ICAS-82-1.6.4

PERFORMANCE CHARACTERISTICS OF A BUOYANT QUAD-ROTOR RESEARCH AIRCRAFT

B. L. Nagabhushan*, P. P. Jacobs**, C. E. Belknap†
Goodyear Aerospace Corporation, Akron, Ohio

and

D. A. Euler††
Hughes Helicopters, Culver City, California

Abstract

Performance characteristics of a Buoyant Quad-Rotor Research Aircraft, which represents a hybrid airship concept for heavy lift application, are described. Celing altitude and endurance for hovering at typical power levels, including partial power failure, are predicted. Climb performance at various altitude and gross weight conditions have been examined. Forward flight performance of this vehicle is illustrated in terms of typical performance parameters such as maximum speed, maximum range and endurance, over the full range of its payload capability. Optimum payload weights have been identified which result in maximum range at sea level density altitude and constant endurance at various altitudes, both during hover and cruise flights.

Introduction

Recent studies have indicated military and civil needs for vertical lift of payloads exceeding the payload capacity of existing and anticipated heavy-lift helicopters. Military needs include lifting heavy battlefield equipment and off-loading container ships over undeveloped shores. Examples of civil applications include logging and transport and emplacement of heavy equipment for large construction projects. Use of multiple rotor vehicles appears to be a cost effective method of lifting these heavy payloads since such vehicles can use existing helicopter propulsion and rotor systems. Hybrid aircraft employing rotor systems and a buoyant lifting hull appear to be particularly attractive, especially for extremely heavy lift, due to the relatively lower cost of buoyant lift.

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 station-keeping 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.

Since the quad-rotor airship is a novel vehicle concept, it is proposed to design and build a small-scale flight research vehicle called the Buoyant Quad-Rotor Aircraft [BQRA] for ground and flight test to prove the feasibility of the concept and investigate its flying qualities. In this paper, the performance characteristics of such a flight research vehicle are discussed.

In configuring such an experimental aircraft, greater emphasis was placed on providing adequate control power to permit flight test evaluation over a flight envelope, approaching that of the full-scale vehicle itself. Consequently, the potential performance benefits of the full-scale aircraft may not be obvious from the corresponding results for this flight research vehicle. However, the characteristic trends observed here are typical and should be inherent to the vehicle concept itself. It is suggested that one should view the results presented here, in this regard.

Performance characteristics of the BQRA in hover, climb and forward flight have been estimated in terms of those of its airship and helicopter components. The corresponding methods used for such a synthesis and the results obtained are described below following a description of the vehicle itself.

Vehicle Configuration

Based on a 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 Figure 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

