Control Characteristics of a Buoyant Quad-Rotor Research Aircraft

B.L. Nagabhushan,* D.W. Lichty,† and N.P. Tomlinson‡
Goodyear Aerospace Corporation, Akron, Ohio

Control characteristics of a buoyant quad-rotor research aircraft are predicted by considering such a vehicle configuration based on preliminary design. Concepts for controlling the vehicle with or without its external sling load are evaluated by using a flight dynamics simulation of the aircraft. Results are presented which show the vehicle response to control inputs, wind disturbances, and systems failure while hovering over a point on the ground. Typical control power and trim characteristics of the vehicle are also discussed.

Nomenclature

a_x = longitudinal acceleration, ft/s^2
a_y = lateral acceleration, ft/s^2
a_z = vertical acceleration, ft/s^2
a_ϕ = roll acceleration, deg/s^2
a_θ = pitch acceleration, deg/s^2
a_ψ = yaw acceleration, deg/s^2
$I_{xx,yy,zz}$ = moments of inertia about reference body axes x, y, and z whose origin is at aircraft center of mass, slug-ft^2
V_w = wind velocity, knots
x = longitudinal ground position of the aircraft, ft
y = lateral ground position of the aircraft, ft
Δ = prefix used to denote step change
ψ = aircraft heading angle, deg
ψ_w = wind azimuth angle, deg

Introduction

RECENTLY there has been a growing need for transporting large and heavy cargo that cannot be handled by conventional aircraft. Consequently, a new generation of vertical takeoff and landing vehicle concepts, particularly rotorcraft with unprecedented lifting capability, is being developed[1] to airlift payloads externally on a sling. A particular concept that belongs to this class of aircraft and offers the possibility of greatly improved low-speed control and stationkeeping characteristics far beyond that of historical lighter-than-air vehicles is the quad-rotor hybrid airship. Basically this concept consists of a nonrigid, buoyant, nonrotating hull that is rigidly attached to a structural frame supporting the propulsion components. The advantage of such an arrangement is that the empty weight of the vehicle is supported by the force due to buoyancy while the propulsive forces are entirely available for lifting the payload and controlling the vehicle. This vehicle is intended for various civil and military applications where externally suspended payloads are transported over short distances such as in off-loading container ships, logging, or moving construction equipment.

Presented as Paper 81-1838 at the AIAA Guidance and Control Conference, Albuquerque, N. Mex., Aug. 18-21, 1981; submitted Sept. 30, 1981; revision received Dec. 9, 1982.

*Engineer Specialist, Flying Qualities Group, Defense Systems Division. Member AIAA.

†Engineer Specialist, Systems Technology Group, Defense Systems Division.

‡Engineer Specialist, Computer Applications Group, Defense Systems Division.

Since the quad-rotor hybrid airship is a novel vehicle concept, it is proposed to design and build a small-scale flight research vehicle called the buoyant quad-rotor research aircraft (BQRRA) for ground and flight tests to prove the feasibility of the concept and to investigate its flying qualities. In this article the control characteristics of such a flight research vehicle are discussed.

For the chosen vehicle configuration, concepts for controlling the vehicle in all of its flight modes are developed and evaluated by using a flight dynamics simulation of the BQRRA. The response of the vehicle to control inputs and atmospheric disturbances are predicted in terms of typical flight parameters. Subsequently the control power characteristics of the vehicle with or without a payload are determined by considering the proposed control concepts. Several failure modes of the BQRRA are simulated to determine their consequence on the flight of the vehicle and its safety.

Vehicle Description

Based on preliminary design study,[2] a vehicle configuration was selected that would use existing hardware and thereby minimize the cost of the aircraft. The proposed configuration consists of four modified Hughes OH-6A helicopters mounted on the outriggers of an interconnecting structure that is attached to a conventional airship envelope with an empennage as shown in Fig. 1. In this vehicle, the main rotor torques are countered by built-in tilt of the main rotor shafts of the helicopters, augmented by vehicle control moments when necessary.

As an option, it is proposed to hinge the helicopters in roll only to obtain additional thrust vectoring capability. Further, four auxiliary propellers, which are, in fact, tail rotors from the Bell Helicopter Sea Cobra (AH-1T), are incorporated so that two of them augment the cruise mode of the vehicle while

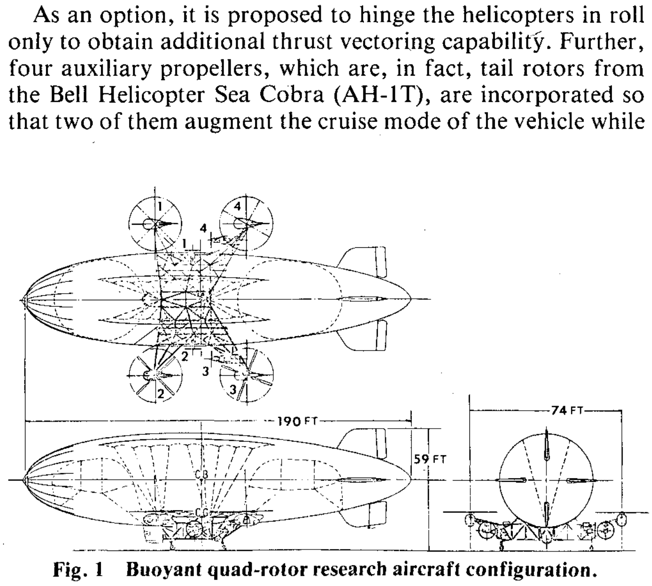

Fig. 1 Buoyant quad-rotor research aircraft configuration.

Table 1 Physical properties of the BQRRA

Item	Estimate/design value
Overall length	192 ft
Maximum width	100 ft
Maximum height	60 ft
Envelope volume (stretched)	205,270 ft^3
Helicopter rotor diameter	26 ft
Auxiliary propeller diameter	8 ft
Static lift (at sea level density altitude)	13,035 lb
Maximum gross weight (at sea level density altitude)	23,435 lb
I_{xx}	363,961 slug-ft^2
I_{yy}	82,276 slug-ft^2
I_{zz}	1,030,234 slug-ft^2

the other two augment the lateral control of the vehicle (see Fig. 1). These tail rotors are selected instead of conventional propellers so that 100% reverse thrusting capability can be provided to enhance vehicle control. In order to determine the effect of vehicle buoyancy on its flight characteristics, it is designed to be closer to neutral buoyancy while it is empty and is 10,000 lb heavy with maximum payload at sea level. The estimated physical properties of the vehicle are given in Table 1. Note that heaviness of the aircraft corresponds to gross weight in excess of static lift for that flight condition. Some of these estimates have been subsequently revised, but they are not likely to significantly alter the inherent control characteristics described here.

