An Experimental Investigation of VTOL Flying Qualities Requirements in Shipboard Landings

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This in flight simulation experiment using the U S Navy X 22A variable stability V/STOL aircraft operated by Calspan was undertaken to generate data for the development of flying qualities and flight control system design criteria for hover and low-speed flight In particular the experiment emphasized the flying qualities characteristics of inertial translational rate control systems in the context of the visual landing mission aboard small aviation capable ships under high wind over deck and sea state conditions Pilot rating data indicate that with suitable command gains velocity response time constants up to about 2 5 s provide satisfactory flying qualities Minimum and maximum satisfactory command gains were determined for a range of velocity time constants

Introduction

NEXT generation Navy VTOL aircraft will be required to operate from small aviation ships under conditions of reduced visibility higher sea states and more severe winds compared to today s Navy helicopters For example the near term goal of the NAVTOLAND SH 2F demonstrator development program was the performance of landings with zero ceiling visibility of 700 ft and in environments up to Sea State 5 One of the factors limiting the operational capability of current VTOL aircraft including helicopters with pitch and roll attitude command and heading hold flight control systems is the high pilot workload associated with stabilization of the aircraft in the presence of ship airwake ambient turbulence and large deck motion Next generation nonhelicopter VTOL s with high disk loading will experience similar difficulties but the pilot workload may be exacerbated by lower effective thrust to weight ratio and moment control power Inertial translational rate command (TRC) systems seem to show considerable promise to alleviate many of the stability and control difficulties in the landing flight phase because of their inherent gust proofing characteristics and relief of the pilot s inner loop attitude stabilization role Furthermore ground based simulator studies indicate that these benefits can be realized with reduced moment control power compared to less sophisticated augmentation systems such as rate or attitude command [1] The recent development of small accurate guidance sensors such as the microwave landing system (MLS) with precision ranging capability has made the implementation of such flight control systems practicable in the small ship environment

Depending on vehicle configuration horizontal translational rate command can be realized either through modulation of X and Y forces or through tilting the thrust vector by rotation of the entire vehicle that is attitude type TRC Direct force attitude and blended TRC systems have been implemented in flight vehicles[2 3] and investigated in ground based simulators [4 6] Unfortunately most of these programs involved specific control configurations so the data base for criteria development is limited These data do indicate however that with optimized dynamics attitude type TRC systems can provide satisfactory flying qualities in demanding hovering tasks although the minimum translational time constant achievable appears to be limited by the magnitude of the attitude excursions required Direct force implementations obviate this difficulty but can produce uncoordinated cockpit lateral accelerations in response to control inputs Possibly the greatest factor against direct force implementations is that unless these devices are also required for other purposes (i e , transition or yaw control) their incorporation in an air vehicle will impose penalties of weight complexity and cost since they will not supplant moment controllers Clearly direct force control implementations will be included in advanced VTOL s only if the requisite performance and flying qualities cannot be provided by attitude type TRC systems

This paper describes an in flight simulation program using the U S Navy X 22A variable stability aircraft whose objectives were to provide stability and control information for attitude type TRC systems in a task representative of VFR shipboard landing Since this task is clearly three dimensional, a brief examination of requirements for augmentation of height control was also included In addition, the experiment addressed the question of force and moment control power requirements for these control implementations The following three sections describe the development of the evaluation task the research flight control system mechanizations and the conduct of the experiment The paper concludes with a presentation of selected results and conclusions A detailed description of the experiment can be found in Ref 7

Task Development

Use of Headup Display

It was recognized at the outset of the program that the validity of the experimental results would be related directly to the fidelity of the visual and motion cues and the degree to

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which these stimuli evoke pilot control and stabilization activities similar to the real world For in flight simulators fidelity of motion cues is generally no problem However for this program a particular challenge was the provision of visual cues equivalent to the near ship environment

Since flight safety considerations precluded prolonged hovering at altitudes less than 50 ft or near external structure duplication of a landing pad and ship superstructure was judged impractical Self paced pilot maneuvering over ground markings was also considered but was rejected because of the difficulty of ensuring that the pilot would approach such an unstructured evaluation with the same dynamic performance standards particularly regarding the ability to rapidly reposition the vehicle Furthermore at the minimum hover altitude of 50 ft the resolution of horizontal position cues would be poor and only limited vertical maneuvering would be possible

For the reasons cited previously the approach taken was to use a combination of the X-22A Headup Display (HUD) for position information and the real world for visual orientation cues The display format devised after preliminary ground and in flight testing by Calspan and Navy pilots is presented in Fig 1 The information presentation is symbolic since the field of view of the HUD effectively precludes a pictorial presentation of the landing pad and the ship superstructure The salient features of the display are the fixed aircraft symbol and altitude ladder with rung separation scaled to 10 ft Longitudinal and lateral displacement from the landing pad (square symbol) are presented in planview in a heading up axis system Referring to Fig 1 closure with the pad would require forward and right stick Height above the landing pad is depicted by the separation of the double dumbbell and the airplane symbol in effect an elevation view of the vertical situation As in the X Y situation presentation the control sense is fly to that is the dumbbell symbol is the landing pad Orientation information pitch and roll attitude and heading was not displayed on the HUD since these cues were derived from the outside world

Discrete Tracking Task (HUD Task)

Initial efforts to devise an evaluation task were directed to achieving as much realism as possible To this end a pre recorded ship motion signal representative of Sea State 5 conditions was prepared using a sum of sine waves model [8] In preliminary ground simulator and in flight evaluations it was found that the majority of the flight time was spent in station keeping hover referenced to the pilot's estimate of the mean deck position Significant control activity resulted only following the detection of a lull in the deck motion where upon the pilot attempted to close with the pad and land Since the aircraft displacement from the landing pad was displayed as an error signal the pilot had difficulty distinguishing control induced motion of the aircraft from deck motion Consequently it was concluded that this task would be both wasteful of valuable flight time and unrealistically difficult from the standpoint of relative motion perception

