# Distributed Control Formulation for Aircraft Flight Control

## A Technical Report

## On the formulation of
## A Control Theory for Aircraft Flight Control that employs
## Distributed Models of the Aerodynamic Flow

by

**Raymond C. Montgomery and Michael A. Scott**

**Dynamics and Control Branch**
**NASA Langley Research Center**
**Hampton, VA 23681-0001**

## Abstract

This report proposes a novel formulation of a control theory for aircraft flight control. Novel, because it incorporates a partial differential equation mathematical description of the flow field about an aircraft into its flight control system. This is in contrast to existing formulations which incorporate state variable models of aerodynamics according to the method of Bryan, i.e. stability derivatives. First, motivation of the work is presented. This is followed by a brief review of available mathematical formulations of aerodynamics with a focus on their readiness for incorporation into real-time control laws. Next, a distributed controller architecture is described which incorporates a distributed, real-time model of the aerodynamics into a state estimator that drives a regulator. Finally, suggestions are made for near-term research centered around this architecture, designed to produce a useful tool for the aerospace community. The report is limited to formulation only and is meant to provide an architecture for further study and future development by the Dynamics and Control Branch and aerospace community in general.

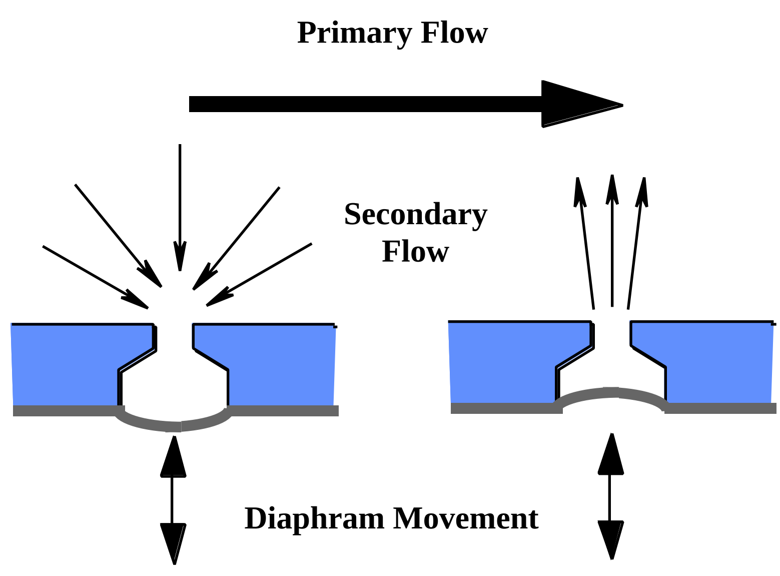
## Introduction

The past half century has witnessed the development of many advances in both analysis and control of aerospace vehicles. The state of the knowledge in the dynamics of aircraft and in classical flight control is well documented in textbooks of which references 1 and 2 are excellent examples. Modern control theory which had its start in the early 1960s and, in application to aerospace vehicles, is laid out in AGARDograph No. 251, reference 3, has now even been applied directly in design, exemplified by the Pegasus launch vehicle, which employs the modern LQG/LQR theory (Linear Quadratic Gaussian/Linear Quadratic Regulator). For Pegasus the Kalman filter design is used in navigation and for atmospheric control Kalman regulator gains the are used, references 4 and 5. Reference 5 goes into a detailed discussion on requirements that led to using LQG/LQR theory for Pegasus.

The state of the art has a general common thread stemming from the method of Bryan, reference 6, the use of a model of aerodynamics which is an algebraic map of the vehicle states into forces and moments on the aircraft. This concept has led to a well developed science both in analysis and experiment to uncover this *map,* an attempt to represent a complex flow phenomena in terms simpler than those required to mathematically describe flow physics more completely.

Areas where this may break down are those involving unsteady aerodynamics, e.g. rapidly moving parts, wind gusts, or control devices that rely on or , by their very nature, produce distributed, unsteady flow effects. Control devices used in the past have not been seriously encumbered by the lack of a control theory that is conceptually capable of handling unsteady or distributed aerodynamic effects since their frequency of operation was usually much less than that necessary to produce substantial deviations from steady flow and, the devices themselves tended to be macroscopic in nature allowing a mapping of their effect into a common control point, e.g. a single control surface deflection.

At this time, however, a new group of devices is being developed which, owing to their very nature, defies the steady, single-point, control assumption. These devices are flow control actuators, active jets, synthetic jets, active porosity devices, and perhaps, even others, and are expected to play a strong hand in the development of a seamless aircraft, one with no moving external control surfaces, merely hundreds or even thousands of small ports through which control of the vehicle as well as morphed performance trimming is accomplished aerodynamically. The developments may be facilitated by a new technology called MEMS (Micro-Electro-Mechanical Systems), an overview of which can be accessed via the world wide web, reference 7. The basic concept of one of these devices, the synthetic jet, is illustrated below. This device is a bellows which pumps fluid back and forth into a primary flow using a diaphram. Because friction coupling the device acts like a doublet, having zero net mass flow, but creating a vortex. These have been shown to have profound effects on the primary flow. Their use in gangs, varying amplitude as well as phasing spatially, are not thoroughly understood and are the subject of intense research.