Vehicle Control Concepts

The forces and moments required to control the aircraft with or without its sling load are obtained by changing the lifting rotor thrust vectors as well as thrust generated by the auxiliary propellers, in unision or differentially. The empennage control surfaces provide additional control moments in pitch and yaw during forward flight. For the present vehicle configuration there are 22 control variables that may be systematically combined for controlling the aircraft in all of its flight modes, including precision hover. They are collective pitch and longitudinal and lateral cyclic pitch angles of the helicopter main rotors, collective pitch of the auxiliary propellers, attitude of the helicopters on the roll hinges, and the elevator and rudder deflection angles of the empennage. It is observed that for each control axis the corresponding force or moment can be produced in a number of ways. For instance, a yawing moment can be produced by using 1) differential collective pitch of the longitudinal auxiliary propellers, 2) differential longitudinal or lateral cyclic pitch of the helicopter main rotors, and 3) rudder deflection when effective. A particular choice of these options, in a sequence or in unison, depends on the flight and operating conditions. Clearly, in the absence of a payload the aircraft is light enough that very little rotor thrust is required to lift and sustain it in flight. Consequently, in this case rotor thrust vectoring for controlling the vehicle is ineffective. Further, if the aircraft flight speed is below 20 knots then its elevator and rudder controls have diminished effectiveness. However, the auxiliary propellers can be conveniently used in this mode to control the vehicle. These constraints suggest that in a case where the aircraft is hovering over a point on the ground the lifting rotors should perhaps be used for vehicle control when it is heavy and the auxiliary propellers when it is light. In the ferry mode when the vehicle is not carrying any payload it may be controlled like a conventional airship, particularly during short takeoff and landing while the lifting rotors may be used for control augmentation only.

Taking all of the above factors into consideration, nominal control concepts have been developed. Basically they could be incorporated into a control system similar to that of a conventional helicopter. A collective stick, which varies the

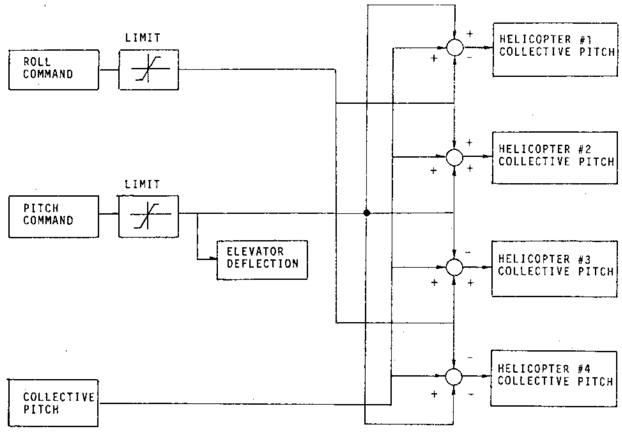

Fig. 2 Concept for roll, pitch, and vertical control.

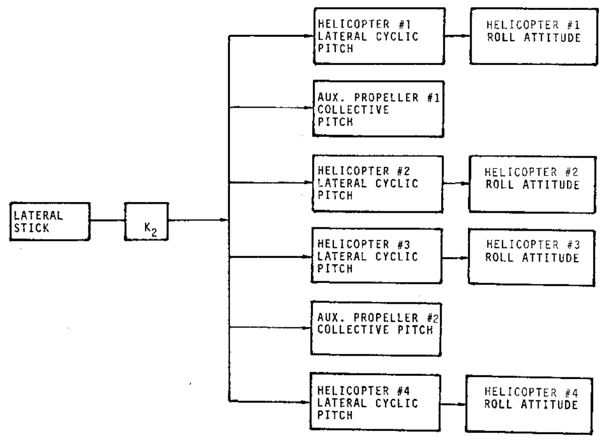

Fig. 3 Concept for lateral control.

collective pitch of all of the four helicopter main rotors in unison (Fig. 2), is proposed for vertical control. A pitch command input which varies the collective pitch of the fore [Nos. 1 and 2] and aft [Nos. 3 and 4] helicopter main rotors differentially is provided as a separate control. Similarly, a roll command input which varies the collective pitch of the right [Nos. 1 and 4] and left [Nos. 2 and 3] helicopter main rotors differentially is also provided. The pitch command input is assumed to simultaneously deflect the elevator surfaces at all airspeeds. The differential collective pitch angles of the helicopter main rotors resulting from pitch or roll command inputs are assumed to be limited to ±10% over the corresponding prevailing values of the collective pitch of individual rotors.

A longitudinal/lateral stick similar to the cyclic stick of a helicopter is envisioned for longitudinal and lateral control. The lateral stick input would vary the lateral cyclic pitch of all of the four helicopter main rotors in unision (Fig. 3). It would also vary the collective pitch of the lateral auxiliary propellers in unision with the lateral cyclic pitch of the helicopter main rotors. Subsequently, in implementing this concept in the flight dynamics simulation, it is assumed that each helicopter on its roll hinge would provide ±12 deg tilt of its thrust vector about its roll axis. Typically, each main rotor would respond to its lateral cyclic pitch input by tilting in roll while the corresponding fuselage attitude would remain unchanged. However, on a steady-state basis it is intended to wash out this lateral cyclic pitch input by implicitly commanding the helicopter roll attitude such that the same thrust vector orientation is maintained while the residual lateral cyclic pitch does not exceed its limit of ±4 deg. Similarly, the longitudinal stick input would vary the longitudinal cyclic pitch of all of the four helicopter main rotors in unison (Fig. 4). It would also vary the collective pitch of both the

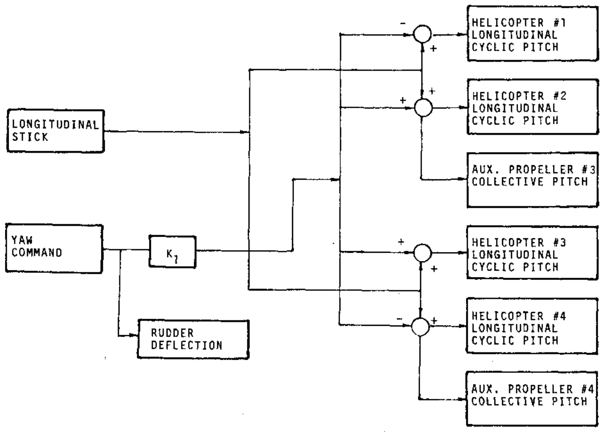

Fig. 4 Concept for longitudinal and yaw control.

longitudinal auxiliary propellers simultaneously. In this case, since the helicopters are fully restrained in pitch axis, they do not rotate with respect to the outrigger structure. Consequently, it is assumed that each main rotor would respond to its longitudinal cyclic pitch by tilting its thrust vector in pitch within the limits of ± 4 deg on a steady-state basis.