As a result the ship motion model was abandoned In its place was substituted a tape recorded series of semirandom discrete changes in landing pad position Pad position changed once every 20 30 s in increments of 20 25 ft and in any combination of X Y or Z directions

This task had two distinct advantages over either the ship motion model or ground referenced maneuvering First since the pad changed location discretely the pilot could distinguish pad motion from aircraft motion easily Second the task was paced; that is the pilot knew he had a finite amount of time in which to reposition himself over the pad Consequently it was felt that the pilot was more likely to apply a consistent performance standard to all con figurations

After a review of helicopter small ship landing procedures and interviews with LAMPS pilot the shipboard landing task was broken down into a sequence of maneuvers or subtasks and an equivalent sequence was devised for implementation in the X 22A flight program This sequence is comprised of

1) X Y maneuvering and precision hovering at constant altitude moving into position over landing pad at termination of approach

2) Y Z maneuvering while maintaining X position positioning over the pad prior to landing maneuvering to avoid deck contact during landing, maintaining clearance with ship superstructure

3) Descending to deck (establishing desired sink rate and descending while maintaining X Y position)

During the X 22A simulation the gross maneuvering was paced by the pilot's desire to track the discretely changing pad location while the precision hovering requirement was satisfied during the period of no pad motion following the discrete changes One half symbol width (± 5 ft) deviation from the pad was used as a performance standard for satisfactory precision hovering Configuration evaluations were comprised first of a period of tracking discrete step changes in X Y pad position then in Y Z pad position Finally with the discrete tracking task off the pilot executed simulated vertical landings to the deck During the program a separate series of evaluations was performed using a ground referenced task to allow comparisons with the discrete tracking task

Disturbances in the tracking task were introduced both through ambient winds and turbulence and through the in sertion of artificial random ship airwake turbulence using the zero mean random component of the airwake mathematical model described in Ref 8 The model was representative of conditions at 15 ft altitude above the landing pad with the aircraft pointed into the wind which was coming from 30 deg to port of the ship's heading Root mean square wind velocity was 8 9 ft/s in all three orthogonal directions corresponding to a wind over deck of 25 knots The control surfaces of the X 22A were moved in such a way that the resulting pitching rolling yawing and vertical motion of the X 22A equaled the motion it would experience in such a wind environment

The random ship airwake turbulence transfer functions as mechanized for this experiment are given below

$$u_g(s) = \left[\frac{ku_g\omega_{ng}^2}{s^2 + 2\zeta_g\omega_{ng}s + \omega^2{}_{ng}}\right]\eta_1(s)$$

$$v_g(s) = \left[\frac{kv_g\omega_{ng}^2}{s^2 + 2\zeta_g\omega_{ng}s + \omega_{ng}^2}\right]\eta_2(s)$$

$$w_g(s) = \tfrac{1}{2}u_g(s) + \tfrac{1}{2}v_g(s)$$

where u_g v_g and w_g are the X Y and Z components of airwake turbulence and where η_1 and η_2 are Gaussian white noise sources from independent analog noise generators whose gains were adjusted to produce root mean square velocities of 8 9 ft/s in each component of velocity Limited onboard hardware precluded independent generation of the w component of airwake velocity The natural frequency w_{ng} was 1 1 rad/s and the damping ratio ζ_g was 0 4

Experiment Design

Longitudinal/Lateral Flight Control System

The objective of the longitudinal and lateral flight control system parameter variations was to generate a data base for the correlation of flying qualities with the vehicle's closed loop modal characteristics Accordingly analyses were performed to establish gain ranges for command and feed back parameters for the experiment matrix and to relate the estimated closed loop dynamics to previous experimental investigations In the discussion to follow the longitudinal axis is used for illustration; the behavior of the augmented lateral axis is similar and is not discussed separately

The velocity command control systems implemented for this experiment are comprised of inner loop attitude stabilization with an outer velocity loop In structure the longitudinal and lateral systems are identical The inertial velocity signals for all axes are derived from complementary filtering of onboard accelerometer and MLS position data resolved into an aircraft heading axis system Additional workload relief functions such as auto trim (i e forward loop integration) were considered but not implemented on the basis that the simulated atmospheric disturbances were zero mean and the task primarily involved maneuvering Accordingly velocity trim was accomplished manually through the X 22A parallel trim system

No attempt was made to modify the stability and control characteristics of the X 22A to match a specific VTOL configuration Since the disk loading of the X 22A lies between that of helicopters and jet lift VTOL's its inherent dynamic characteristics (with the possible exception of drag damping) are considered representative for a generalized VTOL flying qualities investigation

Assuming a decoupled vertical mode the augmentation system depicted in Fig 2, produces closed loop transfer functions for the primary velocity response and the secondary attitude response of the form:

$$\frac{\dot{x}}{\delta_{es}} = \frac{X_{\delta_{es}}(s^2 - M_q s - gM_{\delta_{es}}/X_{\delta_{es}})}{(s+\lambda)(s^2+2\zeta\omega_n s+\omega_n^2)}$$

$$\frac{\theta}{\delta_{es}} = \frac{M_{\delta_{es}}(s - X_u + M_u X_{\delta_{es}}/M_{\delta_{es}})}{(s+\lambda)(s^2+2\zeta\omega_n s+\omega_n^2)}$$

The modal parameters (λ ζ ω_n) are uniquely determined by the feedback gains $K_{\dot{x}}$ K_q and K_θ For a vehicle like the X 22A whose moment controllers generate nearly perfect couples the force derivative $X_{\delta_{\gamma\lambda}}$ is nearly zero Thus the numerator of the velocity transfer function is approximately a pure gain