* Flying Qualities Engineer, Defense Systems Division

**Aerodynamicist, Defense Systems Division

† Weight Analyst, Defense Systems Division

††Member of Technical Staff, Commercial Engineering Division

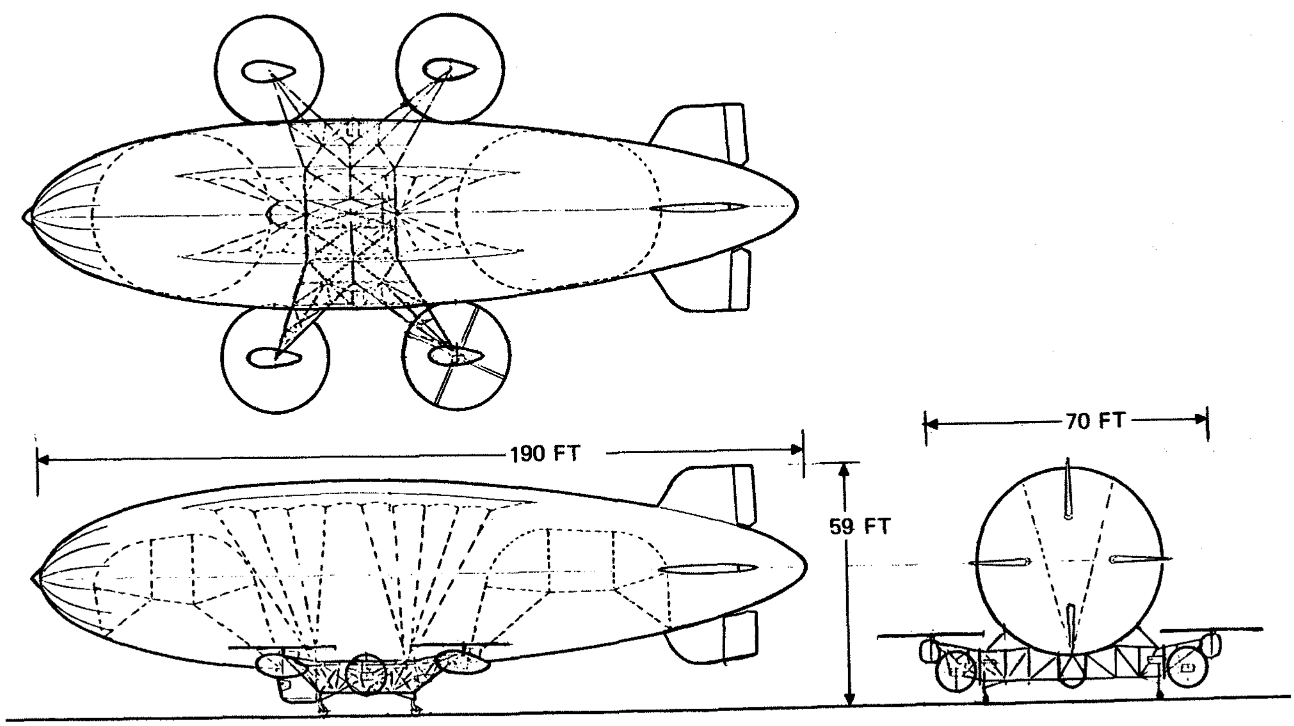

Figure 1 - BQRA Configuration

of the vehicle while the other two augment the lateral control of the vehicle [see Figure 1]. These tail rotors were selected instead of conventional propellers so that 100 percent reverse thrusting capability could 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 pounds heavy with maximum payload at sea level.

Physical properties of the aircraft configuration and its components are given in Table I. The aerodynamic properties of the vehicle were estimated[2] and correlated with existing wind tunnel and flight test data on the airship and helicopter components. Only in cases where such data were not available, theoretical estimates have been used.

As a first approximation, the normal force and pitching moment characteristics of the airship component were assumed to be equal to its side force and yawing moment characteristics. The axial force on the vehicle due to envelope and support frame was computed by combining the estimated zero-lift drag of the airship envelope with the corresponding component of the support frame drag. Similar estimates were made for the vehicle normal force or drag resulting in a vertical climb. The corresponding axial and normal force coefficients for the overall vehicle are 0.1 and 1.2, respectively. These coefficients are based on $\forall^{2/3}$ where $\forall$ is the stretched volume of the airship envelope. The horizontal parasite drag area of each of the helicopter units was assumed to be 6 ft^2. In the case of vertical climb, the download due to each helicopter module was assumed to be 1.5% of the corresponding main rotor thrust.

Hover Performance

A unique characteristic of this aircraft concept is its ability to produce lift by several means; static or buoyant lift from helium, powered lift from rotors and aerodynamic lift on airship envelope at moderate flight speeds. During hover both the buoyant lift and powered lift are used to sustain the vehicle and its payload. The magnitude of these lift forces are affected by the atmospheric conditions in which the aircraft operates. The powered lift component also varies with the power level used and ground proximity. Consequently, it is essential to determine these effects for predicting vehicle hover performance, such as, ceiling altitude, endurance and power requirements for a specified flight condition. The corresponding methods used and results obtained are described below.

Hover Ceiling. To predict ceiling altitudes for Hover Out of Ground Effect [HOGE] as a function of ambient temperature, the corresponding rotor thrust available from the four OH-6A helicopters [Figure 2] as well as the static lift available from the airship envelope [Figure 3] was calculated. The total thrust available from the quad-helicopter system was obtained by summing the individual helicopter thrust data for various altitudes and ambient temperature conditions. Typically, these data were generated by using analytical performance estimation methods developed by the helicopter manufacturer. They were correlated with available flight test data[3], after appropriate correction for absence of tail rotors in the present configuration. It should be noted that the operating thrust levels of these helicopters in this configuration are, for lower