Directional control of the vehicle is provided by yaw command (Fig. 4) which varies the longitudinal cyclic pitch input to left and right helicopter main rotors differentially. This control input simultaneously varies the collective pitch of the longitudinal auxiliar propellers differentially and also deflects the rudder surface at all airspeeds.

An important aspect of the vehicle control configuration was selecting the location of its helicopter and auxiliary propeller components. Typically, the line of action of the resultant of rotor thrust vectors determine the vehicle attitude and its trimmability during hover. In the absence of any readily applicable design criteria, it is proposed to locate the helicopters symmetrically with respect to the vehicle longitudinal and lateral planes at the center of buoyancy, such that the net rotor thrust results in minimal or no pitching moment. The vertical location of these components were selected on the basis of operational and ground handling considerations.

It is observed that in the proposed aircraft configuration the location of the helicopters and auxiliary propellers give rise to inherent control cróss couplings which are significant. For instance, the location of the longitudinal auxiliary propellers are such that a forward thrust also results in a nose-up pitching moment. Consequently, one has to simultaneously apply a nose-down pitching moment by using the differential collective pitch on the rotors or, when effective, by deflecting the elevator in order to maintain a constant pitch attitude. Similarly lateral thrust from the auxiliary propellers tend to roll the vehicle adversely such that the net lateral control force decreases. Again, in this case one has to simultaneously apply a roll moment to counter this effect in order to maintain zero or any other favorable roll attitude. Indeed these cross-couping effects tend to decrease the available control power during some operational flight conditions, as will be demonstrated subsequently.

The control concepts described above were used in a six degree-of-freedom (DOF) flight dynamics simulation of the aircraft to evaluate its control characteristics in several flight modes which are described below.

Response to Control Inputs

A flight dynamics simulation, which has been previously developed[3] for hybrid heavy lift airships, was adapted to simulate the present aircraft configuration. Complete motion (6 DOF) of the vehicle was simulated by using estimated[2] values of the overall physical and aerodynamic properties of the aircraft. Unlike the airplane or helicopter, the handling qualities criteria for hybrid airships are nebulous. Consequently, the control characteristics of the vehicle have been evaluated by considering specific tasks such as ability to maneuver from hover, ability to accelerate into a head wind or cross wind, and ability to hover over a point on the ground in a variable, shifting wind.

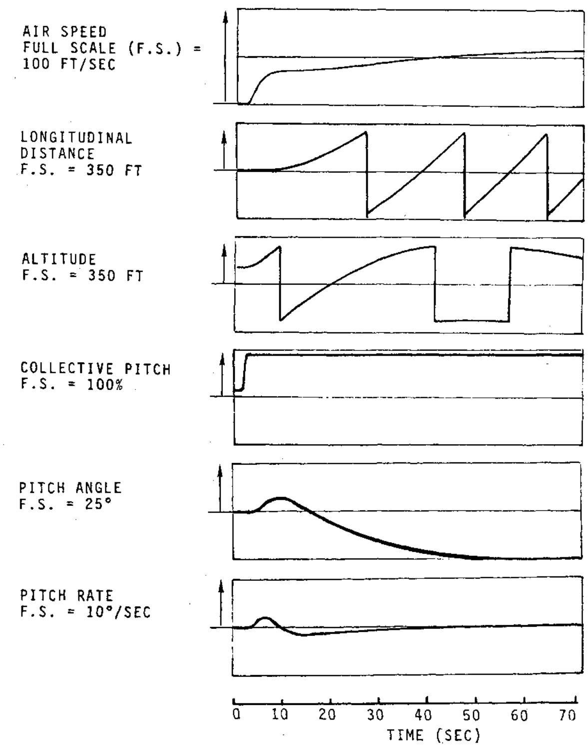

Fig. 5 Vehicle response to collective pitch input (gross weight = 14,292 lb).

Significant insight into hover maneuverability, dynamics, and control cross-coupling effects were obtained by determining vehicle response to control actuations in an open-loop fashion. Initially, the vehicle by itself (gross weight = 14,292 lb) was trimmed in hover and then subjected to individual unit step inputs to each of its controls. The vehicle reponse was recorded on strip charts. A similar procedure was repeated for a case in which the aircraft had an internal payload (gross weight = 22,292 lb).

Vehicle reponse to a unit step input to its collective pitch is illustrated in Fig. 5. Here the vertical arrows indicate the full-scale, positive value of each recorded variable. This response illustrates the presence of inherent aerodynamic and dynamic cross couplings in vertical and pitching motions, which lead to longitudinal motion as well. This can perhaps be explained as follows. Initially, as the aircraft climbs vertically it experiences an aerodynamic nose-up pitching moment which results in a pitch velocity. In conjunction with the prevailing vertical velocity this produces forward motion of the vehicle which decreases its angle of attack. Consequently, the vehicle experiences a nose-down pitching moment which results in a pitch-down attitude and a forward descent.

Response of the vehicle to a unit step input to its longitudinal stick (Fig. 6) consisted of forward motion in which the aircraft developed a nose-up pitch attitude of 20 deg, resulting in climb as well. The pitch-up tendency of the vehicle can be understood by noting that the line of action of the longitudinal control force has a large vertical offset from the aerodynamic center of the airship envelope. Consequently, the control application results in a nose-up pitch attitude that gives rise to aerodynamic pitching moment. This tends to further increase the pitch attitude of the aircraft until a steady state is reached in which the metacentric moment balances the aerodynamic pitching moment. The vehicle with its payload responded similarly but experienced a stable pitch oscillation until a steady nose-up pitch attitude of 14 deg was reached.