The ground simulator experiments described in Refs 1 and 5 indicate a pilot preference for well damped velocity responses (i e little or no overshoot) which implies a lower

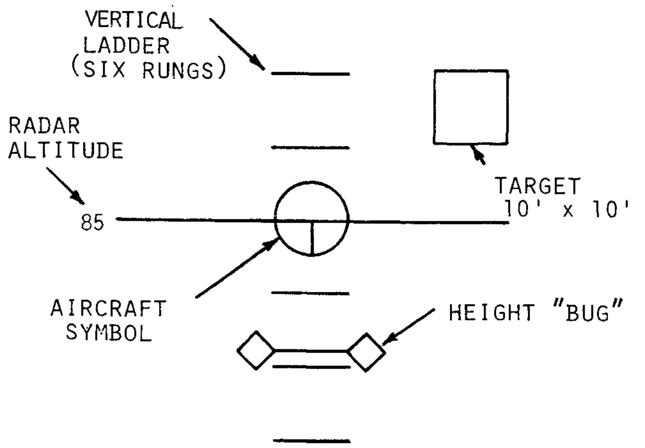

Fig 1 Headup display symbology for hovering task

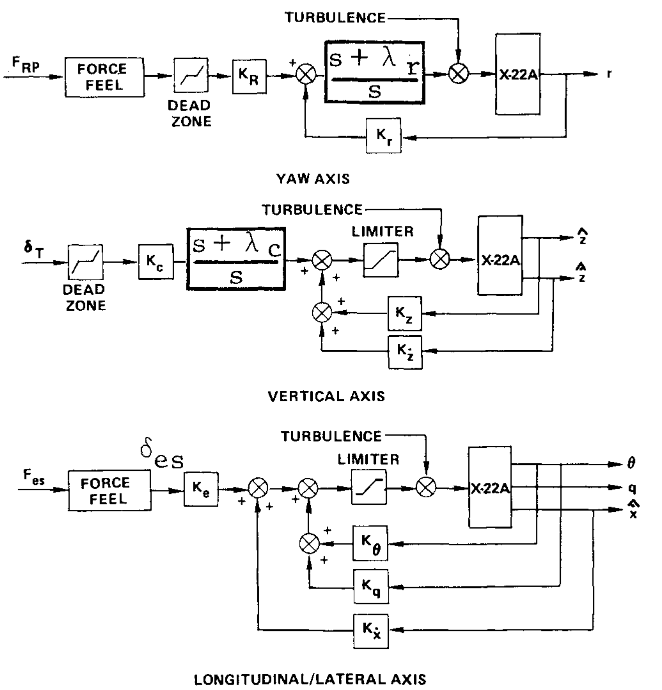

Fig 2 Flight control system block diagrams

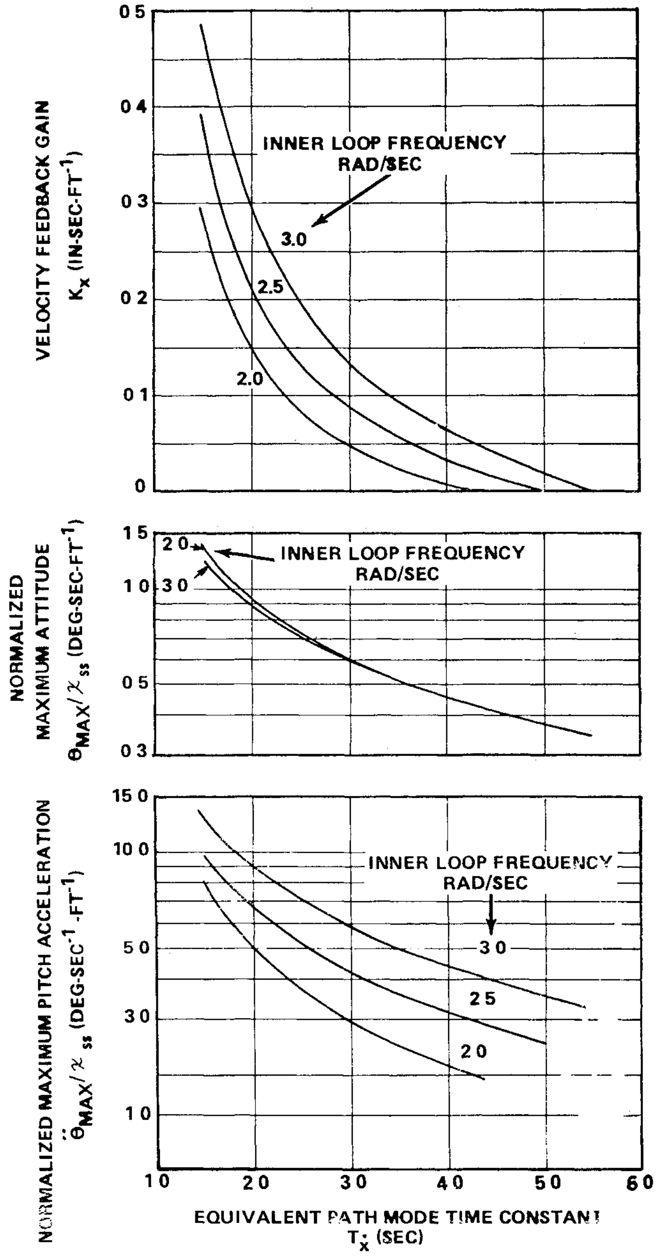

Fig 3 Longitudinal TRC design and response parameters

limit for the damping ratio In fact the majority of evaluations in the Ref 1 experiment were conducted with a binomial form characteristic equation that is $\lambda = \omega_n$ and $\zeta = 1\ 0$