**Synthetic Jet Concept**

Fortunately, the state of the art in theoretical aerodynamics, considering the ability to desensitize the effects of model error through the use of on-board measurements (Kalman estimation and filter theory, references 8 and 9), when coupled with outstanding progress in computer technology, *does* marginally support the inclusion of partial differential equation models in flight control systems. The purpose of this paper is, then, to establish a framework for doing just that, that is, to propose a flight control system architecture that incorporates real-time, distributed models of aerodynamics.

First, we will overview the available aerodynamic theories, highlighting the features that enable or disqualify each as a real-time model of vehicle aerodynamics for flight control applications, finally selecting one for future development and expounding upon it briefly to make transparent the discussion on the proposed flight control system architecture. Then, the proposed architecture incorporating a distributed, partial differential, equation model is presented. This concept follows closely that presented in reference 9. Finally, suggestions are made for future research designed to transition this proposal from a mere concept into a working tool for the aerospace community.

## Brief Summary of Available Aerodynamic Theories

The basic theories involved have not changed since the publication of reference 10 which is a comprehensive treatment of the state of the art as it existed at that time, prior to the evolution of modern computer technology when people had to think and numerical results were few and far between. Excluding molecular flow theory, the fundamental physical equations used to obtain complete descriptions of fluid flow for aircraft are:

the momentum equation
the continuity equation
the energy equation
and, the equation of state of the gas.

Herein, we will not consider the energy equation or the equation of state of the gas but rather will focus on the first two equations, momentum and continuity.

Aerodynamic theories that have been developed with these for aircraft applications include:

the so-called Navier-Stokes theory applicable to viscous, compressible flow,
Euler theory for invicid, compressible flow,
and, potential theory for flow assuming invicid, incompressible flow.

These have been listed in order of decreasing difficulty in application. To represent these relations the following symbols will be used:

- $t$ is time
- *grad* is the gradient operator in 3 dimensional space (x,y,z)
- $\mathbf{w}$ is the instantaneous velocity vector of a particle at (x,y,z)
- $\mathbf{g}$ is the body force vector, force/unit mass (usually gravity) at (x,y,z)
- $p$ is pressure at (x,y,z)
- $\rho$ is fluid density at (x,y,z)
- $\gamma$ is the ratio of specific heats for the gas

and dyadic notation is used, as in reference 10.

The continuity equation takes the general form:

$$\frac{\partial \rho}{\partial t} + \mathbf{w} \circ grad(\rho) + \rho \circ div(\mathbf{w}) = 0$$

and is applicable to all above formulations with the caveat that density may be a constant for the last case considered, invisid, incompressible flow.

For the Navier-Stokes theory, in addition to the continuity equation, the energy equation, and the equation of state one must:

include both normal and tangential stresses when applying Newton's law to an element;
assume that moments about an arbitrary axis through the element are zero, i.e. the stress tensor is symmetric;
assume shearing stresses are proportional to the rate of change in the deformation of the element;
geometrically relate deformation rates of the element to gradients in the velocity field;
and, rewrite surface stresses in terms of the pressure (defined to be the average normal stress over a spherical element) and gradients in the velocity field;

If this is done the so-called Navier-Stokes equations result, reference 10:

$$\frac{\partial \mathbf{w}}{\partial t} + \mathbf{w} \circ grad(\mathbf{w}) = \mathbf{g} - \frac{1}{\rho} grad(p) + \frac{1}{3}\nu \; grad(div\mathbf{w}) + \nu \; \Delta\mathbf{w}$$

where $\nu$ is the coefficient of viscosity and $\mathbf{g}$ is the body force vector (force/unit mass), usually gravity. The general features of these flows, if they cannot be described by a simpler theory, are that:

no velocity potential exists anywhere in the flow field;
one must grid the entire flow field as well as boundaries to obtain numerical solutions which can be difficult for mixed flow problems (i.e. transonic flow);
and, the authors are not aware of any existing production codes suitable for control applications (they take too much computer time and can produce unstable solutions, just as a result of the numerical computations, i.e. the instabilities are not real).

The next simpler theory is Euler theory wherein the coefficient of viscosity is set to zero in the momentum equation, resulting in the Euler equation (Newton's law) for inviscous flow:

$$\frac{\partial \mathbf{w}}{\partial t} + \mathbf{w} \circ grad(\mathbf{w}) = \mathbf{g} - \frac{1}{\rho} grad(p)$$

and, under the assumption of adiabatic flow, the energy equation for a perfect gas becomes:

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

For Euler theory, one may have velocity potential in shock free regions if circulation is zero. This allows a two degree of freedom reduction of the solution variables list and is generally worthwhile. However, one must still grid flow field as well as boundaries and variable grid structures are necessary for mixed flows. Still, the authors are not aware of any existing production codes suitable for control applications and solutions have the same basic grid difficulties present in the Navier-Stokes theory.

Finally, we arrive at potential theory. Under the additional assumption of incompressibility, the continuity equation above reduces to $div(\mathbf{w}) = 0$. With the additional assumption that the fluid is initially irrotational and inviscous, a velocity potential, $\Phi$, does exists, meaning that $\mathbf{w} = \mathbf{grad}(\Phi)$ so, the continuity equation becomes $\Delta\Phi = 0$ where

$\Delta = \;^{2} = \left(\frac{\partial^2}{\partial x^2} + \frac{\partial^2}{\partial y^2} + \frac{\partial^2}{\partial z^2}\right)$ is the Laplacian operator. The solution to the flow problem is enormously simplified since now, by virtue of Green's theorem, the solution in the field can be expressed in terms of that on the boundaries and so, we need only grid the near field boundaries -- aircraft surfaces and wakes assumming that the far field is handled analytically. Most of the problems in modeling flow using this theory centers around determining boundary conditions that meet the surface geometry requirements as well as the singularities required in the flow to generate aerodynamic forces, i.e. modeling the vorticity ans singular surfaces in the flow, the wakes.