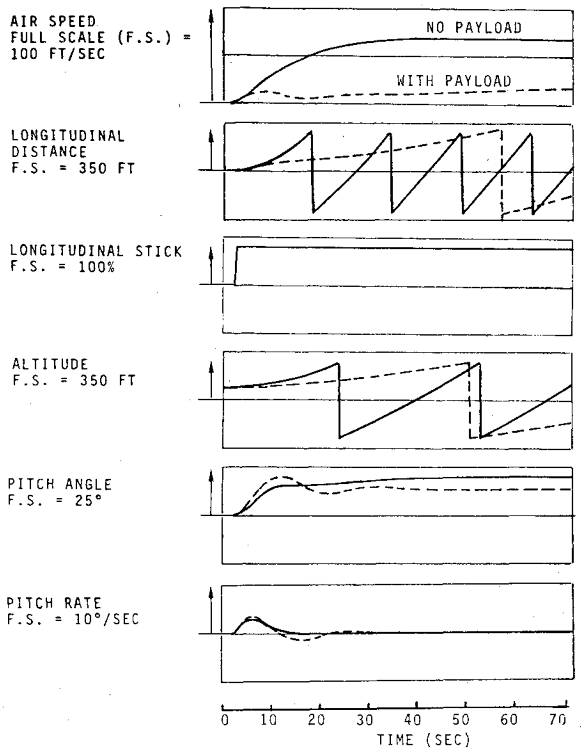

Fig. 6 Vehicle response to longitudinal stick input.

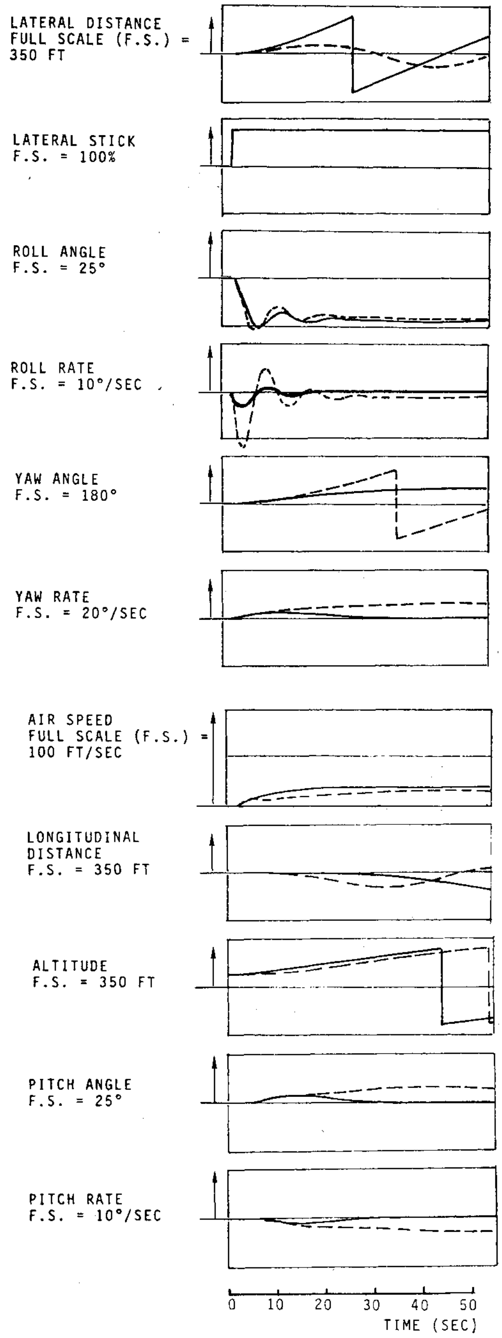

Fig. 7 Vehicle response to lateral stick input.

The aircraft responded to a unit step input to its lateral stick (Fig. 7) by moving laterally while developing a roll attitude, which results from the control cross coupling similar to that of longitudinal-pitch axes coupling described above. The line of action of the lateral control force is such that a control force to the right results in a roll tilt toward the left, away from it. As shown by the response in this case, the vehicle moved in the starboard direction while it reached a negative roll attitude (tilt to the port side) of 20 deg. The application of this control also results in a stable, roll oscillation in which the envelope metacentric moment generates the roll stiffness and the helicopters generate the damping moment. Meanwhile the aircraft was found to drift backward and upward as well as change its heading at a slow rate. The same control input to the vehicle with payload resulted in a similar response coupled with pitching motion as well. The rolling oscillation in this case had larger peak amplitude and the vehicle developed a steady yaw rate of 6 deg/s.

A unit step input to roll command tends to oscillate the aircraft in roll while it drifts laterally (Fig. 8). The rolling oscillation is stable and subsequently resulted in a steady roll attitude of 14 deg. In the case of a vehicle with payload the loss in altitude observed results from a decrease in net vertical component of rotor thrust forces following a roll command input. This can be alleviated to some extent by varying the authority of the roll command to generate collective pitch on the helicopter rotors, such that the resultant thrust sustaining the aircraft weight is unaltered. However, it should be possible to maintain constant altitude in this case by appropriately modifying the control loop or by reducing payload.

The vehicle responded to a unit step input to its pitch command (Fig. 9) by oscillating in pitch with a peak rate of 5 deg/s and amplitude of 22 deg while it drifted backward and downward. As a consequence the aircraft began to drift laterally and experienced both rolling and heading excursions simultaneously. These cross-coupling effects are perhaps due to dynamic coupling associated with aircraft pitch velocity at large pitch attitude angles. In the case of the vehicle with payload the response in the pitch plane was similar but the pitch oscillation was weaker with a peak rate of 2 deg/s and amplitude of 6 deg. No excursions in the lateral and directional motions were observed here.

A unit step input to yaw command (Fig. 10) resulted in the aircraft developing a steady yaw rate of 20 deg/s. A loss of altitude was also observed in the case of the vehicle with payload. As described earlier, this is due to the reduction of net vertical component of rotor thrust forces following this control input.

Trim and Control Power Characteristics

One of the principal operational requirements of this flight research aircraft is that it should be able to hover in windy environment. Consequently, the operational flight envelope within which the vehicle, with or without its payload, can be trimmed in hover were determined by considering various combinations of wind magnitude and its direction (Fig. 11). These results have been obtained by considering vehicle motion in the horizontal plane alone (3 DOF), as well as completely unrestrained motion of the aircraft (6 DOF).

