Initial Military Flight Tests of the X-22A VSTOL Research Aircraft

WILLIAM B. RHODES*
Naval Air Test Center, Patuxent River, Md.

The X-22A aircraft was evaluated to establish the status of its development and its potential usefulness for conducting flight research using a variable stability system. The more significant aspects and results of the tests are covered in this paper. Ground familiarization and planning conducted prior to the flight tests included the use of a fixed-based simulator programmed with X-22A design data. Test techniques were developed and/or refined during both ground simulator work and actual flight tests as necessary to adapt to the unique characteristics of the aircraft. Quantitative and qualitative evaluation of the stability and control characteristics in terms of their effect on the aircraft to perform its intended mission of flight research have shown that it is acceptable in all respects considered. Unusually high sideforce, with no means of variation, was the most detrimental characteristic of the aircraft noted relative to its potential use as a variable stability flight research vehicle.

Introduction

THE X-22A VSTOL Research Aircraft was designed for the Navy-managed portion of the Tri-Service VSTOL Research Program by the Bell Aerosystems Company, Niagara Falls, N. Y. Its primary design objective was that of general VSTOL flying qualities research using a variable stability system (VSS). Previous descriptions of the X-22A program and related testing include those given in Refs. 1 and 2.

Two X-22A aircraft were originally built. Contractor flight tests began with the number one aircraft in March 1966; the aircraft received irreparable damage in August 1966 during an emergency vertical landing. At this time, 15 flights for a total of 3.1 flight hours had been completed. Contractor flight tests resumed with the number two aircraft in January 1967. An additional 104 flights for a total of 41.1 flight hours were obtained before the Phase I Military Preliminary Evaluation (MPE) commenced in January 1968.

The Phase I MPE, which represented the military's first real look at the X-22A, was conducted during a three-week period by a joint Navy-Army-Air Force test team. The purpose of the MPE was to establish the status of the aircraft's development and determine its potential usefulness for conducting flight research with the variable stability system (VSS), which was incorporated at a later date. The VSS itself was not evaluated. The testing was accomplished by 5 military test pilots (3 Navy, 1 Army, and 1 Air Force) during 14 flights and 10.1 flight hours obtained in 9 actual flying days. The aircraft was evaluated over a limited flight envelope that ranged from −10- to +150-kt indicated airspeed (IAS), 0.5 to 2.0 g, and sea level to 6000-ft pressure altitude.

Description of the X-22A Aircraft

The X-22A, shown in Fig. 1, is a two-place (side-by-side) aircraft that weighs about 14,000 lb with no fuel. It has retractable tricycle landing gear and carries a maximum of 465 gallons of fuel in a single fuselage tank. The aircraft is a unique concept in that thrust, lift, and control moments are all derived from 4 tilting ducted propellers mounted in a dual tandem arrangement. The ducted propellers, which are 7 ft in diameter, are driven through cross shafting by four 1250-hp turboshaft engines such that each engine drives all 4 propellers. With this arrangement, an engine failure does not result in an asymmetric thrust condition. The aircraft is designed such that engine power management can be accomplished through either throttle (power) or collective (propeller blade pitch) control modes. The desired control mode is not selectable in-flight but must be configured on the ground. Only the collective mode was evaluated during the Phase I MPE.

The dual cockpit controls consist of a conventional stick and rudder pedals for attitude control and either a collective stick or throttles, depending on the configuration, for thrust control. A thumb switch on the thrust control is used to control duct rotation angle, which can be varied between 0° and 95° at a maximum (continuous) rate of 5°/sec. Control surface actuation is accomplished through dual, irreversible hydraulic systems. An airspeed sensitive electro-hydraulic feel and trim system provides feel forces in the pitch and roll axes. A mechanical spring system, which can be engaged and disengaged by the pilot, provides feel forces in the yaw axis and is available in addition to the electro-hydraulic feel and trim system in the pitch and roll axes. The natural damping of the aircraft is supplemented with a 3-axis, dual channel stability augmentation system (SAS). The SAS gains are programmed to decrease with increasing airspeed due to the increase in the aircraft's natural damping.