Table I - Physical Properties of the BQRA

ITEM	ESTIMATE/DESIGN VALUE
Airship:	
Envelope Volume [Stretched]	205,270 Ft^3
Overall Length	192.2 Ft
Maximum Diameter of Envelope	45.9 Ft
Fineness Ratio	4.14
Propulsion:	
Hughes OH-6A Helicopter (4):	
Maximum continuous lift at 2600 Lb each	10,400 Lb
Power Plant - Allison T63-A-5A or - Allison T63-A-700	317 Shp [derated to 278 Shp; 236 Shp Max continuous]
Number of blades on main rotor [articulated]	4
Rotor Diameter	26.3 Ft
Auxiliary Power Units:	
(4) Allison T250-C18 [Modified to turboprop configuration or equivalent]	300 Shp installed
Propellers (4):	
Tail Rotor from Bell Sea Cobra [AH-1T]	
Number of Blades	2
Diameter [for engine mod. with 1480 rpm output]	9.7 Ft
Chord	1.0 Ft
Static thrust at sea level, maximum continuous power [each]	1,400 Lb
Tip Speed [at 1480 RPM]	750 Ft/Sec

Weight and Lift Data For Entire Vehicle	At Sea Level Density Altitude	At 5000 Ft Density Altitude
Empty weight + fuel, oil and crew	18,018 Lb	18,018 Lb
Fuel	3,200 Lb	3,200 Lb
Static Lift	13,035 Lb	11,223 Lb
Net Buoyancy	- 4,983 Lb	- 6,795 Lb
Helicopter Lift Available	10,400 Lb	10,040 Lb
Payload Capability [Helicopter lift plus net buoyancy]	5,417 Lb	3,245 Lb
Gross Weight	23,435 Lb	21,263 Lb

heaviness of the aircraft, significantly below their nominal value in a free flight. In such cases, analytical estimates have been used on the basis of best judgement.

The static lift of the airship envelope was calculated by the following method illustrated here with an example:

Static Lift at 2000 Ft Pressure Altitude

= [Stretched Volume of Envelope] [Unit Lift of Helium] [Ballonet Inflation at 2000 Ft]

where,

Ballonet Inflation at 2000 Ft

$$= \frac{\text{Air Density at 2000 Ft}}{\text{Air Density at Sea Level}}$$

consequently,

Static Lift at 2000 Ft = [205270 Ft^3] [0.0635 Lb/Ft^3] [0.943] = 12292 Lb.

This lift was corrected for temperature effect based on absolute temperature ratio with standard sea level conditions. For instance, static lift at 2000 ft at 80 degrees F is given by

$$(12{,}292)\left(\frac{59 + 459.4}{80 + 459.4}\right) = 11{,}813 \text{ Lb}$$

Similar calculations were made for other temperature and altitude conditions, shown in Figure 3. These results were combined with the corresponding powered lift data for the four helicopters [Figure 2] to determine the maximum permissible gross weight of the vehicle during hover at a given altitude and ambient temperature. Figure 4 shows hover ceiling in a case where the helicopters were assumed to be operating at maximum continuous power, while Figure 5 shows similar data corresponding to operation with one engine out, or, in this vehicle, one helicopter unit out, condition. In the latter case the aircraft could be trimmed such that residual pitch and roll attitudes are small.

Note that in hovering at maximum take-off power, the gross weights shown in Figure 5 correspond to the maximum gross lift that can be generated. Consequently, no allowance has been made here for thrust required for maneuver from hover at these