With this pilot preference for high system damping Ref 5 notes that the velocity response approximates that of a first order system that is

$$\frac{\dot{x}}{\delta_{es}} \approx \frac{K_{\dot{x}c}}{T_{\dot{x}}S+1} \text{ and } \frac{\dot{y}}{\delta_{as}} \approx \frac{K_{\dot{y}c}}{T_{\dot{y}}S+1}$$

Reference 5 also proposes, as a possible criterion, the equivalent path mode time constant $T_{\dot{x}}$ (i e the time to 63% of steady response) The appeal of such a simple flying qualities criterion is obvious since it reduces the number of modal parameters required for correlation from three to one and the system can be described simply in terms of the steady state velocity gain $K_{\dot{x}_c}$ (ft/s/in), and the equivalent path mode time constant, $T_{\dot{x}}$ The question, however, is whether the equivalent path mode time constant adequately addresses all of the characteristics of importance to flying qualities For example the data of Refs 1 and 5 indicate that a limiting factor for the minimum (fastest) realizable path mode is the magnitude of the initial attitude transient As will be discussed, the abruptness of the attitude response of TRC systems is not completely governed by the path mode time constant Furthermore $T_{\dot{x}}$ is a measure of velocity control response characteristics only and does not reflect the relative contribution of aerodynamic and inertial speed stability to gust and turbulence sensitivity

Specifically the path mode time constant for any level of inner loop attitude stabilization is a function of the sum of aerodynamic speed stability M_u and inertial velocity stability $M_{\dot{x}}$ As indicated in Fig 3, even with zero velocity feedback, the X 22A exhibits a finite path mode time constant due to M_u Although M_u and $M_{\dot{x}}$ are equivalent from a control response standpoint in disturbed air, M_u tends to couple the aircraft to the air mass while $M_{\dot{x}}$ acts to suppress the inertial velocity disturbances

From a design standpoint, high frequency inner loop attitude augmentation would appear advantageous since for a given path mode higher gain velocity feeback is required The resulting increased ratio of $M_{\dot{x}}$ to M_u thus produces greater immunity to turbulence In terms of the secondary responses the maximum attitude is relatively invariant with level of inner loop attitude stability (Fig 3) However, the pitch attitude abruptness a measure of which is the maximum pitch acceleration $\ddot{\theta}_{max}/\dot{x}_{ss}$, is a strong function of the inner loop frequency Increasing inner loop frequency from 2 to 3 rad/s doubles the effective abruptness at any path mode (Fig 3) It is clear therefore, that the level of gust and turbulence immunity and attitude abruptness as well as the velocity response time constant, $T_{\dot{x}}$, are interrelated factors influencing TRC flying qualities These factors were addressed in the experiment design by implementing several path mode time constant configurations with variations in inner loop attitude mode natural frequencies

The X 22 force feel system was configured so that the response of stick deflection δ_{es} to pilot pitch force F_{es}, and roll force inputs was second order with a natural frequency of 12 rad/s and damping ratio of 0 6 Pitch and roll stick force gradients were selected to be 2 5 and 2 lb/in respectively, while the pitch and roll stick breakout forces were 1 5 and 1 0 lb, respectively There was no dead zone in stick deflection

Yaw Axis Flight Control System

Because of the potentially large number of configuration variables, certain measures were taken to reduce the size of the experiment matrix First to focus on the characteristics of the translational dynamics the yaw axis augmentation was held fixed for the experiment and was designed to provide satisfactory, and, hence, unobtrusive flying qualities The achievement of this objective was verified in preliminary practice evaluations The system illustrated in Fig 2a provides yaw rate responses to pedal commands with pseudo heading hold through the action of the forward loop integrator The nominal heading dynamics were second order with a natural frequency of 2 0 rad/s, a damping ratio of 1 0 and a command gain of 8 0 deg/s/in of pedal

Vertical Axis Flight Control System

The X 22A exhibits the inherent low vertical damping typical of intermediate disk loading VTOL's Thus, height control augmentation was considered essential for the evaluation task described previously The system depicted in Fig 2b, can operate in two modes In the first mode inertial vertical velocity and position feedback to the blade collective pitch were employed to produce an altitude stabilized configuration Vertical rate responses to throttle commands were achieved by means of an integral proportional network in the throttle command path An electrical command dead zone centered on the throttle detent position assured zero vertical rate with the throttle in the detent (altitude hold) In the second mode the altitude feedback gain and the command integrator gain were set to zero thus providing rate responses to throttle commands but with no altitude hold capability

Although there exists a substantial body of data relating to augmented height dynamics, the applicability of most of these data to the current program was considered questionable because of uncertainties in the tasks employed and because much of the experimental effort was concerned with augmentation of aerodynamic vertical damping However, from a control standpoint, the data indicated a pilot preference for some minimum level of vertical damping of the order of 0 5 to 1 $0 s^{-1}$

Because of pilot preference for first order like velocity responses to control commands, a quasimodel following approach was used with the altitude stabilized control systems For each vertical augmentation configuration, the altitude and altitude rate feedback gains were selected to realize a critically damped second order characteristic equation The integral proportional prefilter gains were then selected to cancel one of the characteristic roots The remaining real root then determined the time constant of the vertical velocity response This design approach was taken to allow direct comparison of vertical rate command configurations with and without the altitude hold feature

Experimental Matrix

To address the utility of the path mode time constant, the approach taken in the experiment design was to first establish a baseline configuration matrix comprised of variations in equivalent path mode and steady state velocity gain Then, having established the sensitivity of flying qualities to these parameters, selected configurations from the baseline matrix would be repeated with configurations having identical path

Table 1 Flight operational summary

Pilot	No of configurations evaluated	No of evaluation flights	Flight hours
A	84[a]	21	20 0
B	16	4	3 1
C	6	2	1 6
Total	106	27	24 7