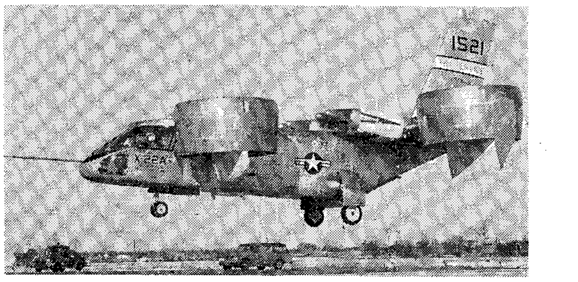

Fig. 1 X-22A aircraft in hovering flight.

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* Supervisory Aerospace Engineer, Flying Qualities and Performance Branch, Flight Test Division. Member AIAA.

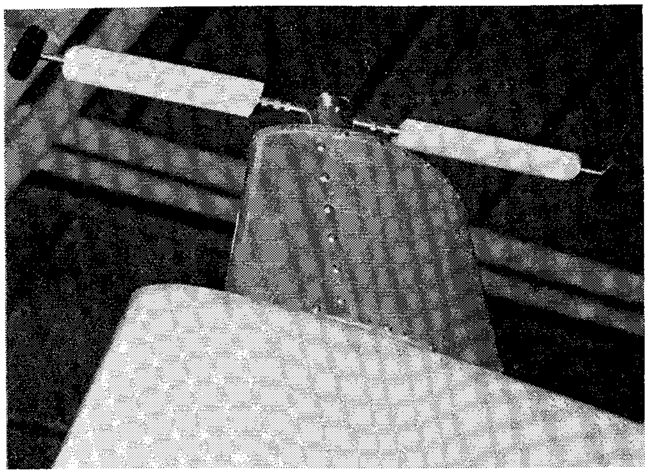

Fig. 2 Low-range airspeed system rotating pressure sensor.

Attitude control is achieved by differential thrust modulation and/or deflection of flow deflectors (elevons) located in all 4 ducts, depending on duct angle (DA). At 90° DA (corresponding to hovering flight), pitch and roll control are accomplished by differential fore-aft and left-right propeller blade angles, respectively; yaw control is effected by differential left-right elevon deflection. At 0° DA (corresponding to conventional flight), the elevons are utilized for pitch and roll control, whereas differential left-right propeller blade angle is used for yaw control. The vertical fin is fixed and has no rudder. At intermediate duct angles (corresponding to powered lift flight), control about each axis is achieved by a combination of elevon deflection and differential thrust modulation, the proportion of which is varied as a function of duct angle to minimize the cross coupling of roll-yaw controls.

The test aircraft was well instrumented for flying qualities and performance testing, although no specific performance tests were conducted during the MPE. The test instrumentation consisted of an airborne magnetic tape system, a 50-channel oscillograph, special cockpit instruments for in-flight data recording, and telemetry for ground station monitor. In addition to a test nose-boom for measuring airspeed, altitude, angle of attack, and angle of sideslip, the aircraft had a low-range airspeed system (LORAS) rotating pressure sensor mounted on top of the vertical fin (see Fig. 2). The LORAS, developed by Cornell Aeronautical Laboratory, was used to measure airspeeds from −10- to +60-kt IAS.

Ground Planning and Familiarization

Prior to the military flight tests of the X-22A, considerable effort was spent in becoming familiar with the aircraft and trying to predict its flight characteristics and possible problem areas. This was accomplished in several ways. As is the usual case with an unfamiliar aircraft, the test pilots and engineers went through a comprehensive ground school covering all the aircraft's systems, including instrumentation. This training gave the test team members an insight into the reasons the aircraft came to look the way it does in addition to a better understanding of its operation. Aircraft design data (stability derivatives) were studied and compared with those of aircraft previously flown by the pilots. This gave the pilots a qualitative feel for what flying qualities to expect relative to what they had experienced in other aircraft.

By far the most useful familiarization technique available to the pilots was a fixed base simulator programmed with X-22A design data. The simulator allowed not only general pilot familiarization with the aircraft's flying qualities but also provided an opportunity to try and modify test techniques to fit the characteristics of the X-22A. In preparing for hover control response flight tests with stability augmentation system (SAS) OFF, which were planned in order to define the characteristics of the basic aircraft, the simulator was used to determine the size of the control input required to get a significant response while, at the same time, allowing a controllable recovery. This was important because the high control power with attendant high control sensitivity designed into all 3 rotational axes meant that relatively large control moments could be generated with relatively small cockpit control deflections. The magnitude of the control sensitivity is best illustrated by the fact that the maximum permissible cockpit control deflection for abrupt movements was 1 in. in any axis, which, the pilots later found, was more than they would care to use in most tasks requiring rapid inputs.

