICIENT TO MAKE THE SIMULATION "REALISTIC", ie, HE SENSES THAT HE IS ACTUALLY "FLYING" THE AIRCRAFT.

FIG. 1

SIMULATION DESCRIPTION

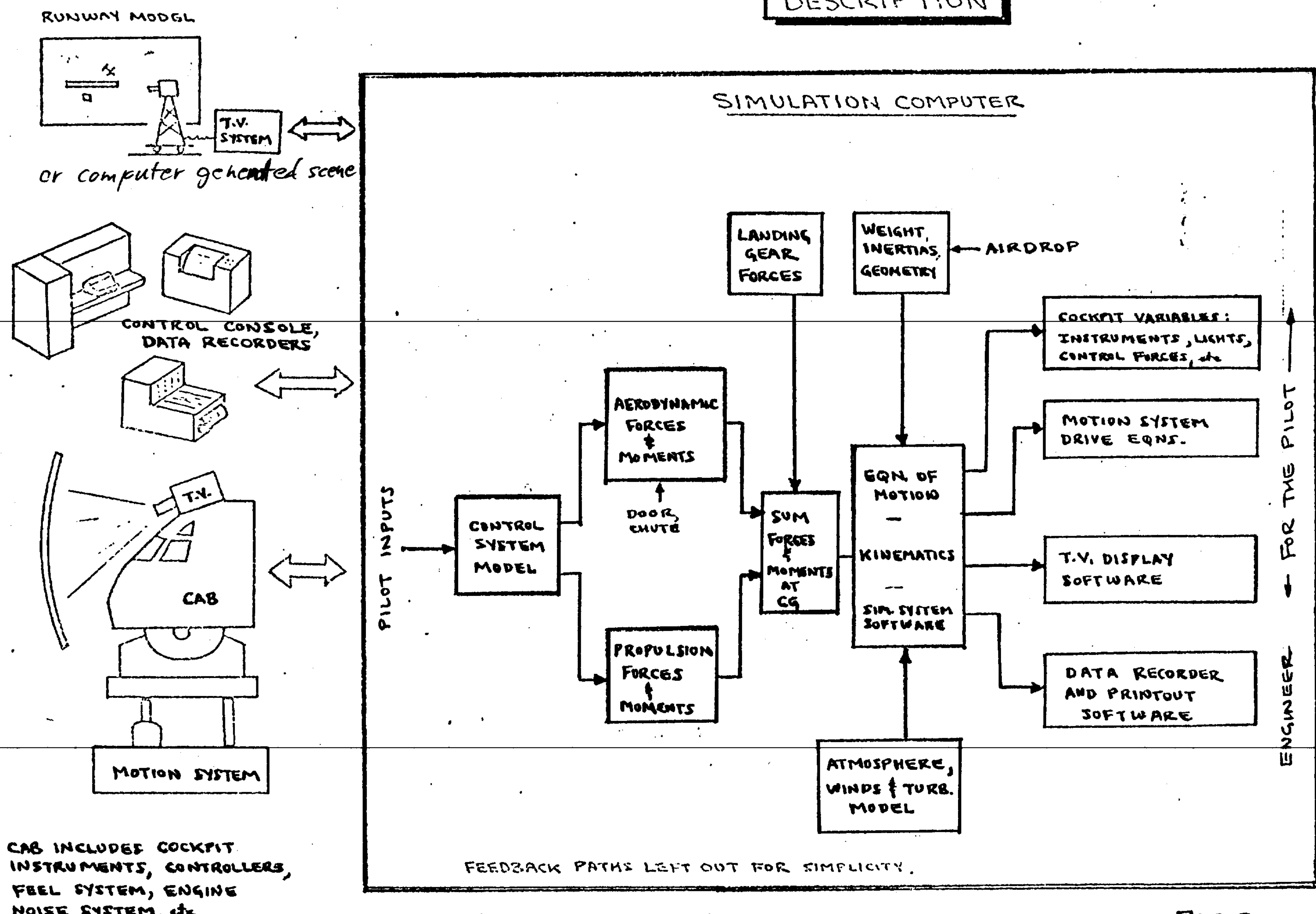
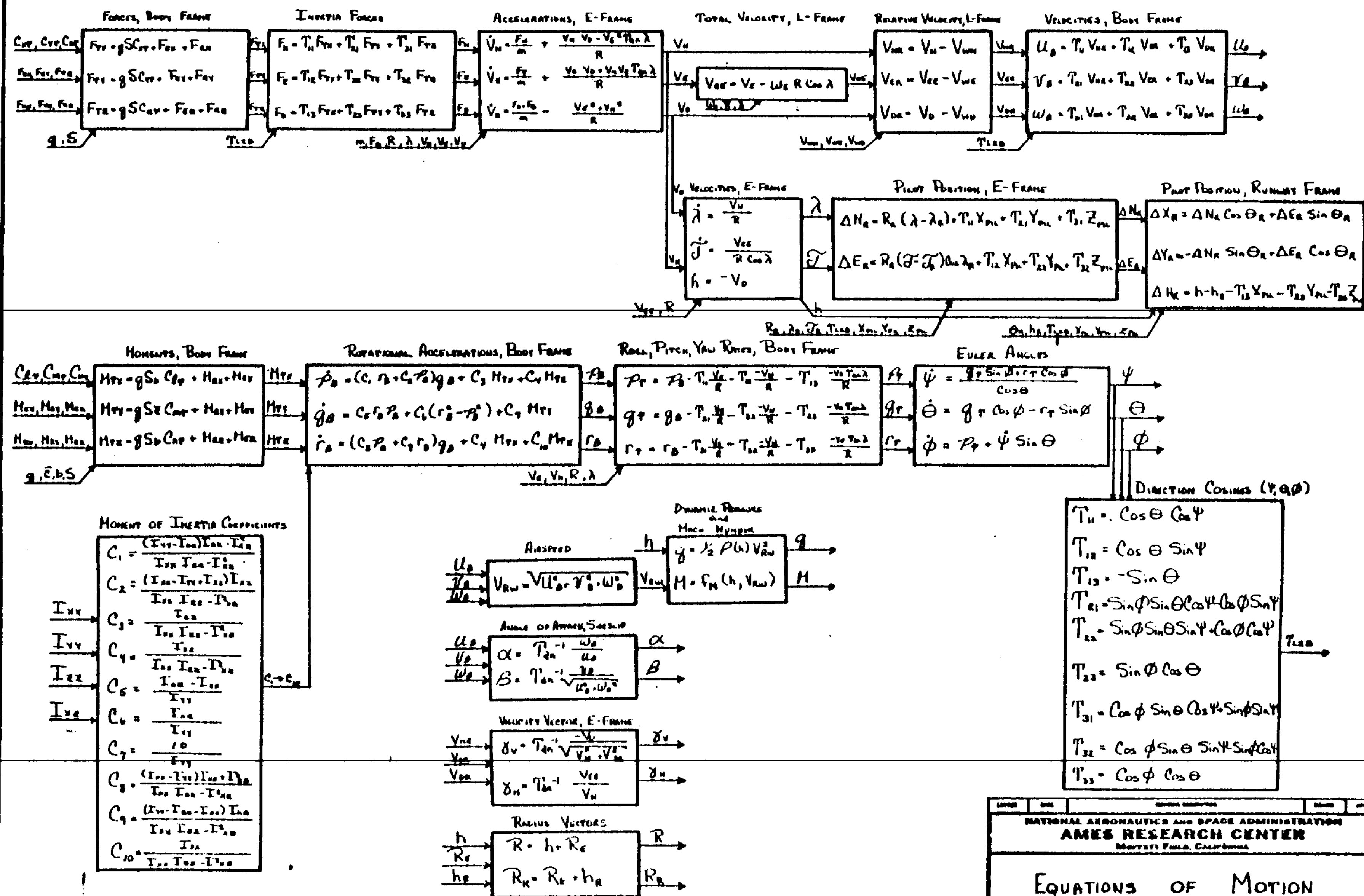