For potential theory, the momentum equations can be integrated analytically to yield:

$$\frac{\partial \Phi}{\partial t} + \frac{\mathbf{w}^2}{2} + \frac{p}{\rho} - U = f(t)$$, the Bernoulli equation,

where U is the body force potential, usually $(-g\ \angle z)$. For most applications the function f(t) is zero and the later equation is used to merely to define pressure distributions given the time-varying solution to the potential equation $\Delta\Phi = 0$ which satisfies the boundary conditions.

Over the past several years tremendous progress has been made in solutions to fluid flow problems using potential theory and this has led to, so-called, panel methods which incorporate far field boundary conditions analytically and the near-field boundary conditions numerically. The Green's function solution to the potential equation is:

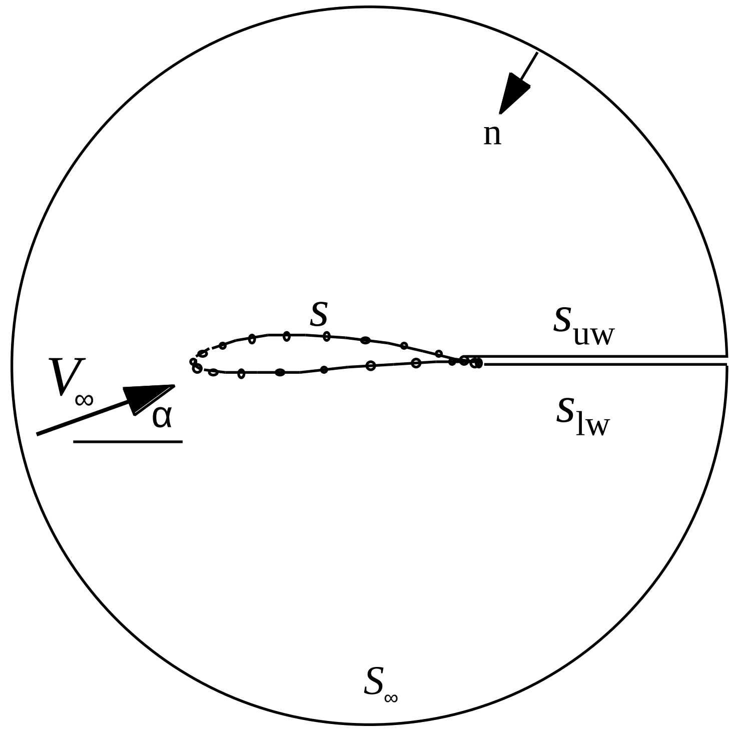
$$\Phi(x,y,z) = \int_{\partial} \Phi(\xi,\eta,\varsigma)\mathbf{n} \quad \left(\frac{1}{r}\right) d\xi\, d\eta\, d\varsigma$$

$$+ \int_{\partial} \left(\frac{1}{r}\right)\mathbf{n} \quad \Phi(\xi,\eta,\varsigma)\, d\xi\, d\eta\, d\varsigma$$

$$^{2}\Phi(x,y,z) = 0 \qquad (x,y,z)$$

$$V(x,y,z) = \mathrm{grad}(\Phi(x,y,z))$$

which represents the solution in a region , below, bounded by, $\partial \;= (S_{\text{uw}}, S_{\text{lw}}, S_{\infty}$ and the airfoil surface $S$). The far field boundary condition is $V = (V_{\infty}, \alpha)$.