Development and Refinement of Test Techniques

As previously noted, some test techniques were tried and refined during the fixed base simulator work; however, no new techniques were developed during that time. During the flight tests, some existing techniques were refined as necessary to adapt to special stability problems encountered, and some new techniques had to be developed in order to define certain specific aircraft characteristics.

Limited adjustments or refinements were made to the test techniques used for control-fixed dynamics tests and control response and effectiveness tests. During control-fixed dynamics tests in conventional fixed wing aircraft, experience at NATC has shown that it is usually sufficient for the pilot to maintain constant control position after the initial input (with many irreversible control systems, the cockpit control will inherently remain fixed in the trim position after it is released, requiring no deliberate pilot effort). During similar tests in rotary wing aircraft, NATC experience has shown that better results are obtained using a control jig to brace the cockpit control, particularly during tests conducted in the longitudinal and lateral axes. In evaluating longitudinal long period dynamics in the X-22A, initial tests revealed that the control stick would move as a function of pitch attitude when left free (as the nose pitched down, the stick would move forward). With the high longitudinal control sensitivity of the aircraft, the pilots could not adequately maintain constant stick position since even the smallest stick movement would degrade the test results. Therefore, a control jig was braced against the instrument panel and adjusted such that the stick would be held fixed in the trim position after exciting the long-period mode.

During control response and effectiveness tests in conventional fixed wing aircraft, NATC experience has shown that the pilot can normally gauge the size of the step input and hold it sufficiently constant during the ensuing aircraft motion. As with the control fixed dynamics tests, experience has shown that better results are obtained during similar tests in rotary wing aircraft when a control jig is employed. In the X-22A, the pilots could not adequately maintain the stick constant after making a step input due to the high longitudinal and lateral control sensitivity. In addition, because the size of the step inputs being used varied from $\frac{1}{4}$ in. to 1 in., the pilots found it very difficult to gauge differences in their magnitudes. Therefore, a control jig was employed in all longitudinal and lateral control response and effectiveness tests. The refined test technique consisted of the jig being braced against either the instrument panel or the side of the cockpit (depending on axis being tested) and adjusted to allow the desired input size, followed by rapid displacement of the stick from trim to the stop provided by the control jig. In these tests and the stick-fixed dynamics tests, the copilot took care of placing, adjusting, and holding the control jig.

An example of a new test technique that had to be developed during the flight tests was the one associated with the excitation of the Dutch Roll mode in forward flight. In both fixed wing and rotary wing aircraft, this mode can usually be excited by a rudder pulse or doublet or a release from a steady sideslip. However, none of these methods worked with the X-22A, apparently due to the long period of its Dutch Roll oscillation and the relatively slow forward airspeeds involved. After expending considerable effort, it was finally found that good results could be obtained with a slow, deliberate cross control input (rudder pedals one way and lateral stick the other) and release. This method was successful because the input rate tended to match the frequency of the Dutch Roll mode, and because the cross controlling allowed development of moderate sideslip angle while inducing little or no bank angle change.

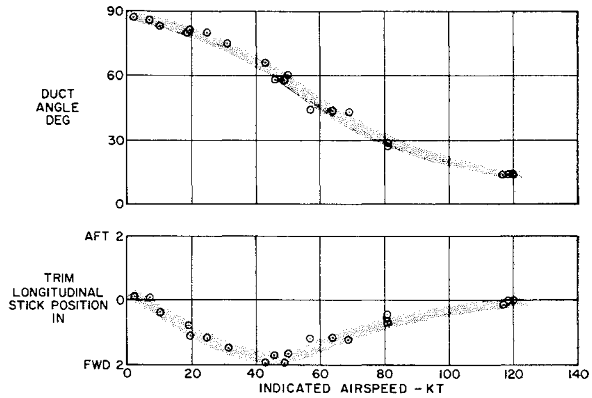

Fig. 3 Relationships derived from level flight trim points with level aircraft pitch attitude in the powered lift flight regime.