[a]This included 58 HUD tasks 10 ground referenced maneuvering tasks and 16 abbreviated HUD tasks involving side to side and vertical maneuvering only

mode and gain characteristics but with higher and lower frequency inner loop attitude dynamics

The starting point for the baseline matrix was a well damped inner loop attitude system having a nominal natural frequency of 2 5 rad/s and a nominal damping ratio of 0 8 Velocity feedback gains were selected to produce path modes ranging from 1 5 to 4 s since Refs 1 and 5 indicate that these values span the region of satisfactory flying qualities Additional configurations designed for identical path mode and velocity gain characteristics as the primary matrix but with different attitude response and gust sensitivity characteristics were based on inner loop attitude dynamics of 2 0 and 3 0 rad/s natural frequency

Conduct of the Experiment

Specific flight procedures for this experiment involved one of three tasks: the HUD task, ground referenced maneuvering task or calibration record gathering The primary configuration discriminant the HUD task consisted of the pilot following a semirandom discrete position command displayed on the HUD The ground referenced maneuvering task was flown without the HUD and involved the pilot maneuvering the aircraft around a pattern of runway and taxiway markings on the airport Nominal altitude was 75 ft for the HUD task and 50 ft for the ground referenced task Calibration record gathering was accomplished by the pilot applying special manual inputs designed to enhance the accuracy of the parameter identification process discussed in the next section

Engineering data for this experiment came from two sources: first a telemetry link and second through taped pilot commentary and Cooper Harper pilot ratings. All telemetry data were recorded on magnetic tape for subsequent analysis

Three test pilots were involved in this handling qualities experiment The primary pilot (pilot A) and the secondary pilot (pilot B) were Calspan engineering pilots with approximately 120 and 60 h of helicopter flight time respectively Pilot C was a LAMPS qualified U S Navy helicopter pilot with extensive handling qualities simulation experience A summary of the flight operations flown is provided in Table 1

System Calibration

This section describes the methods used to determine the X 22A closed loop dynamics from actual flight measured responses using parameter identification methods

At the start of the experiment an identified mathematical model of the X 22A was available from earlier flight experiments [9,10] This model employing linear aerodynamics

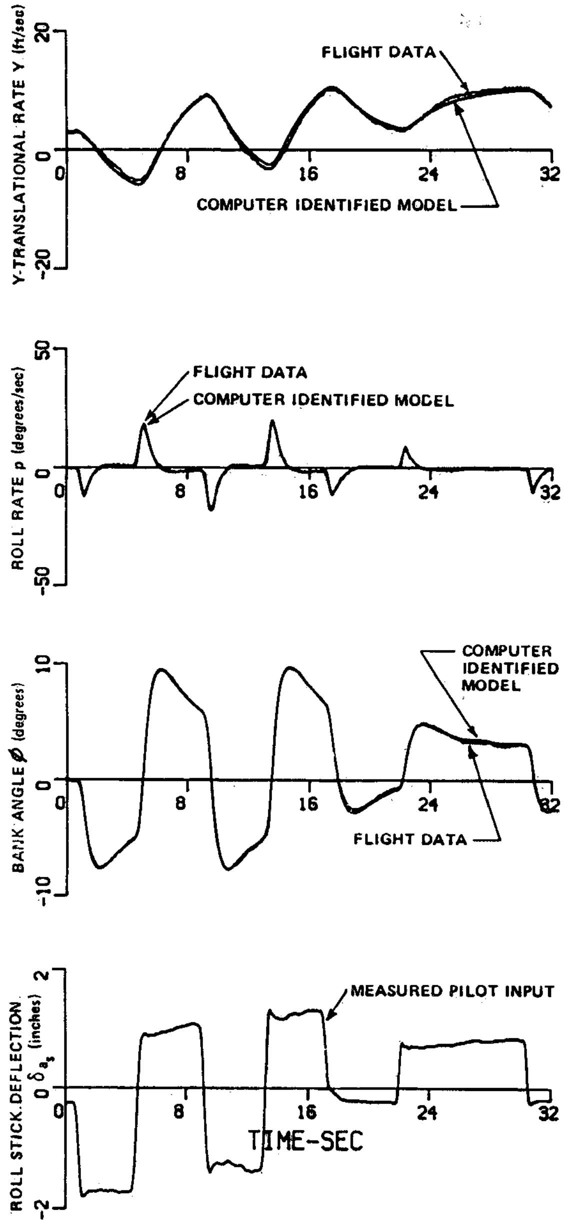

Fig 4 Overplots of lateral directional flight data on computer identified mathematical model response to flight measured pilot input

Table 2 Transfer functions for configuration 207 A

Identified results[a]	Nominal third order model (experiment design)
$\dfrac{\dot{x}(s)}{\delta_{es}(es)} = (-2\ 465s - 29\ 12)/D_e(s)$	$\dfrac{\theta(s)}{\delta_{es}(s)} = (-0\ 2976s^2 + 0\ 06844s - 23\ 32)/D_e(s)$
$\dfrac{\theta(s)}{\delta_{es}(s)} = (57\ 10s + 9\ 971)/D_e(s)$	$\dfrac{\theta(s)}{\delta_{es}(s)} = (41\ 49s + 5\ 968)/D_e(s)$
$D_e(s) = s^3 + (5\ 984s^2 + 9\ 978s + 2\ 781)$	$D_e(s) = s^3 + 4\ 233s^2 + 7\ 165s + 1\ 794$
$\dfrac{\dot{y}(s)}{\delta_{as}(s)} = (I - 3\ 788s + 36\ 46)/D_a(s)$	$\dfrac{\dot{y}(s)}{\delta_{as}(s)} = (-1\ 211s + 25\ 28)/D_a(s)$
$\dfrac{\phi(s)}{r_{as}(s)} = (73\ 81s + 12\ 47)/D_a(s)$	$\dfrac{\phi(s)}{\delta_{as}(s)} = (44\ 97s + 2\ 698)/D_a(s)$
$D_a(s) = s^3 + (7\ 386s^2 + 13\ 29s + 5\ 127)$	$D_a(s) = s^3 + 4\ 311s^2 + 1\ 944$