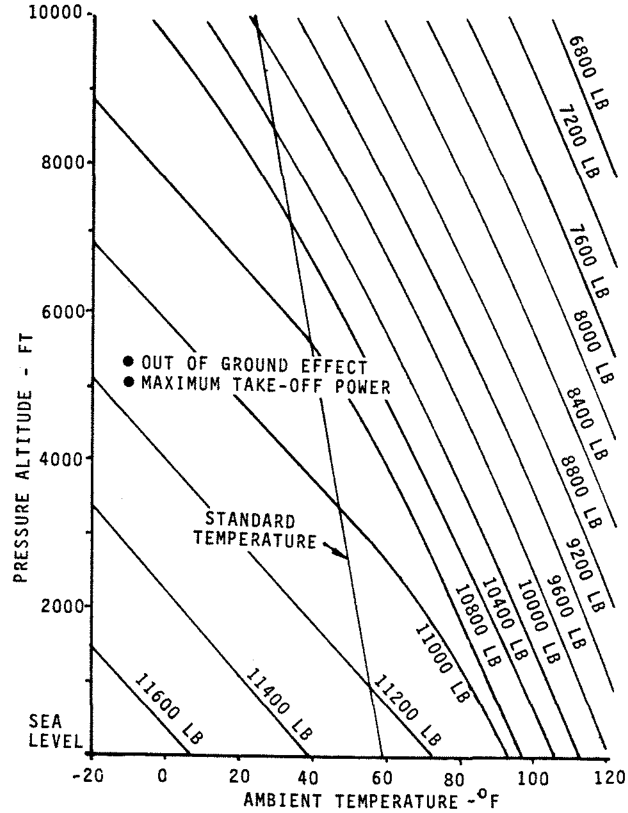

Figure 2 - Effect of Altitude and Ambient Temperature on Total Helicopter Thrust

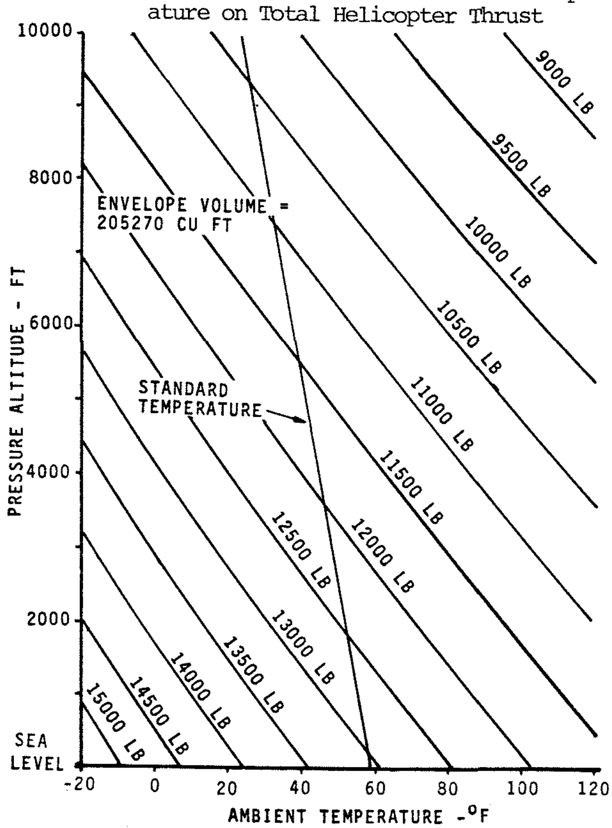

Figure 3 - Altitude and Ambient Temperature Effects on Airship Envelope Static Lift

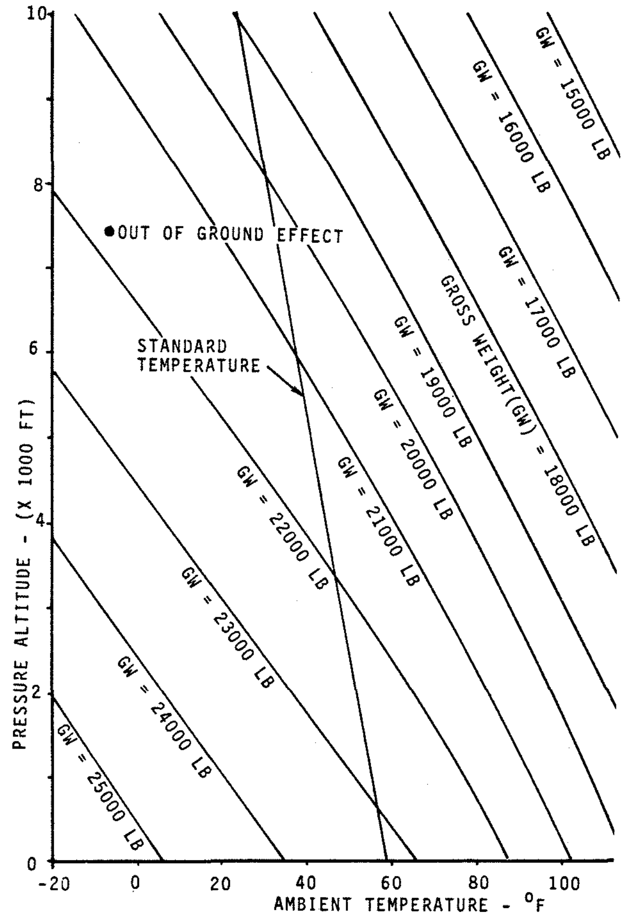

Figure 4 - Hover Ceiling at Maximum Continuous Power vs Ambient Temperature

gross weight conditions.