Discussion of Flight Test Results

Before discussing the more significant test results, an indication of the scope of the MPE is in order. As previously noted, all tests were conducted within a flight envelope ranging from −10- to +150-kt IAS, 0.5 to 2.0 g, and sea level to 6000-ft pressure altitude. Takeoff and landing characteristics were evaluated at gross weights of up to 15,600 lb during 5 short takeoffs, 6 short landings, 25 vertical takeoffs, and 24 vertical landings. The short takeoffs and landings were accomplished at 15° and 30° DA. Characteristics of transitions between hover flight and forward flight and vice versa were evaluated during 16 conversions and 15 reconversions. (A conversion is a transition in which duct angle is decreased, i.e., airspeed is increased, and a reconversion is one in which duct angle is increased, i.e., airspeed is decreased.) Most conversions/reconversions were accomplished with continuous duct rotation (5°/sec) in constant heading, constant altitude flight; however, some were performed in turning, descending, or climbing flight, and with incremental duct rotation obtained by intermittent actuation of the duct rotation switch. Static and dynamic longitudinal and lateral-directional stability, maneuvering longitudinal stability, and lateral control effectiveness characteristics were evaluated in the aerodynamic (conventional) and powered lift flight regimes, usually in 15° DA increments between 0° and 60° DA. All of the previously mentioned tests were conducted with full SAS (both channels operating) only. Hovering flight characteristics were evaluated during steady hovering tests, longitudinal and lateral translations, and longitudinal, lateral, and directional control response tests at duct angles ranging between 75° and 95°. These tests were performed with full SAS, $\frac{1}{2}$ SAS (one channel operating), and SAS OFF. In conjunction with all of the above tests, evaluations of the cockpit, ground operations, and aircraft systems, including the flight control system, were also obtained.

Takeoff and Landing

Both short and vertical takeoff and landing characteristics were found to be acceptable for the research role of the aircraft. Short takeoffs at 30° DA were accomplished with stick fixed and zero longitudinal trim. The aircraft rotated to a flying attitude at 50-55-kt IAS and became airborne with no deliberate effort on the part of the pilot. Vertical takeoffs were somewhat hindered by the lack of an adequate indication of power output to allow anticipation of liftoff. The MPE testing was conducted with high thrust-to-weight (T/W) ratios that varied between 1.25 and 1.60. If the tests had been conducted at lower T/W ratios, the problems caused by the inadequate indication of power output would have been more serious.

Short landings required a high degree of pilot attention when conducted in crosswinds. Even a 10-kt wind 10°–20° off the nose made the approach feel uncomfortable and uncoordinated to the pilot due to the excessive lateral-directional cross-controlling and bank angle required to maintain a constant ground track. In the same wind, nearly full upwind lateral control and downwind directional control were required after touchdown to counteract the tendency of the aircraft to lean away from the crosswind and weathercock into it simultaneously. During vertical landings, the pilots found it difficult to detect actual main gear touchdown. This was attributed to a strong positive ground effect combined with soft, under-damped main gear struts. This feature caused a typical landing to consist of several bounces on the main gear before the pilot realized the aircraft had touched down.

Powered Life/Transition Flight

An interesting characteristic of an aircraft like the X-22A, where the thrust vector could be varied between 0° and 90° relative to the fuselage reference line, was that a given level flight speed in the powered lift flight regime could be attained with numerous combinations of duct angle, aircraft pitch attitude, and power setting. Of course, at the upper and lower limits of duct angle travel, the aircraft behaved like a helicopter and a conventional airplane, respectively, in that, for a given gross weight in level flight, there was a single pitch attitude-airspeed relationship. In order to keep the number of variables to a minimum and also establish some meaningful relationships with airspeed and/or duct angle, most flying qualities tests in the powered lift flight regime were initiated from level flight trim conditions that included level aircraft pitch attitude. The variation of duct angle and longitudinal stick position with airspeed for these various trim conditions is presented in Fig. 3.