[a] Units are ft/s and deg

with nonlinear gravitational and kinematic terms was linearized and became the nominal model for the purpose of feedback gain selection in this experiment However because of the limited data base from which the model was obtained a more thorough identification effort was required to ensure that the closed loop dynamics for this experiment were ac curately known

System calibration involved the use of advanced parameter identification methods to determine the parameters of a constant coefficient linear differential equation model which best fit a set of measured X 22A closed loop responses when the aircraft was excited by carefully designed and executed test inputs The identification method was a batch processing algorithm [generalized partitioned identification algorithm (GPIA)] developed at Calspan[11 12] which accounts for the effects of random noise on the flight measurements Parameter identification was performed using the pilot's stick and throttle displacements as the inputs and the closed loop vehicle responses as the outputs

Identification Results

Twenty two pitch and twenty two roll records were taken for identification purposes These spanned the entire range of control system dynamics and control sensitivity studied in this experiment

Engineering outputs from the parameter identification algorithm consisted of the following information

1) Identified transfer functions the resulting pole and zero locations and path mode time constant

2) Integration of the identified equations of motion forced with the measured control input and overplotted on top of the measured X 22A responses

An example of the latter for a roll stick input is shown in Fig 4 The excellent quality of the time history match (the flight data overlays the computer generated responses) is strong evidence that the model form and parameter values are correct

A direct comparison between identified and nominal transfer functions (4 s path mode time constant and 13 ft/s/in command path gain) for one flight condition is presented in Table 2 Closed loop identification results such as these plus results from many other configurations were combined with known feedforward and feedback gains to determine an updated open loop model of the X 22A The resulting updated unaugmented stability and control derivatives are shown in Table 3 together with the nominal unaugmented derivatives These data indicate that the pitch and roll control sensitivities are substantially higher than was believed previously As a result the loop gain of each augmented configuration was higher than the nominal value used in the experiment design

The effect of loop gain is illustrated by root loci of Fig 5 which show the variation of X 22A poles with lateral velocity feedback gain corresponds to a nominal 2 5 rad/s inner loop roll attitude stabilized system The higher loop gain for the updated model is reflected in increased damping of the closed loop poles (real roots rather than a complex pair)

A comparison between the path mode time constant and command path gain for the nominal and updated models with feedback in the lateral case is given in Fig 6 Data points with symbols represent individual parameter identification results for specific flight conditions The differences between the nominal and updated models in the longitudinal case were only one half as great as for the lateral case and are not presented Because of these differences pilot rating data have been correlated with the identified as opposed to the nominal dynamic parameters

Table 3 X 22A open loop third order model

Derivative	Updated model[a]	Nominal model[a]
X_u	−0 16	−0 15
X_q	−3 52	0
X_{δ_e}	0	−0 143
M_u	0 023	0 015
M_q	0 2	0 23
M_{δ_e}	0 479	0 348
Y_v	−0 175	−0 06
Y_P	3 67	−1 67
Y_{δ_a}	0	0
L_v	−0 038	−0 0148
L_P	−0 15	0 0698
L_{δ_a}	0 588	0 38

[a]Units are ft/s and rad

Experiment Results

The results presented herein are based primarily on the pilot ratings and commentary obtained during formal evaluations and observations made during the experiment A more detailed discussion of other aspects of the experiment such as task performance and control power utilization (pitch roll thrust) can be found in Ref 7

Height Augmentation

Preliminary evaluations revealed a pilot preference for vertical rate command with altitude hold It was found that

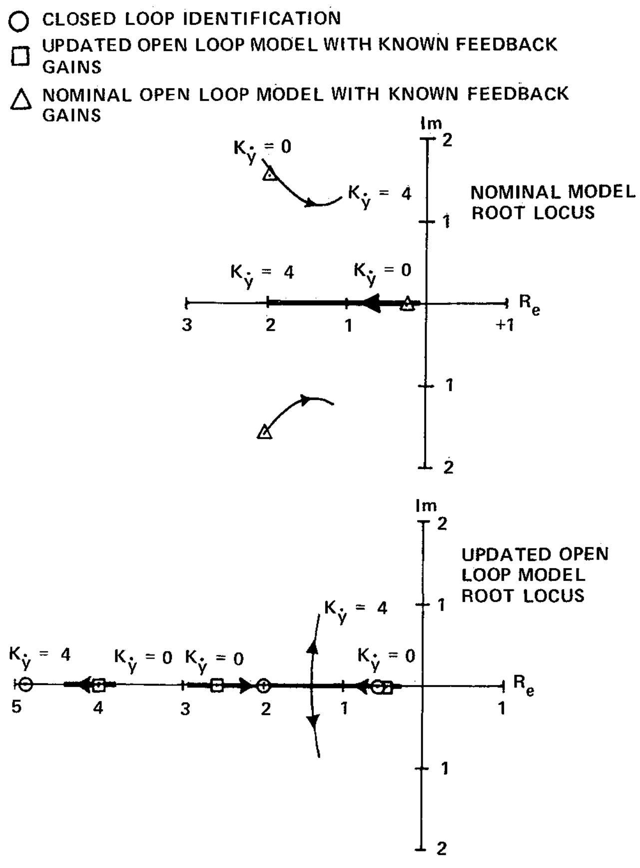

Fig 5 Comparison of lateral directional root loci as a function of velocity feedback gain

without altitude hold, horizontal translations tended to excite the vertical degree of freedom which increased workload significantly A series of evaluations was conducted to determine a minimum gain vertical configuration which would provide satisfactory control response and minimize height translational coupling The final system had a natural frequency of 0 75 rad/s and a damping ratio of 1 0 The effective throttle control sensitivity was 0 15 g/in Thus, the steady state velocity response time constant was 0 75 s This vertical augmentation was employed for the TRC system evaluations discussed in the next subsection