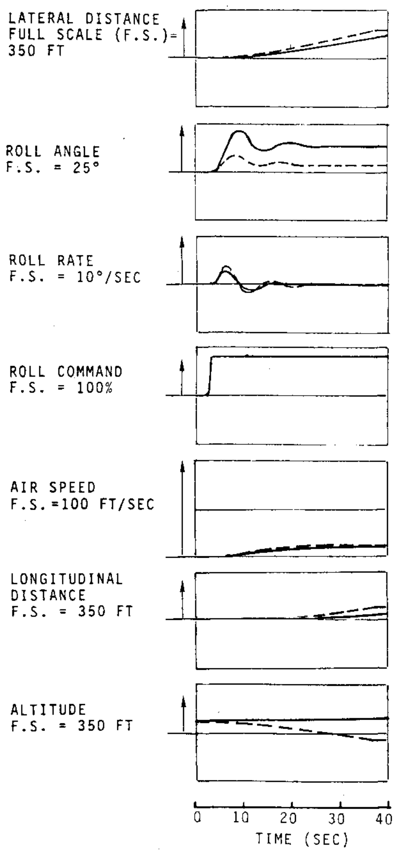

Fig. 8 Vehicle response to roll command input.

Based on the trim flight envelope predicted by the 3 DOF model, it is observed that the presence of a payload permits the rotor thrust vectoring to provide additional control in longitudinal, lateral, and yaw axes. Consequently, the vehicle is able to maintain trim at more severe wind conditions than when it is without a payload or lighter. However, comparing these flight envelopes with those predicted by the 6 DOF model, it is observed that both give similar results for the case of aircraft without payload. But the 6 DOF model predicts a significantly smaller flight envelope for the aircraft with payload. This indicates possible inadequacy of the control forces and moments available to decouple the vehicle motion in the horizontal plane from its corresponding lateral and directional motions, while operating the aircraft with a heavy payload.

The control power available to maneuver the vehicle while it is hovering with or without its payload has been estimated in terms of maximum possible acceleration in each of its control axes (Table 2). Both lateral and directional control power are relatively large by design since it is intended to provide greater maneuverability in these axes. Typically, the hovering aircraft would be required to maintain ground position and head into the wind following a wind shift during operations near ground. The vertical control power is maximum for the vehicle alone while it diminishes as the aircraft becomes heavy with increasing payload. The angular accelerations of this aircraft are quite small compared to those of a conventional helicopter. But they are significantly large when compared to those of a conventional airship. It is important to note that these control power estimates are functions of the control concept itself as well as the vehicle heaviness and prevailing trim condition. The actual control power available in a given operational flight condition would depend also upon the various control cross-coupling effects brought into play.

In order to examine the control power available while operating in windy environment, the vehicle was initially trimmed while it was hovering into a head wind. Sub-

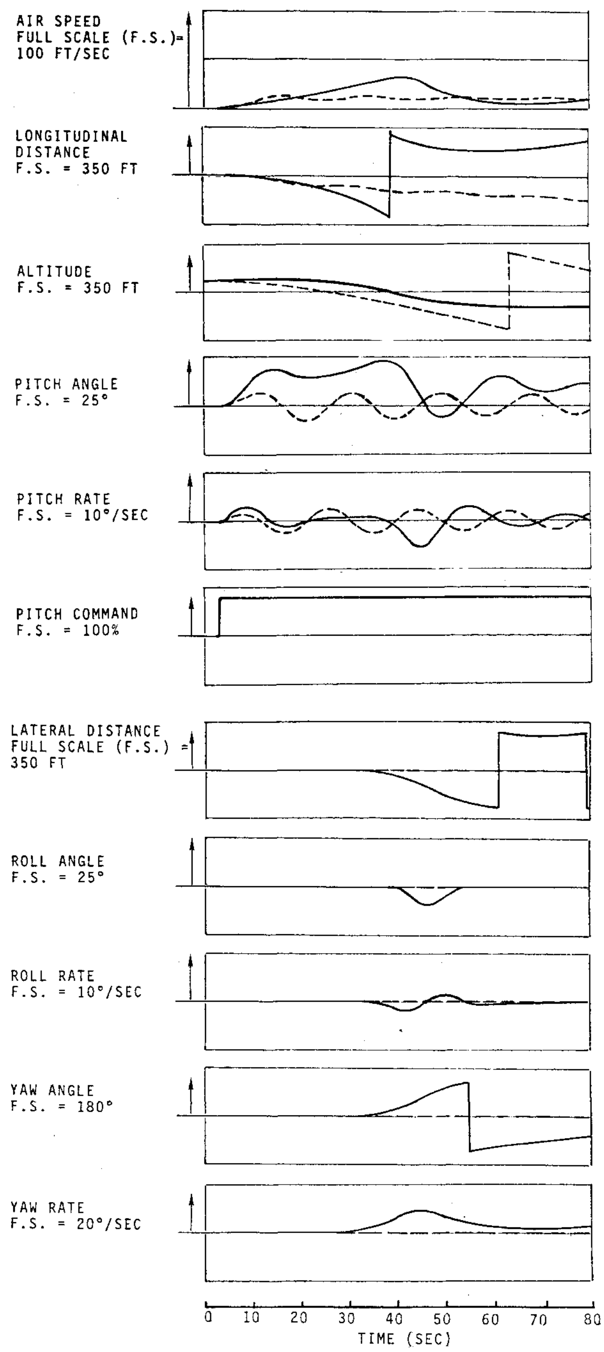

Fig. 9 Vehicle response to pitch command input.

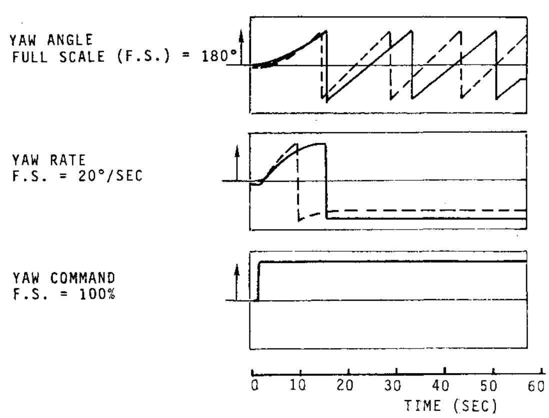

Fig. 10 Vehicle response to yaw command input.