This same variable thrust vector characteristic affected the choice of level aircraft pitch attitude as the common parameter for all conversions/reconversions at duct angles greater than 15°. Between 0° and 15° DA, the aircraft had to be rotated slightly because a nose-up pitch attitude was required to maintain level flight at duct angles below 15°. This rotation, which occurred at the start of a reconversion and at the end of a conversion, was usually accomplished at 120 KIAS where the pitch attitude required for level flight with 0° DA was 10°–15°. As was expected, the relationships of Fig. 3 generally held true for conversions/reconversions, particularly when they were performed at the slower rates associated with incremental duct rotation. Note the rather large trim change due to duct angle/airspeed change in the powered lift flight regime as indicated by the variation of longitudinal stick position with airspeed required to hold a level aircraft pitch attitude. This last

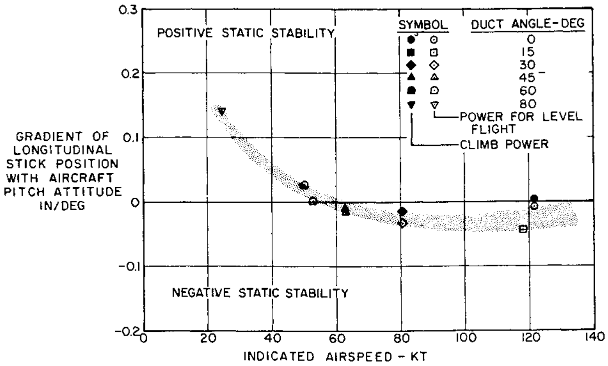

Fig. 4 Static longitudinal stability summary.

statement partially implies another interesting characteristic of an aircraft like the X-22A. This characteristic was related to pilot control functions in the powered lift flight regime and was best seen by analyzing a level flight conversion/reconversion performed at a constant pitch attitude. Stated simply, the pilot controlled airspeed with the duct rotation switch (thrust vector angle), maintained altitude with the collective stick (power or thrust), and maintained pitch attitude with the longitudinal stick (pitching moment). At the upper and lower limits of duct angle travel, the pilot control functions were the same as in a helicopter and a conventional airplane, respectively. Although the longitudinal stick motion required during conversions/reconversions was deemed undesirable, the characteristics were considered satisfactory for the X-22's intended mission of flight research. It is interesting to note that pilot opinion showed a preference for continuous duct rotation conversions/reconversions over the incremental duct rotation type.

Static Longitudinal Stability

Results of static longitudinal stability tests conducted in forward flight are summarized in Fig. 4 in terms of the gradient of longitudinal stick position with aircraft pitch attitude. In both fixed wing and rotary wing aircraft, the degree of static stability is usually expressed in terms of the gradient of stick force or position with airspeed. However, as previously noted, in the X-22A, pitch attitude instead of airspeed was controlled with the stick in the powered lift flight regime. Since most of the data were obtained in this flight regime, the gradient of stick force or position with pitch attitude was considered to be a more valid measure of the static stability. This is not to imply that the results were obtained with airspeed held constant while pitch attitude was varied. On the contrary, both parameters varied about each trim point as is typical in the determination of static longitudinal stability by flight tests. The results indicated the existence of positive stability at duct angles of 60° or more and negative stability at 45° or less within the airspeed range tested. Note that increasing power had a stabilizing effect on the static stability. Although the aircraft was difficult to trim under negative stability conditions, the static longitudinal stability characteristics were acceptable for the flight research mission.

Dynamic Longitudinal Stability

Results of longitudinal long-period dynamic stability tests conducted in forward flight with full SAS are summarized in Fig. 5. Aperiodic divergence was exhibited at duct angles of 45° or less within the airspeed range tested. The most rapid divergence was experienced between 0° and 20° DA (100- to 120-kt IAS), where the time to double amplitude was less than 5 sec. At 60° DA, the aircraft exhibited an oscillatory long period motion that was neutral to divergent in nature. At duct angles above 60°, the oscillatory motion became neutral to convergent. At all duct angles, the long-period motion was less stable with stick left free than with stick held fixed because of the previously discussed tendency of longitudinal stick position to change with aircraft pitch attitude. As with static stability, increasing power had a stabilizing effect on the long-period dynamic stability. The negative long-period dynamic stability characteristics at the lower duct angles (higher airspeeds) were marginally acceptable for the flight research mission.