$$\partial \quad = S \quad S_\infty \quad S_{lw} \quad S_{uw}$$

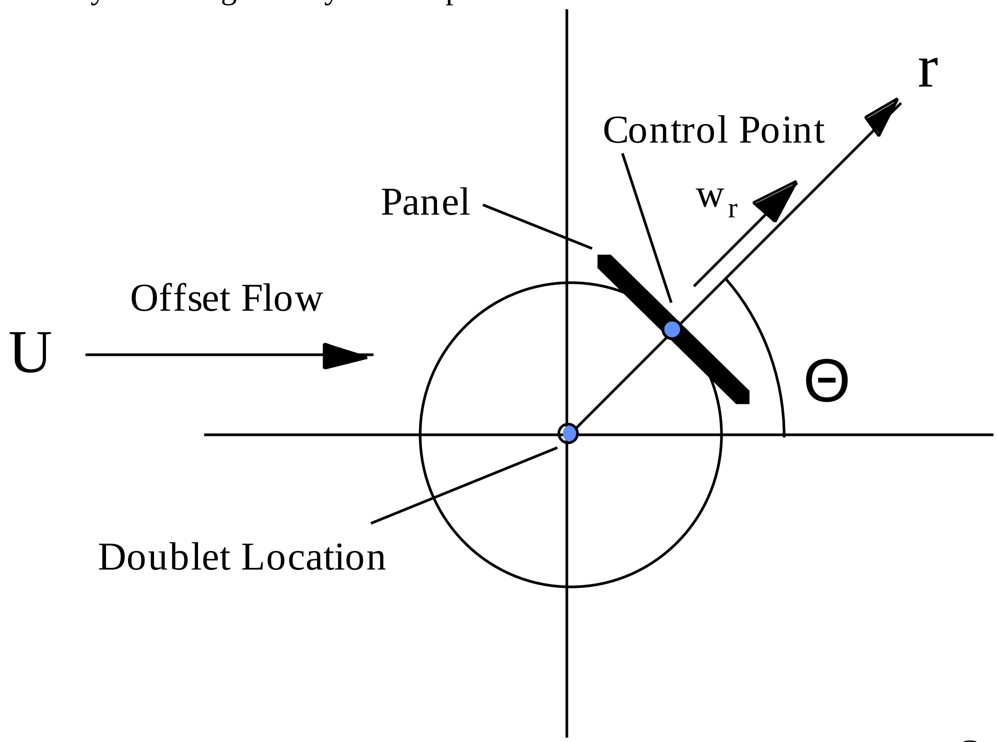
The first integral can be thought of as a distribution of doublets of strengths $\Phi(\xi,\eta,\varsigma)d\xi d\eta d\varsigma$, integrated over $\partial$ and, the second as a distribution of sources $\mathbf{n} \quad \Phi(\xi,\eta,\varsigma)d\xi d\eta d\varsigma$, integrated over $\partial$ . So, the solution to the problem is the flow field generated by the distribution of sources and doublets and the problem reduces to that of generating appropriate boundary conditions for $\partial$ and then, generating the distribution of sources and doublets over $\partial$ that meet those conditions. Panel methods do this numerically by analytically representing the boundary conditions over the far field of $\partial$ and discretizing near field, i.e. breaking up the surface of an aircraft into surface elements, or panels. Reference 11 provides an excellent tutorial on these methods and a historical perspective on their development for low speed aerodynamics. A very brief summary of the panel method is presented below, for completeness, so that the control formulation proposed herein is understandable.

Although not directed to real-time flight control, the panel code PMARC, reference 12, is a production code that incorporates the nuances required for real-time flow prediction and unsteady, time-varying flow that results either from the deformation of the boundaries as well as flow through them. So, in control jargon, provided Kalman filter theory, reference 8, can be cast so that on-board sensors can close the loop on general flow models of this type, thereby desensitizing on-board estimation to inherent modeling errors (largely resulting from the invisid, incompressible assumption), this model may prove suitable for predicting a broad range of dynamics conditions in the rapid maneuvering of aircraft handling incorporating appropriate models of novel, flow-control actuators.

**Synopsis of panel codes - an example**

A brief outline of the panel code solution method is presented in this section. The general steps, 1 through 4, in panel methods are indicated below and are illustrated using a two-dimensional doublet in a uniform offset flow:

1. Derive a formula for the influence of source/doublet strengths for a type of panels to be used on the velocities in general space and, in particular, for the space including the vehicle. Sometimes the source/doublets are distributed with uniform strengths over a panel which might geometrically be a trapezoidal plane surface. Generally, the source/doublet does not have to be collocated with the panel. There may be computational advantages to that however but, for the aircraft morphing problem the source/doublet location is not generally located at the control point, i.e. the point where a boundary condition is to be enforced. For an example we take the panel to be a plane surface and the a source/doublet to be a doublet at the origin of strength C. The sketch below lays out the geometry for this panel.

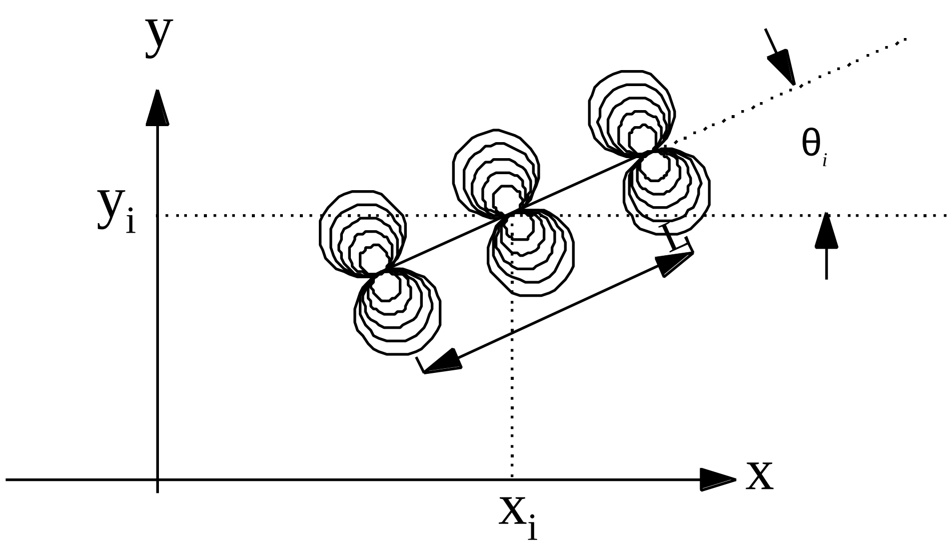



Then, the appropriate formula is for the added potential of the panel is $\varphi = -\frac{C}{2\pi}\frac{\cos(\theta)}{r}$

and the velocity field, in general, is $w_r = \frac{\partial\varphi}{\partial r} = \frac{C}{2\pi}\frac{\cos(\theta)}{r^2}$ , $w_\theta = \frac{1}{r}\frac{\partial\varphi}{\partial\theta} = \frac{C}{2\pi}\frac{\sin(\theta)}{r^2}$.