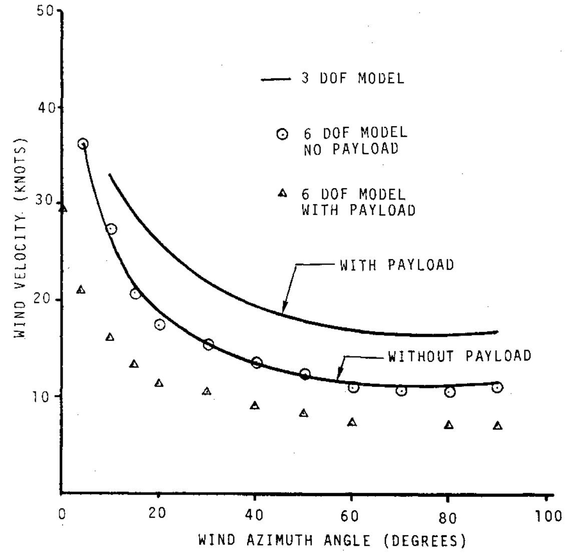

Fig. 11 Flight envelope of aircraft trim in hover.

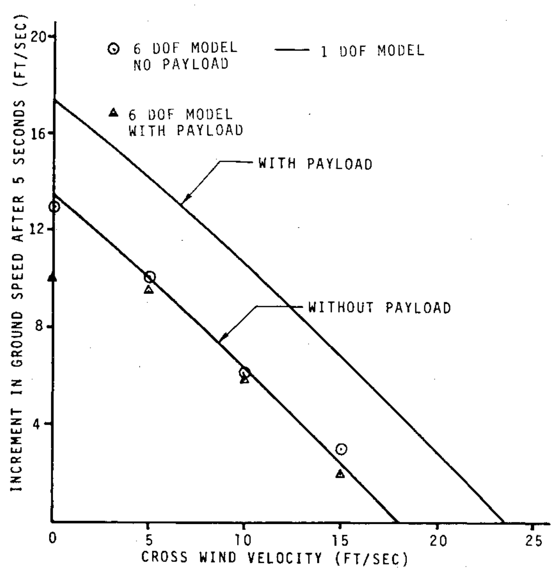

Fig. 13 Vehicle lateral control power in a cross wind.

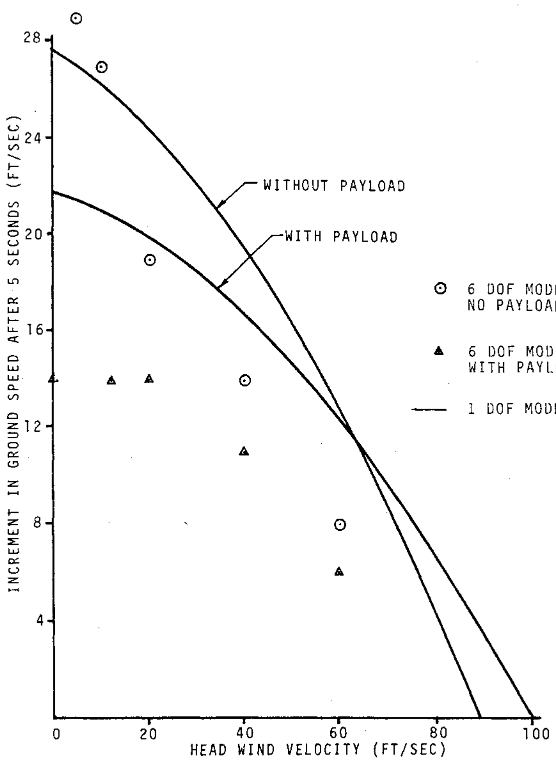

Fig. 12 Vehicle longitudinal control power in a head wind.

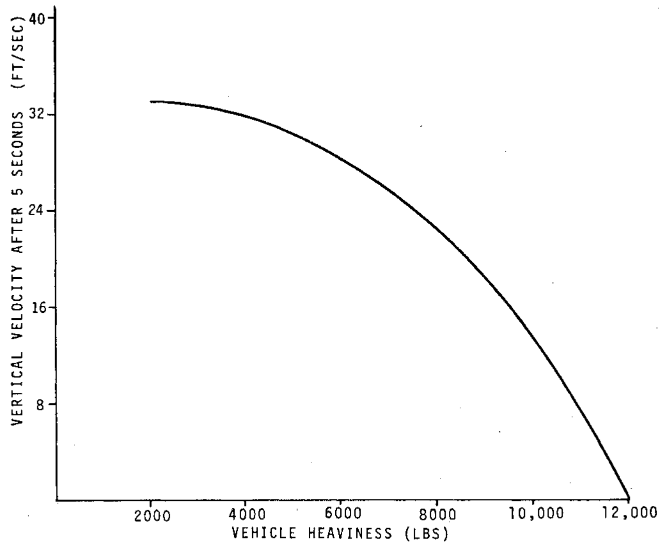

Fig. 14 Effect of heaviness on vertical control power.

Table 2 Vehicle control power in hover

Acceleration components	Without payload (gross weight = 14,292 lb)	With payload (gross weight = 22,292 lb)
a_x, ft/s^2	5.30	4.09
a_y, ft/s^2	3.28	4.18
a_z, ft/s^2	11.65	0
a_ϕ, deg/s^2	1.10	3.31
a_θ, deg/s^2	0.18	0.55
a_ψ, deg/s^2	2.89	3.68

sequently, a unit step input to its longitudinal stick was applied and the resulting increment in ground speed in 5 s, a measure of longitudinal control power, was determined by considering isolated longitudinal motion (1 DOF) as well as unrestrained motion (6 DOF) of the aircraft. In the latter case, autopilots were used to maintain altitude and heading, and zero pitch attitude. It is found from the 1 DOF model (Fig. 12) that the vehicle without its payload tends to accelerate better in lower headwinds while the vehicle with its payload accelerates better in higher head winds. Typically, the aicraft with its payload has greater rotor thrust available and hence more longitudinal control force than when it is without a payload. Apparently for the vehicle with payload this increase in longitudinal control force exceeds the drag increase at higher speeds and leads to greater longitudinal acceleration than possible with the vehicle alone. Comparing these results with those obtained from the 6 DOF model, it is observed that the latter model predicts smaller control power for both the cases of aircraft with and without its payload. This difference may be construed as a loss of control power resulting from the decoupling of longitudinal motion from the others. Similarly, response of the aircraft to a unit step input to its lateral stick while the vehicle was hovering in a cross wind was also examined (Fig. 13). Here again an isolated lateral motion model (1 DOF) was used and the corresponding results were compared with those from 6 DOF models of the aircraft. It is observed that both these models give similar results for the vehicle without payload, thereby indicating the complete decoupling of the lateral motion in this case from the others. However, the 6 DOF model predicted a smaller flight en-