FIG. 2



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EQUATIONS OF MOTION

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A.D. JONES

FIG. 3

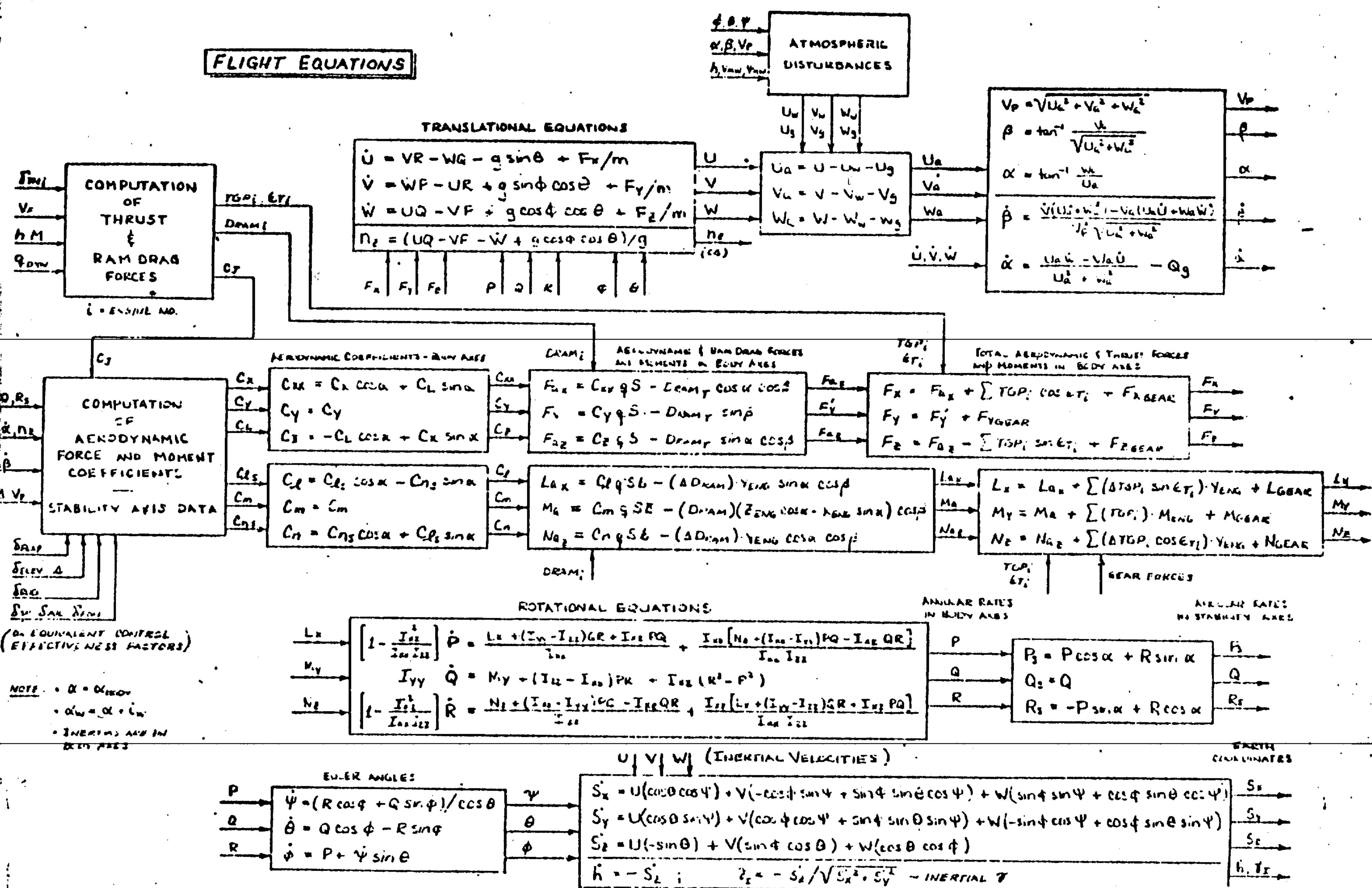
FLIGHT EQUATIONS
COMPUTATION OF THRUST & RAM DRAG FORCES
i = ENGINE NO.
TRANSLATIONAL EQUATIONS
ATMOSPHERIC DISTURBANCES
COMPUTATION OF AERODYNAMIC FORCE AND MOMENT COEFFICIENTS — STABILITY AXIS DATA
AERODYNAMIC COEFFICIENTS - BODY AXES
AERODYNAMIC & RAM DRAG FORCES AND MOMENTS IN BODY AXES
TOTAL AERODYNAMIC & THRUST FORCES AND MOMENTS IN BODY AXES
GEAR FORCES
ROTATIONAL EQUATIONS
ANGULAR RATES IN BODY AXES
ANGULAR RATES IN STABILITY AXES
(OR EQUIVALENT CONTROL EFFECTIVENESS FACTORS)
EULER ANGLES
(INERTIAL VELOCITIES)
EARTH COORDINATES
INERTIAL

FIG. 4