This is often replaced by a more suitable source or doublet panel type, e.g. one with a uniform or other source or doublet distribution as in the example below. These replacements are now facilitated by the availability of symbolic manipulators which take much of the drudgery out of the process and, for a 2 dimensional panel of uniform doublet distribution of length L, located at coordinate $x_i$, $y_i$ and inclined at an angle $\theta_i$ to the x axis we have the equation below which was obtained by integrating a uniformly distributed doublet over the panel
length.

$$\Phi(x,y,x_i,y_i,L_i,\theta_i,C_i) = C_i \left\{ \tan^{-1}\left[ \frac{(x-x_i)\cos\theta_i + (y-y_i)\sin\theta_i + \frac{L_i}{2}}{(y-y_i)\cos\theta_i + (x-x_i)\sin\theta_i} \right] - \tan^{-1}\left[ \frac{(x-x_i)\cos\theta_i + (y-y_i)\sin\theta_i - \frac{L_i}{2}}{(y-y_i)\cos\theta_i + (x-x_i)\sin\theta_i} \right] \right\}$$




2. Somehow represent the vehicle in panels of the type selected in step 1. There may be errors in this. Even good panel methods have discontinuities between adjacents. For the example of the point doublet we will use only one panel, normal to the ray θ=Θ. The point where boundary conditions are evaluated and are to be satisfied by the solution is called the control point. We take it to be at r=R and θ=Θ. In this case the geometry is represented by s=(R,Θ).

3. Generate boundary condition equations in terms of geometry and offset flow. This equation will be linear in source/doublet strengths and elementary for usual offset flow conditions. For the point doublet example, we take the velocity outward normal to the panel. Thus, to make the velocity normal to the panel zero, no flux, then the velocity generated by the potential function should be set equal and opposite to that of a uniform offset flow along the line Θ=0, i.e. $w_r(s) = -U\cos(\theta) = \frac{C}{2\pi}\frac{\cos(\theta)}{r^2}$ . So, generally we expect that the boundary condition will lead to $w_r(s) = -U\cos(\theta) = \frac{C}{2\pi}\frac{\cos(\theta)}{r^2}$ or to equations of the type $f(U,s) = g(s)\cdot C$. In this case, the function f is nonlinear in s and is usually elementary in U and the function g is a square matrix which is nonlinear in s. C is a matrix of size equal to the number of panels. A complication exists in that wake surfaces must also be included in some way and boundary conditions for them must also be generated. Reference 11 has a detailed discussion of this matter but,

suffice it to say that the key to success of the entire process resides in this step. For a two-dimensional problem, as shown in the example below, these conditions are a Kutta condition, $\mu_1 - \mu_N = \mu_W$, stating that the difference in doublet strength of the first and last panel generating a closed surface is the strength of the doublet generating the wake. These conditions are not the derivative of physical laws but, follow from insight on the character of fluid flow. This is an alarming state of affairs for individuals used models based on, say, Newton's laws. The option is to model the shedding of vorticity in a more precise manner, namely using viscous flow equations.

4. Solve boundary conditions for source/doublet strengths. The solution may be accomplished by an inversion of g. Important note: If this is done, the resulting $g^{-1}$ can be used to generate solutions for other uniform offset flow conditions by a cheap vector multiplication. This means that it need be done only once in a system with no moving parts and, it can be done at initialization. The evolution of the dynamics is then obtained by recalculating the strengths using $C = g^{-1}(s) \cdot f(U, s)$, the elementary f being the only element to change between steps of the integration of the rigid body motions. For a two-dimensional example with two spatially separated lifting surfaces these equations become:

$$\begin{bmatrix} a_{1_1 1_1} & a_{2_1 1_1} & \cdots & a_{N_1 1_1} & a_{1_2 1_1} & a_{2_2 1_1} & \cdots & a_{N_2 1_1} & a_{W_1 1_1} & a_{W_2 1_1} \\ a_{1_1 2_1} & a_{2_1 2_1} & \cdots & a_{N_1 2_1} & a_{1_2 2_1} & a_{2_2 2_1} & \cdots & a_{N_2 2_1} & a_{W_1 2_1} & a_{W_2 2_1} \\ \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots \\ a_{1_1 N_1} & a_{2_1 N_1} & \cdots & a_{N_1 N_1} & a_{1_2 N_1} & a_{2_2 N_1} & \cdots & a_{N_2 N_1} & a_{W_1 N_1} & a_{W_2 N_1} \\ a_{1_1 1_2} & a_{2_1 1_2} & \cdots & a_{N_1 1_2} & a_{1_2 1_2} & a_{2_2 1_2} & \cdots & a_{N_2 1_2} & a_{W_1 1_2} & a_{W_2 1_2} \\ a_{1_1 2_2} & a_{2_1 2_2} & \cdots & a_{N_1 2_2} & a_{1_2 2_2} & a_{2_2 2_2} & \cdots & a_{N_2 2_2} & a_{W_1 2_2} & a_{W_2 2_2} \\ \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots \\ a_{1_1 N_2} & a_{2_1 N_2} & \cdots & a_{N_1 N_2} & a_{1_2 N_2} & a_{2_2 N_2} & \cdots & a_{N_2 N_2} & a_{W_1 N_2} & a_{W_2 N_2} \\ 1 & 0 & \cdots & -1 & 0 & 0 & \cdots & 0 & 1 & 0 \\ 0 & 0 & \cdots & 0 & 1 & 0 & \cdots & -1 & 0 & 1 \end{bmatrix} \begin{Bmatrix} \mu_{1_1} \\ \mu_{2_1} \\ \cdots \\ \mu_{N_1} \\ \mu_{1_2} \\ \mu_{2_2} \\ \cdots \\ \mu_{N_2} \\ \mu_{W_1} \\ \mu_{W_2} \end{Bmatrix} = \begin{Bmatrix} V_{1_1} \\ V_{2_1} \\ \cdots \\ V_{N_1} \\ V_{1_2} \\ V_{2_1} \\ \cdots \\ V_{N_2} \\ 0 \\ 0 \end{Bmatrix}$$