Table 3 Closed-loop response to wind disturbances

		Vehicle gross weight = 18,018 lb					
			Maximum error			Control input	
Hover trim	Step input	x, ft	y, ft	ψ, deg	LONSTK,%	LATSTK,%	YAWC,%
$V_w = 15$ knots $\psi_w = 30$ deg	$\Delta\psi_w = 10$ deg	0	−4.9	−1.35	−0.5	67	7.5
$V_w = 10$ knots $\Psi_w = 30$ deg	$\Delta\psi_w = 10$ deg	0	−2.1	−0.9	−1	27.5	3
$V_w = 5$ knots $\Psi_w = 30$ deg	$\Delta V_w = 10$ knots	−0.2	−1.4	−0.9	1	20	5
$V_w = 0$	$\Delta V_w = 10$ knots $\Delta\Psi_w = 90$ deg	1.6	−3.3	4.5	−15	45	−15

velope for the aircraft with payload, which indicates inadequacy of control forces and moments available to decouple in this case. It is important to note that albeit the isolated lateral control power available is larger for the vehicle with a payload rather than by itself (Table 2) the same may not be realized in operation as illustrated here.

Response of the hovering aircraft to a unit step input to its collective pitch was obtained to illustrate the vertical maneuverability of the vehicle. The resulting vertical velocity after 5 s following the control application are shown in Fig. 14 for various heaviness conditions. It was found that the vertical control power is strongly dependent on the vehicle heaviness.

Response to Wind Disturbances

The capability of this research aircraft and hence, the quad-rotor hybrid airship concept itself, to perform under adverse weather conditions depends on the adequacy of its control for satisfactory operation. In the absence of specific performance requirements it is convenient to consider various operational flight conditions involving step changes in wind magnitude and shifting wind direction to illustrate typical performance of the aircraft. Consequently, the 6 DOF model of the vehicle was trimmed in hover in calm air or a prevailing wind. Simple autopilots consisting of error and rate feedback were used to form a closed-loop control logic that would come into play when the aircraft was disturbed from its trim. Both pitch and roll attitudes were commanded to zero while maintaining heading. An altitude autopilot was used to maintain a reference height of 300 ft. The ground position was maintained by longitudinal and lateral autopilots. Table 3 shows the corresponding closed-loop response of the aircraft to several wind disturbances. The relatively short excursions of the vehicle observed here could perhaps by attributed to its large time constant or sluggishness and the assumption of instantaneous tilting of the thrust vectors and availability of vehicle motion cues. In comparing these results with similar data for a heavier vehicle, it was found that the presence of a heavier payload had no favorable effect on vehicle response, albeit larger control forces and moments were produced in such a case. This can be explained by noting the control power lost in decoupling the inherent cross couplings of longitudinal-pitch and lateral-roll motions of the aircraft which tend to offset the greater control forces and moments of the heavier vehicle.

Response to Systems Failure

Since this is a unique, experimental aircraft, its failure modes need a thorough investigation both on ground and in flight. The hybrid nature of the vehicle lends itself to a combination of emergency modes of operation associated with system failures in a conventional airship and helicopter. When the vehicle is hovering without any payload and low on fuel, it is light enough to maintain altitude and land safely in the event of power failure in one of the helicopters. However, when the aircraft is hovering with a heavy payload and carrying full fuel and has power failure, it may descend quickly and may become uncontrollable. In order to determine the consequences several failure modes of the vehicle were simulated and corresponding open-loop responses of the aircraft were recorded, which are described below.

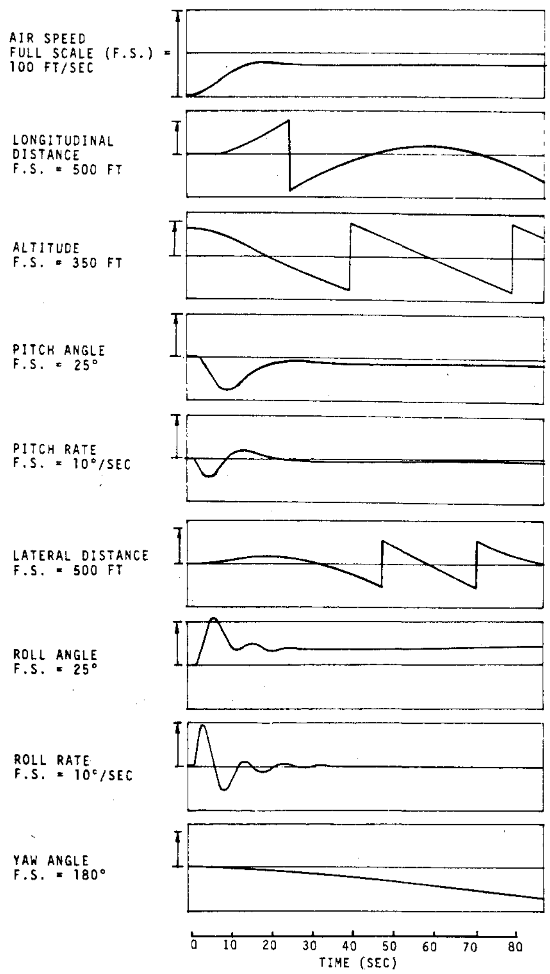

Fig. 15 Vehicle response to power failure in right front helicopter (gross weight = 22,292 lb).

In a case where the aircraft was hovering with a gross weight of 22,292 lb at an altitude of 300 ft in calm air, the power failure in its right front helicopter was simulated. The correspoding response (Fig. 15) indicates that subsequent to the failure the aircraft oscillated in roll with a peak rate of 10 deg/s and eventually developed a right roll attitude of 10 deg. Meanwhile it also developed a steady nose-down pitch at-

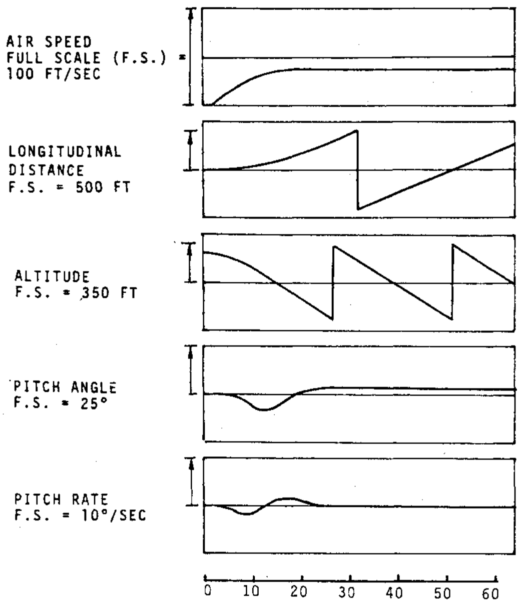

Fig. 16 Vehicle response to right front and left rear helicopter power off (gross weight = 22,292 lb).