TRC Pilot Rating Results

Parameter identification results revealed certain differences between the nominal TRC dynamics used to design the ex periment and the dynamics of the configurations actually flown These differences result in lower than nominal steady state velocity gains and path mode time constants in both the longitudinal and lateral axes The differences are considered small in the longitudinal axis (approximately 10 20%) but for the lateral axis the differences are approximately 20% at short path modes to 40% at the longest path modes tested Since pilot commentary generally indicated that longitudinal control was the most difficult pilot ratings, in this paper, are correlated with the longitudinal dynamic characteristics It is recognized that for the long path mode configurations some lack of harmony in the longitudinal and lateral axes existed However, a detailed review of pilot commentary indicates that control asymmetry was not a significant factor in the evaluations [7]

The pilot rating data for evaluations of the baseline con figuration matrix (inner loop natural frequency 2 5 rad/s) by the primary and second pilot are summarized in Fig 7 The axes of this graphical presentation are the steady state velocity gain $K_{\dot{x}_c}$ and the equivalent first order time constant $T_{\dot{x}}$ described previously Pilot ratings for repeated evaluations have been averaged and the symbols are shaded to denote the level of flying qualities

In comparison to other flying qualities' data[1 5] for these control mechanizations the pilot ratings of the primary pilot indicate that the region of satisfactory flying qualities is small The lower velocity gain boundaries (PR = 3 5 and 6 5) are associated with large control forces and displacements required for maneuvering while the upper (PR = 3 5) is attributable to abruptness in attitude response and corresponds roughly to a pitch attitude acceleration of 80 deg/s^2/in For velocity gains within these upper and lower bounds only a limited range of path mode time constants produced satisfactory flying qualities Beyond a path mode of ap proximately 2 3 s, configurations were judged unsatisfactory because of lack of precision and predictability in velocity and position response The realization of path modes less than 1 5 s appears impractical because the maximum and minimum boundaries tend to merge rapidly in this region

The ratings of the second pilot indicate a considerably smaller satisfactory region but results must be viewed in the light of his relatively short exposure to the simulation ex periment (four flights) Learning effects are evident in the datw in that when configurations from the first flight were repeated on subsequent flights the ratings improved Further more both pilots are in agreement with respect to the center of the optimum region It is likely that substantially better agreement would have been achieved if the second pilot had been afforded additional flight time

Effect of Pilot Technique

In order not to prejudice pilot technique the pilots were briefed at the start of the program regarding the flight control mechanizations to which they would be exposed It was pointed out that the conventional pitch and roll stick would now directly command inertial velocity as opposed to the more usual angular rate or attitude command systems with which they were familiar Furthermore they were instructed

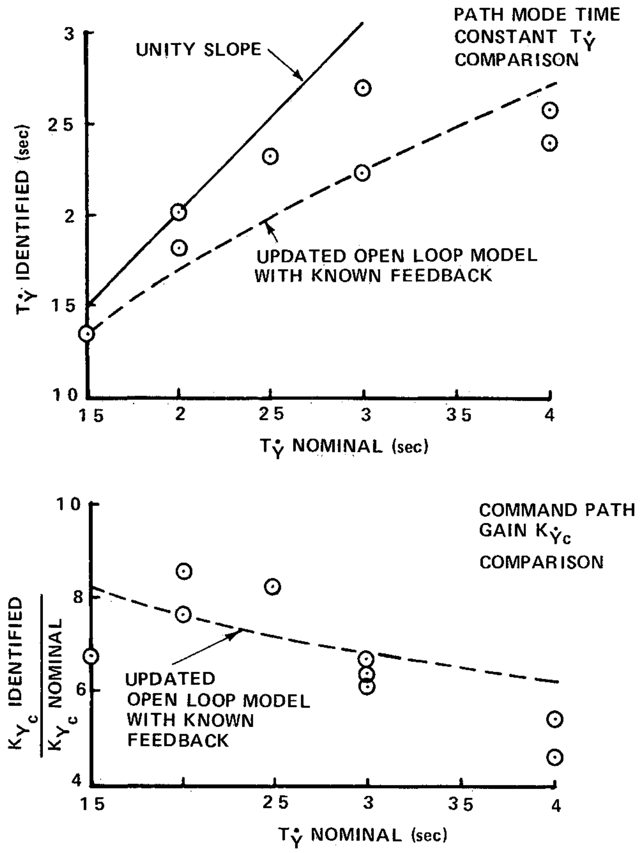

Fig 6 Comparison of lateral directional handling qualities parameters (inner loop attitude $w_n = 2$ 5 rad/s)