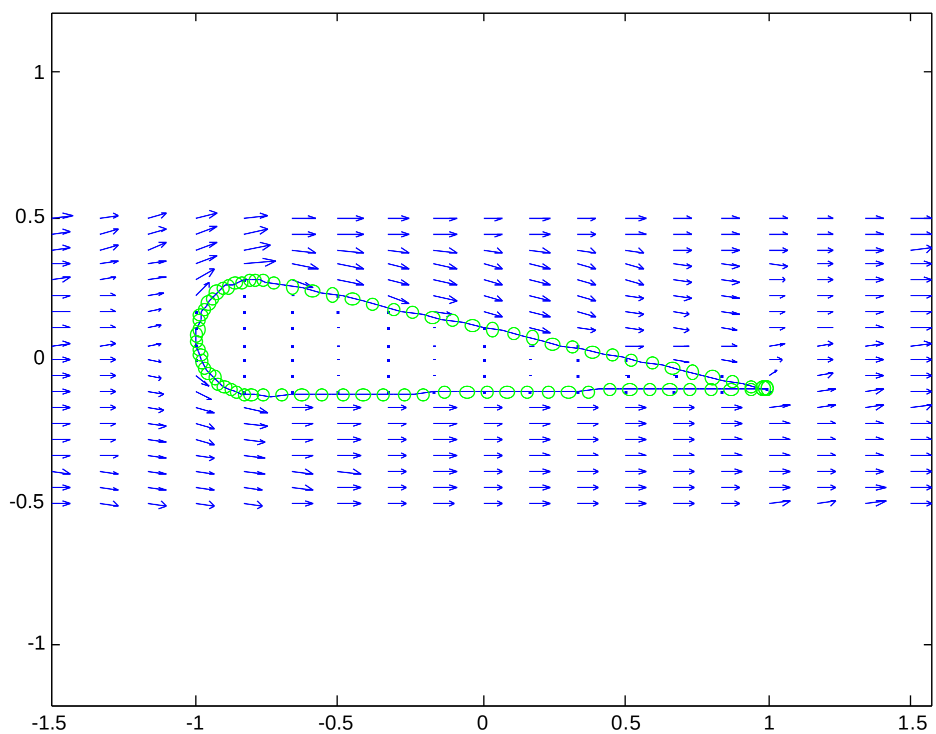
wherein the symbols in the table below have been used.

| **Parameter** | **Definition** |
|---|---|
| Boundary Condition<br>$V_{i_J} \equiv -U_\infty \cdot n_{i_J}$ | Free stream velocity normal to panel i on surface J |
| Influence Coefficient<br>$a_{i_I j_J} \equiv (\mu)_{i_I j_J} \cdot n_{i_I}$ | The contribution of a unit strength singularity from control point j on Surface J to control point i on surface I |
| Singularity Strength<br>$\mu_{i_I}$ | The strength of the singularity at control point i on panel I |
| $N_I$ | Number of panels on surface I |

5. Given C, you have everything you need to calculate the flow velocities at any point in space or, on the surfaces of the vehicle. Just superimpose potential functions for the source/doublets of each panel with the strengths given and you have the total potential function for the flow. The velocity at any point is given by: $\mathbf{w} = grad(\Phi)$ and the

pressures can be calculated with the aid of the Bernoulli equation above. In this case the function f is usually a constant evaluated at the initial time at a infinite distance from the vehicle where w=0. This is because w is the increment in velocity over the offset flow and is zero at infinity.

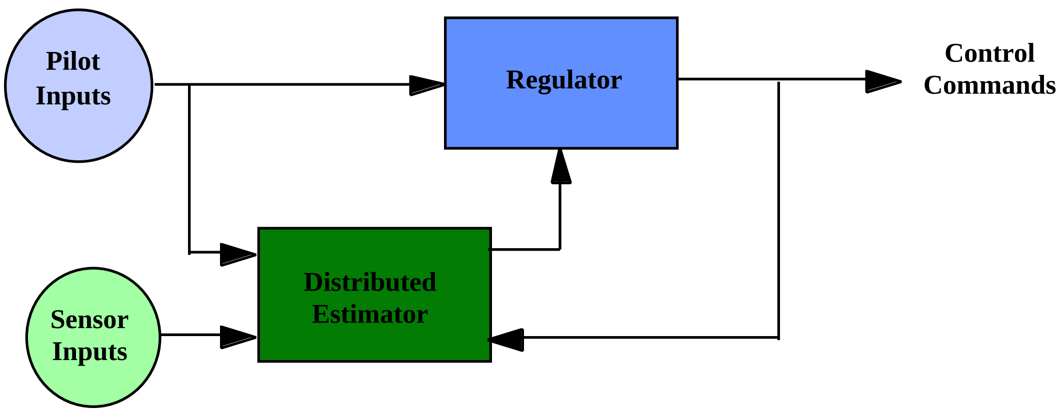
Below is an example of an airfoil in two dimensions. Nodes of the panel ends are circled and the flow field calculated using the above technique is show in a so-called quiver plot, a plot of the velocity showing both magnitude and direction, at points in the field. Close examination of the flow reveals that the boundary conditions at the trailing edge are not quite satisfied. this is because of the strength of the wake vortex which generates lift and is concentrated at the trailing edge. This particular airfoil has a shape that is being considered for use in a study of synthetic jet effectiveness at Langley Research Center.