Maneuvering Longitudinal Stability

Maneuvering longitudinal stability characteristics in forward flight were evaluated during steady pullups, steady turns, sudden pullups, and sudden pushovers. As with conventional airplanes, the gradient of longitudinal stick force or position with normal acceleration provided the best measure of the degree of maneuvering stability at the lower duct angles (higher airspeeds). However, as duct angle increased, the pilot cues to normal acceleration became less perceptible, and pitch rate became the most obvious cue to the pilot. This was particularly true at duct angles above 45° (airspeeds less than 60-kt IAS), where very little normal acceleration could be developed by the aircraft. Therefore, the degree of maneuvering stability was best expressed in terms of the gradient of longitudinal stick force or position with pitch rate at the higher duct angles in the powered lift flight regime. Within the normal acceleration-airspeed envelope tested, all maneuvering gradients were positive, but shallow.

It was found that coordination of steady turns with full SAS required an inordinate amount of directional control pedal, which resulted in excessive pilot effort. At 0° DA and 120-kt IAS, two inches of pedal deflection were required in a 1.3-g turn. At 45° DA and 70-kt IAS, the same pedal deflection was required in a 1.1-g turn. Maximum pedal deflection available was 3.25 in. The excessive directional control pedal needed in turns was partially attributed to the requirement to overcome the opposition of the yaw SAS to turn rate or, more precisely, yaw rate (the component of turn rate about the aircraft yaw axis). The yaw SAS was relatively simple in design and did not contain a wash-out circuit (high pass filter) to prevent feeding back steady state yaw rates. All in all, the maneuvering longitudinal stability characteristics were acceptable for the flight research mission.

Static Lateral-Directional Stability

Results of static lateral-directional stability tests conducted during constant heading steady sideslips in forward flight are summarized in Fig. 6. The static directional stability, as evidenced by the gradient of directional control pedal position with sideslip angle, varied from strongly positive at 0° and 15° DA (approximately 120-kt IAS) to very weakly

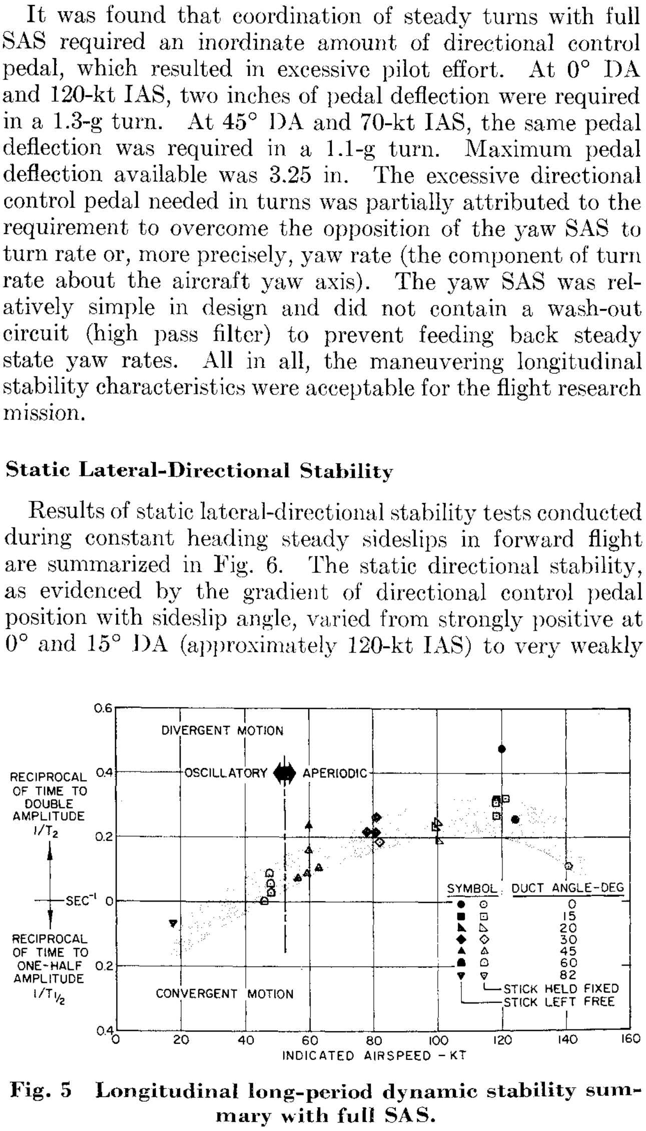

Fig. 5 Longitudinal long-period dynamic stability summary with full SAS.