The power requirements of the aircraft for hovering [Figure 6] have been determined by combining the corresponding power requirements of the individual helicopters. Typically, these were evaluated for a specified atmospheric condition and thrust level or fraction of aircraft gross weight sustained by each helicopter. The maximum permissible gross weights for both the maximum takeoff and maximum continuous power limits in a standard day operation are shown in Figure 6. Note that the power available to the helicopter is transmission-limited. The T63 engine on the OH-6A is derated from 317 shaft horsepower [shp] to 236 shp [maximum continuous] when installed. Consequently, the engine is capable of delivering 236 shp at 5000-feet density altitude.

Hover ceiling in ground effect is perhaps not significant to this aircraft, since the helicopter rotors are located nearly out of ground effect.

Hover Endurance. The eight engines in the proposed vehicle configuration could consume significant amounts of fuel in a relatively short time in comparison to other aircraft. Consequently, hover endurance is important for operational consideration of this research vehicle.

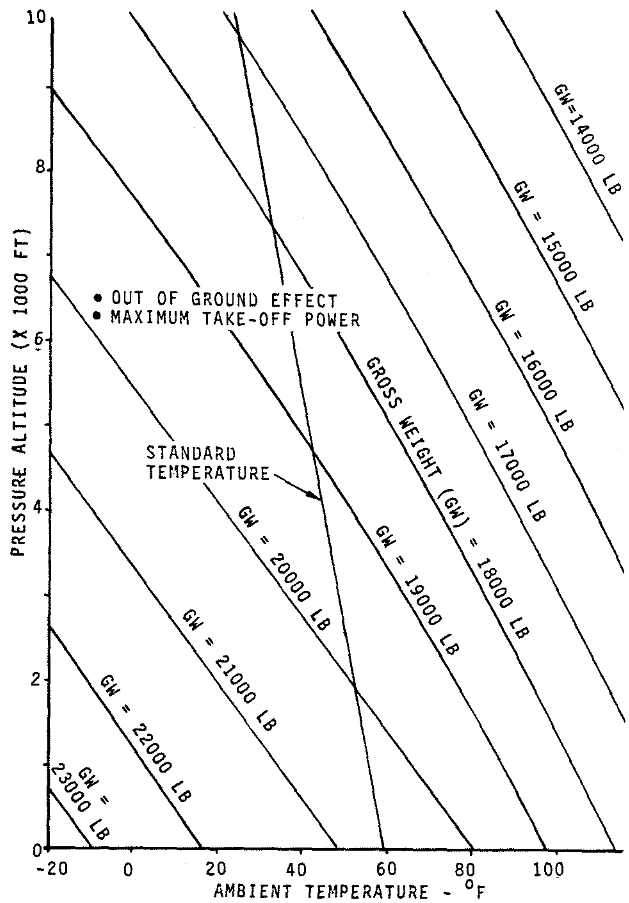

Figure 5 - Hover Ceiling with One Helicopter Power Out vs Ambient Temperature

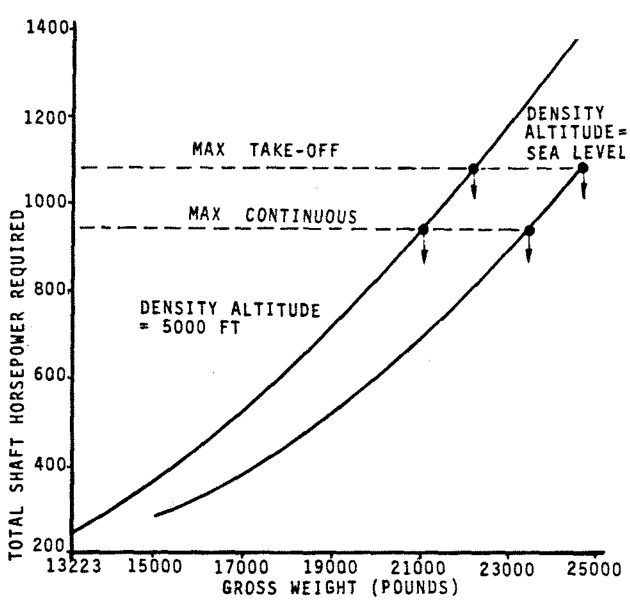

Figure 6 - Shaft Horsepower Required for Hover vs Gross Weight

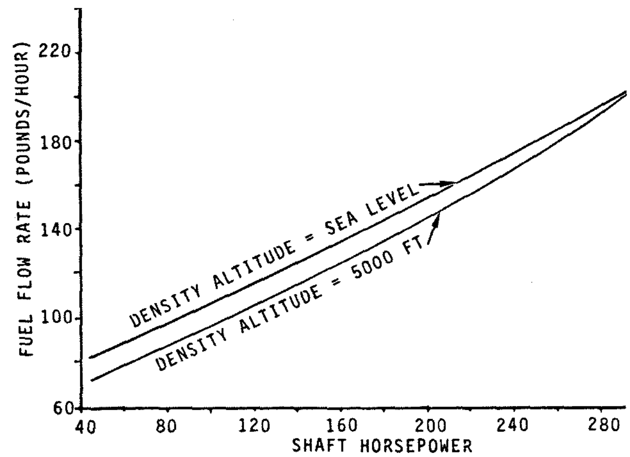

Figure 7 - Fuel Flow Rate vs Shaft Horsepower for T63 Engine

For a given payload the hover endurance has been calculated by considering the corresponding power required and the associated fuel consumption rate ($\dot{w}$) [Figure 7]. In all cases, maximum available fuel is fixed at 3200 pounds with a 10 percent allowance for warmup, taxi, take-off, and climb and a 10 percent reserve at the end of flight. Consequently, usable fuel per flight is assumed to be 2560 pounds. Typically, as the fuel is consumed, the weight of the vehicle decreases causing a drop in required power, which leads to lower fuel consumption.