In three dimensions wakes and lifting surfaces complicate the picture greatly but generally, the Kutta condition can be applied at the trailing edge of all lifting surfaces to calculate lift and to generate wake surfaces which are stream surfaces. This assumes that a trailing edge is defined for all lifting surfaces. The trailing edge may not have a good definition on say, a fuselage but some separation point is required and must be assumed.

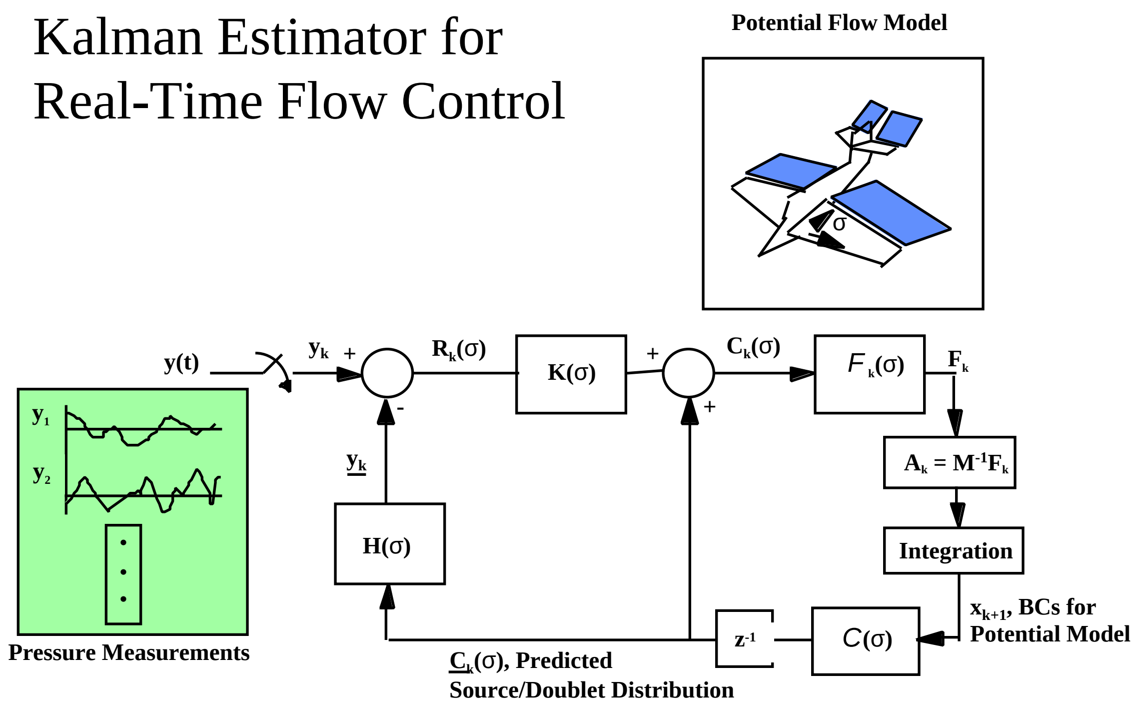
## Flight Control Architecture

The architecture proposed herein incorporates a distributed, real-time, state estimator that drives a regulator. This is, of course, a gross simplification of what is required but serves to highlight the novel ingredients from the research perspective. Pilot inputs of the form of control stick deflections or mode switches are input into both the Regulator and Estimator blocks and Sensor Inputs of the form of angular rate information, accelerometer measurements, control surface transducers, pressure transducers, valve/switch closure indicators, and possibly other signals, are input to the Estimator section. The function of the Estimator is to determine the state of the aircraft and model its evolution given the sensor data, control commands, and the current estimated aircraft state. The Regulator takes the aircraft past and present states as well as commands from the pilot and generates commands to the aircraft flight controls which may include conventional surfaces, pressure ports, MEMS devices, slew commands for nozzles, and others. This architecture, in a non-distributed form, has been successfully employed by Orbital Sciences in their Pegasus launch vehicle, references 4 and 5. From the perspective of this paper, the main difference between this architecture and that employed in, say, Pegasus is that the Estimator block will include a distributed model of the aerodynamics of the vehicle to adequately model high frequency aerodynamic phenomena and distributed control devices.