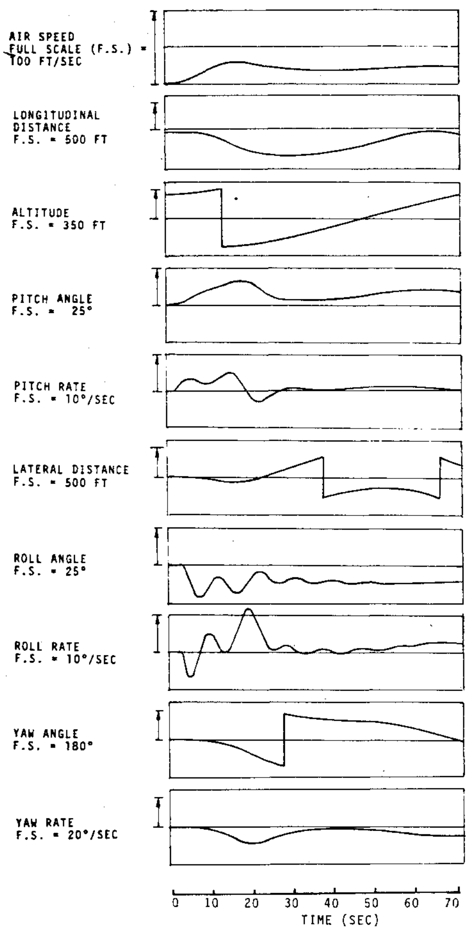

Fig. 17 Vehicle response to control system failure in right front helicopter (gross weight = 22,292 lb).

titude of 3 deg and a descent rate of 1200 ft/min. The aircraft was found to be drifting laterally as well as changing its heading continuously. When a similar failure occurred on the rear left helicopter, the vehicle responded similarly in roll but developed a negative or left roll attitude of 12 deg and a nose-up pitch attitude of 6 deg. It also developed a descent rate of

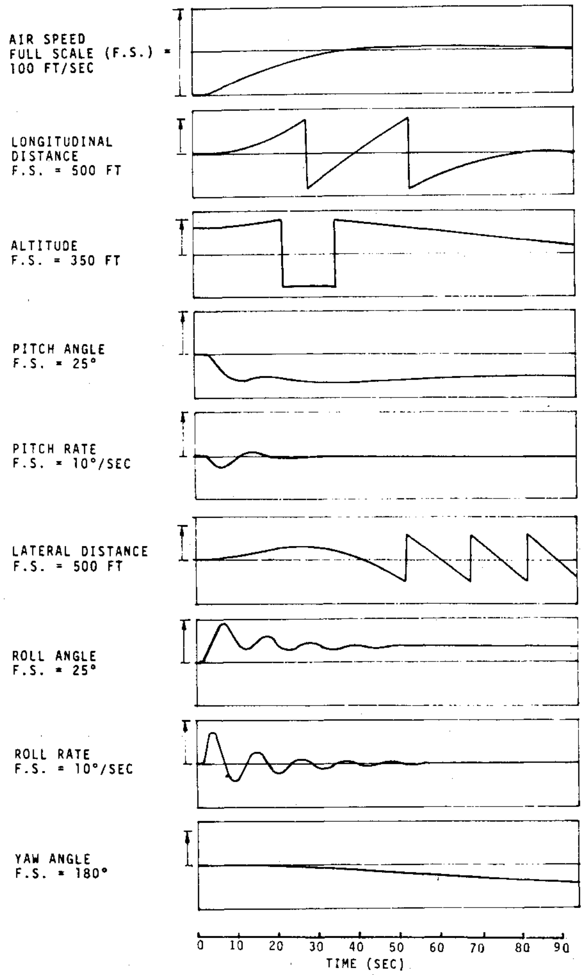

Fig. 18 Vehicle response to control system failure in left rear helicopter (gross weight = 22,292 lb).

1200 ft/min and drifted laterally and changed its heading as before. However, when the power of the rear left helicopter was automatically shut off simultaneously with the right front helicopter power failure, the vehicle developed a steady nose-up pitch attitude of 3 deg, while descending at a rate of 1750 ft/min (Fig. 16).

A failure in the control system of the hovering aircraft that resulted in maximum thrust on the right front helicopter, while others remained at their trim values, was also simulated. The consequent response of the vehicle indicates (Fig. 17) strong rolling oscillation of the vehicle with peak rate of 12 deg/s, which eventually resulted in a steady left roll attitude of 10 deg. The aircraft began to climb and drift backward and subsequently developed a nose-up pitch attitude of 10 deg. It also drifted laterally and developed a yaw rate of 5 deg/s. A similar control system failure in the left rear helicopter caused the vehicle to develop a nose-down pitch attitude of 13 deg, a right roll attitude of 9 deg, and a descent rate of 330 ft/min (Fig. 18). The aircraft also drifted laterally and developed a steady yaw rate as in the previous case. Maximum safety can be ensured in these failure modes by jettisoning the payload as part of the recovery procedure.

Conclusion

The control characteristics of the buoyant quad-rotor research aircraft, as determined from its flight dynamics analysis and simulation, indicate the inherent capabilities of the vehicle to investigate its operational characteristics, to develop handling qualities criteria, and enable refinement of the vehicle concept itself. The economic constraint introduced in the preliminary design to consider existing hardware components does limit the exploration of full potential of the vehicle concept. For instance, one could use dedicated rotor systems which have significant negative or reverse thrust capability instead of existing helicopters. This would permit the vehicle without a heavy payload to hover and develop larger control forces and moments by using equal and opposite rotor thrust vectors such that resulting vertical force is zero or a desired value. However, the proposed research aircraft with all its limitations could still provide a unique tool for further development.

Acknowledgment

This work was done under contract to NASA Ames Research Center, Moffett Field, California, Contract No. NAS2-10777. It was monitored by Peter D. Talbot, Program Manager for Hybrid Airship Technology. The flight dynamics simulation used in this study was developed for Goodyear Aerospace Independent Research and Developement Program.

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