positive at duct angles of 60° and above (airspeeds below 50-kt IAS). The characteristic most noticeable during these tests was the high sideforce exhibited by the aircraft. The sideforce characteristic, as evidenced by the bank angle required per degree of sideslip angle to maintain constant heading in a sideslip, increased from a value of 0.5° of bank/deg of sideslip at 50-kt IAS (60° DA) to 2.5 at approximately 120-kt IAS (0° and 15° DA). The high sideforce was attributed to the symmetrical lift pattern produced by the ducts, i.e., sideforce produced by sideslip angle was of the same order of magnitude as lift produced by angle of attack. Although not entirely practical, it would have been desirable to provide a means of varying the sideforce during flight research (e.g., this could have been accomplished by locating variable angle flow deflectors in the vertical axis of each duct). Although high sideforce may be inherent in VSTOL aircraft designs, the unusually high sideforce of the X-22A, with no means of variation, was the most detrimental characteristic noted relative to its potential as a flight research vehicle.

Dynamic Lateral-Directional Stability

Dynamic lateral-directional stability characteristics in forward flight with full SAS are summarized in Fig. 7. As previously noted, the best results were obtained by cross-controlled inputs (rudder pedals one direction, lateral stick the other) and release. The damping ratio of the Dutch Roll mode was positive at both extremes of duct angle and airspeed tested, but became neutral to slightly negative at duct angles near 30° (airspeeds near 80-kt IAS). A logical explanation for this unusual variation of damping ratio with airspeed is that 80-kt IAS was the point at which the sum of the artificial damping provided by SAS (which was programmed to decrease with increasing airspeed) and the natural damping of the aircraft (which increased with airspeed) reached its lowest value. The period of the Dutch Roll mode varied from 4 sec at 140-kt IAS (0° DA) to 7 or 8 sec at 40 kt IAS (60° DA). The roll to yaw ratio ranged from 1.0 at 60° DA to about 3.0 at 0° and 15° DA within the airspeed range tested. The dynamic lateral-directional stability characteristics were acceptable for the flight research role.

Lateral Control Effectiveness

Lateral control effectiveness characteristics were evaluated in forward flight with full SAS for lateral stick step inputs ranging from $\frac{1}{4}$ to 1 in. Directional control pedals were held fixed in the trim position during these tests. The aircraft exhibited an interesting roll response that was characterized by a roll to a steady-state bank angle in response to a step lateral stick input; the magnitude of the steady-state bank angle was a function of the amount of stick displacement. The steady-state bank angles and sideslip angles obtained for 1 in. lateral stick deflections are summarized in Fig. 8. For a given lateral input, the steady-state bank angle increased

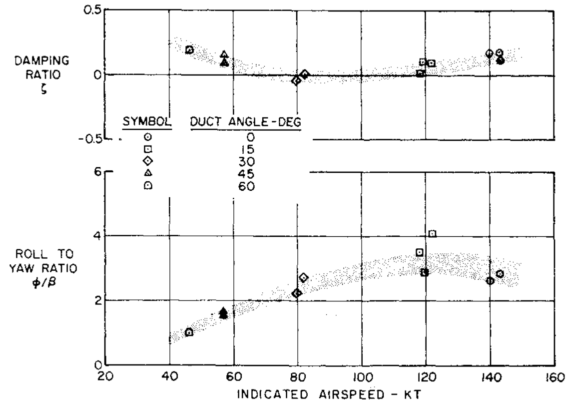

Fig. 7 Dynamic lateral-directional stability summary.

with airspeed while steady-state sideslip angle decreased within the airspeed range tested. In both fixed wing and rotary wing aircraft, lateral stick normally commands roll rate instead of bank angle. The fact that it commanded bank angle in the X-22A was because the rolling moment due to sideslip counteracted the rolling moment due to lateral control. The previously mentioned opposition of the yaw SAS to turn rate was partially responsible for the balance of moments because it tended to increase adverse sideslip. The lateral control effectiveness characteristics of the X-22A were acceptable for its intended mission of flight research.

Hovering Flight Characteristics

Steady hovering out of ground effect (OGE) was evaluated at all three levels of SAS operation (full SAS, $\frac{1}{2}$ SAS, and SAS OFF). With full SAS, the X-22A was an excellent hovering platform. In fact, momentary hands off hover was accomplished on several occasions. Although pilot effort increased as SAS contribution was decreased, hovering OGE presented no particular problem with SAS OFF. Hovering in ground effect (IGE) with full SAS was characterized by large and rapid random disturbances about the pitch and roll axes. With $\frac{1}{2}$ SAS, the angular disturbances were significantly more pronounced, so much so that the pilots decided against trying a $\frac{1}{2}$ SAS vertical landing originally planned.