**Flight Controller Architecture**

The Estimator is structured after the normal Kalman filter, reference 8, except that aircraft forces and moments will be provided by a potential model, possibly adjusted for the effects of skin friction, separation, and other effects not directly modeled by potential theory. This is illustrated in the skecth below wherein the vector y consists of all sensor inputs. The state is $x_k$ includes the source/doublet distribution, $C_k(\sigma)$, of the aerodynamic model as well as the normal rigid body dynamics variables used in the dynamics and control of aircraft. This source/doublet distribution is a function of the surface coordinate variable, $\sigma$. The Kalman gain K as well as the sensor mapping variable H are also functions of $\sigma$ as is the map $F_{\mathbf{k}}(\sigma)$ which generates the aerodynamic forces and moments, $F_k$, given the source/doublet distribution, $C_k(\sigma)$. Finally, given the aerodynamic forces and moments, the acceleration, $A_k$, is numerically integrated to predict the future state of the vehicle at the next computer sample time, including the next source/doublet distribution. This is then used to predict the measurements, $y_k$, be they pressures or other more conventional measurements, through the contraction operator H. Finally, once the measurements at the next interval are sampled, the Kalman gain, $K(\sigma)$, is used to operate on the innovations sequence, $R_k(\sigma)$, the difference between the measurements estimated during the last sample and actual measurements taken, to adjust, or correct, the source/doublet distribution prediction based on the actual, sampled measurements. This gain operator allows the designer to tune the final estimate based on the best qualities of the potential model and the measurement system.




**Estimator Architecture**

## Suggestions for Future Research

The suggestions put forth here are directed to control theoretic development to support the use of active flow control in maneuvering of aircraft as opposed to device development of synthetic jets, active porosity, or other concepts that might be used for such. With that caveat, one needs models of the devices that are consistent with potential theory. These are generally models that represent the macroscopic effect of the device rather that a detailed model it. An example might be to represent a synthetic jet as a doublet. Realizing that a doublet in a uniform stream produces a cylinder, one might extend this idea to modelling the effect of a synthetic jet as a virtual surface distortion, or bump. Establishing valid models of the effect of active flow control devices is a fruitful area of research in and of itself. This requires correlation with experiment and is currently in progress. Past that developing an effective control law architecture still remains and is the subject of the suggestions that follow.

Although significant differences exist between two- and three-dimensional flow which center around the very nature of space, dictating the way vorticity is modeled, e.g. the space surrounding conventional airfoil in three dimensional space is simply connected whereas it is not in two dimensional space, focusing on a two-dimensional problem will remove many of the more tedious difficulties such as flow visualization from consideration so that the control problems of implementation can be effectively addressed and, is therefore recommended.

One may use a two-dimensional airfoil, such as referred to herein, and test the proposed architecture on that system in a simulated wind-tunnel environment. Synthetic jets, as described herein, could be the actuation concept. They would be mounted on the top surface along with pressure transducers to serve as sensor inputs. If this tack were taken it would be necessary to model the effect of the synthetic jet in a macroscopic manner, that is, create a potential code compatible model as mentioned above. This could be to model the synthetic jet as a doublet or as a virtual displacement, or bump, on the top surface of the airfoil. The control process objective would be to modulate lift at contstant angle of attack. The output of the work would be suggestion for locating the jet and pressure transducers for optimal control effectiveness as well as a preliminary evaluation of the control architecture proposed herein.

Another suggestion would be to use multiple airfoils and simulate a complete flight vehicle in two dimensions, modelling only the wing and tail, considering them as rigidly connected. Again the one must model the control device with a potential theory compatible model but, the vehicle model should produce interesting results relative to the use of distributed models in maneuvers.

Either of these suggested focus applications will profit from accelerated codes to deal with the flow solver, not focusing on accuracy only, but, studying the tradeoff between model accuracy and computational speed which governs the real-time suitability of the system.

# References


1. Etkin, B.: Dynamics of Flight - Stability and Control. John Wiley and Sons, Inc., New York, 1959.
2. Blakelock, John H.: Automatic Control of Aircraft and Missiles. John Wiley and Sons, Inc., New York, 1965.
3. Kant, Pieter, editor: Theory and Applications of Optimal Control in Aerospace Systems. AGARDograph No. 251, July 1981.
4. Rovner, D. "GN&C For Pegasus Air-launched Space Booster - Design and First Flight Results" SAE Paper 911105, Presented at the SAE Aerospace Atlantic Conference, Dayton, OH, 22-26, April, 1991.
5. Rovner, D. and A. Elias, "Air-Launched Orbital Booster GN&C Requirements - The Pegasus Design Experience", AAS Paper 89-041, Published in Guidance And Control 1989: Proceedings of the Annual Rocky Mountain Guidance And Control Conference, Keystone, CO, Feb 4-8 1989, Univelt Inc., San Diego, 1989.
6. Bryan, G. H.: Stability in Aviation. Macmillan Co., London, 1911.
7. Pisano, Albert: MicroElectroMechanical Systems (MEMS) at DARPA. Summer, 1997. http://web-ext2.darpa.mil/ETO/MEMS/MEMS-OV.
8. Sage, A. P. and J. L. Melsa: Estimation Theory. McGraw-Hill, Inc. New York, 1971.
9. Montgomery, R. C. and D. Ghosh: On the Use of Distributed Sensing in Control of Large Flexible Spacecraft. ASME publication G00533, Advances in Dynamics and Control of Flexible Spacecraft and Space-Based Manipulators, DSC-Vol. 20, November 1990, pp. 7-12.
10. Prandtl, L. and O. G. Tietjens: Fundamentals of Hydro- and Aeromechanics. Dover Publications, Inc., New York, 1957.
11. Katz, J. and A. Plotkin: Low-Speed Aerodynamics - from Wing Theory to Panel Methods. McGraw-Hill, Inc. New York.
12. Ashby, D. L., Dudley, M. R., and S. K. Iguchi: Development and Validation of an Advanced Low-Order Panel Method. NASA TM 101024, Oct. 1988.