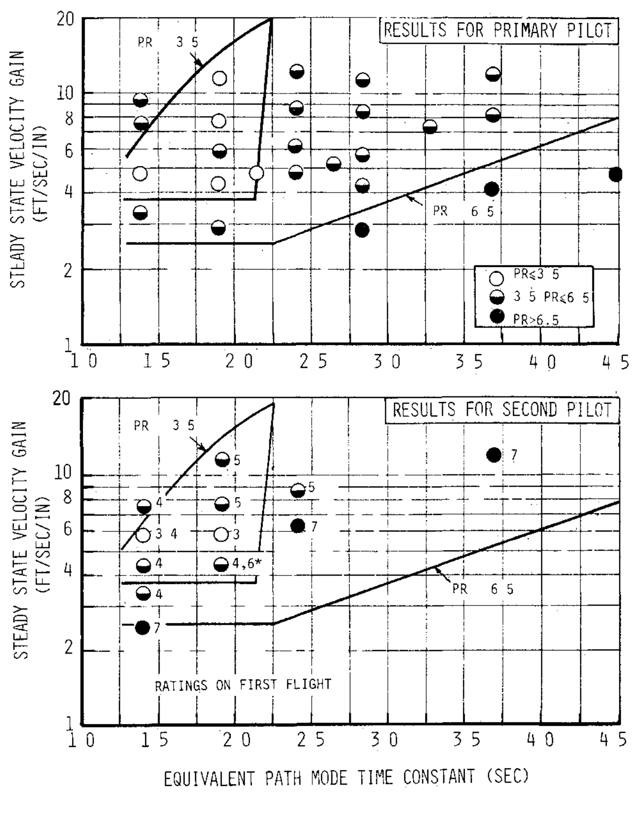

Fig 7 Pilot rating summary for baseline configuration matrix (inner loop attitude $w_n = 2.5$ rad/s)

that pitch and roll attitude were now dependent states manipulated by the flight control system to achieve the commanded inertial velocity In addition although the pilots were told the general range of the parameter variations to be evaluated they were not informed of the specific con figuration for each evaluation The pilots were not directed to any specific control technique but, instead were encouraged to be flexible and to describe in their comments any unusual aspects of the technique employed for each configuration

Although the pilot rating data for the primary and secondary pilots are in general similar it is clear from the recorded pilot commentary that the two pilots were at least for some configurations adapting different control techniques For example the primary pilot in describing the flying qualities and his control technique used phrases like "poor attitude response predictability" "can't find trim attitude for hover" and "forces initially light then heavy up" The latter comment in particular suggests that the pilot was attempting to control the inner loop and maintain a constant pitch attitude by increasing stick deflection to compensate for bleedoff by the flight control system In contrast, the second pilot tended to concentrate on outer loop (position) control His commentary references attitude only when its response was large or abrupt The pilot ratings for a long path mode time constant configuration ($K_{\dot{x}_c}=13\ 0$ ft/s/in $T_{\dot{x}}=3\ 7$ s) tend to substantiate this hypothesis The second pilot's rating was 7 while the primary pilot's rating was 4 5 Configurations with a long path mode time constant (4 to 5 s) have very little velocity feedback and in effect resemble attitude command systems in their response charac teristics It is surmised that with attitude like dynamics, the primary pilot's technique of inner loop control allowed for adequate task performance albeit at the expense of con siderable pilot compensation The second pilot on the other hand, found that adequate performance was not attainable Clearly verification of these conclusions requires more detailed analysis of pilot control activity and may be a fruitful area for application of pilot model identification techniques

Ground Referenced Task Evaluations

A group of evaluations was performed in a ground referenced task for comparison with results from the baseline matrix For flight safety reasons only limited vertical maneuvering was attempted In general the maneuvering with all configurations was more aggressive and distances traveled in translating stopping and reversing tended to be longer than in the HUD task Attitude excursions angular rates velocity and control inputs were all larger as well Pilot confidence in the better TRC configurations was apparent At one point, a translation at 18 ft/s backward along the taxiway was recorded a maneuver that would not be attempted in the basic X 22A with rate SAS

The pilot rating results, although limited in extent indicate a preference for higher control gains than for the HUD task and with satisfactory gains, a tolerance for longer path mode time constant It is clear from these data that the ground reference task would tend to produce results reflecting higher control usage and a more extensive region of satisfactory flying qualities compared to the HUD task

Effects of Turbulence

A potentially significant factor influencing the evaluation data is the effect of ambient turbulence It would have been ideal to fly only under calm atmospheric conditions in order that the disturbance environment could be controlled through the synthetic turbulence; however, the vagaries of the weather and the constraints of the program precluded this arrange ment Therefore for some evaluations the pilot was sub jected to both ambient and artificial turbulence disturbances Some indication of the interaction of ambient and artificial disturbances can be obtained by comparison of pilot ratings for a single configuration evaluated in various combinations of natural and synthetic turbulence A configuration with an inner loop frequency of 2 0 rad/s, $K_{\dot{x}_c}=6\ 0$ ft/s/in; and $T_{\dot{x}}=1\ 4$ s received a pilot rating of 5 when evaluated with little or no ambient turbulence and with the nominal synthetic airwake turbulence Under conditions of moderate to high gusting and turbulence but with zero synthetic turbulence the pilot rating was also 5. With zero synthetic turbulence and calm ambient conditions the rating was 3 Based on these evaluations it was concluded that the synthetic turbulence was equivalent in effect to moderate to high gusting and turbulence Because of the relatively low inner loop frequency this configuration would tend to exhibit higher turbulence sensitivity than either the baseline or high frequency inner loop systems with this path mode time constant

Conclusions

The flight experiment described herein employed the X 22 variable stability V/STOL aircraft to investigate criteria for inertial translational rate control systems The simulation scenario was intended to be representative of small ship visual landings under high sea state and wind over deck conditions Synthetic shipwake turbulence and a HUD generated discrete tracking task were employed as analogs of the actual flight environment and tasks In this context the following con clusions are drawn

1) With the height axis augmented to provide vertical command with altitude hold it is possible to provide satisfactory flying qualities ($PR\leq 3\ 5$) with attitude type translational rate control systems

2) With the same vertical augmentation, pitch and roll attitude command systems will provide acceptable but not satisfactory flying qualities $3\ 5\leq PR\leq 6\ 5$)

3) The HUD discrete tracking task employed in this ex periment provided a repeatable, paced evaluation task suitable for discrimination of hover flying qualities characteristics

4) Time domain identification of X 22A closed loop dynamics from flight measured responses has provided verification of the low order linear dynamical model form used and has helped to update the constant coefficients of that model This has increased confidence in the flight simulation and aided in interpretation of the results

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