Longitudinal, lateral, and diagonal translations were evaluated OGE at airspeeds up to 20-kt IAS with full SAS. Longitudinal translations could be accomplished at both forward and rearward airspeeds by either rotating the ducts while maintaining a level aircraft pitch attitude or by changing the pitch attitude at a constant duct angle. As pilots gained experience in the aircraft, their preference changed from initially favoring control of translation speed with pitch attitude to a preference for speed control with duct angle. In lateral translations, the bank attitude required for a given lateral airspeed was somewhat excessive, which is

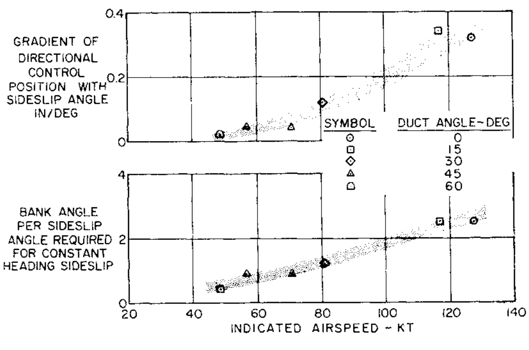

Fig. 6 Static lateral-directional stability summary.

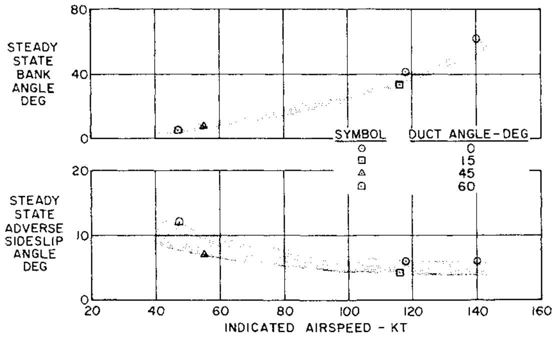

Fig. 8 Lateral control effectiveness summary for 1-in. lateral stick inputs with full SAS and directional control pedals held fixed.

further evidence of the aircraft's high-sideforce characteristic.

Longitudinal, lateral, and directional control response characteristics were evaluated OGE for all 3 SAS conditions (full SAS, $\frac{1}{2}$ SAS and SAS OFF) with cockpit control step inputs varying from $\frac{1}{2}$ to 1 in. The results indicated the existence of very high control power and sensitivity about all 3 rotational axes. (Control power is the total control moment available about a given axis for full control displacement while control sensitivity is the control moment generated per unit of control displacement. It is convenient to measure both in terms of angular acceleration, since it is equal to control moment divided by moment of inertia about a given axis.) Longitudinally, about 1 rad/sec^2 of angular acceleration was produced for a 1-in. step input. Assuming constant control sensitivity and knowing that the maximum longitudinal stick displacement was a little over 4 in. in either direction, extrapolation to full stick displacement indicated that the longitudinal control power was slightly better than 4 rad/sec^2 (the design value was 3.25 rad/sec^2). Laterally, the same control sensitivity (1 rad/sec^2 per inch of control displacement) resulted. The maximum lateral stick displacement was nearly 4.5 in. in either direction, indicating that the lateral control power approached 4.5 rad/sec^2 (the design value was 3.40 rad/sec^2). Directionally, the control sensitivity was about 0.2 rad/sec^2 per inch of pedal deflection. Since the maximum pedal deflection was 3.25 in. in either direction, extrapolation to full pedal displacement indicated that the directional control power was about 0.7 rad/sec^2 (the design value was 0.7 rad/sec^2). All hovering flight characteristics of the X-22A were acceptable for the flight research mission.

Conclusions

The planning, conduct, and results of the initial military flight tests of the X-22A VSTOL Research Aircraft have been described. Evaluation of the flying qualities results in terms of their effect on the potential of the aircraft as a variable stability flight research vehicle has shown that it is acceptable in all respects considered. The most detrimental characteristic noted relative to its potential as a flight research vehicle was the presence of unusually high sideforces, with no means of variation.

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