In the present calculation a piece-wise, decremental approach was taken in accounting for the effect of decreasing fuel weight on vehicle hover endurance.

For example, consider the vehicle with a full payload of 5417 lbs and fuel corresponding to maximum gross weight of 23,435 pounds. Since 10 percent of the available fuel is consumed prior to hover the vehicle begins to hover at a gross weight of 23,115 pounds. The endurance resulting from burning the first 97 pounds of fuel was calculated in one step by using constant values of power required and resulting fuel flow rates corresponding to the initial gross weight of the vehicle. Subsequently, the fuel weight was decremented by 500 pounds for each step, except the last. These computations are illustrated in Table II. Here, it was assumed that the four Auxiliary Propulsion Units [APU] are idling during hover and consume fuel at a constant rate of 60 lbs/hr/engine. Similar results were obtained for the aircraft with various payloads at sea level and 5000 ft density altitude [Figure 8].

From these results it is observed that the endurance in the case of 2000 pound payload is invariant with density altitude change. Typically, with an increase in altitude the power required increases and the static lift decreases; both conditions that lead to lower endurance. However, since the engines operate more efficiently at higher altitude, less fuel is demanded and this yields higher endurance. Apparently, for a payload larger than 2000 lbs, the latter effect dominates.

Table II - Sample Calculation of Hover Endurance

Density Altitude: Seal Level							Fuel Capacity: 3200 Pounds		
Payload (Lb.)	Vehicle Gross Weight (Lb.)	Fuel Weight (Lb)	Total SHP Req'd	SHP Req'd Each Rotor	$\dot{w}$ Each Rotor [Lb/Hr.]	$\dot{w}$ Rotor System [Lb/Hr]	$\dot{w}$ Rotor +$\dot{w}$ APU** [Lb/Hr]	Δ Fuel [Lb]	Time to Burn Δ Fuel [Hr]
5417	23115	2880*	980	227	166	664	904	97	0.107
5417	23018	2783	895	224	165	660	900	500	0.556
5417	22518	2283	850	213	159	636	876	500	0.571
5417	22018	1783	800	200	152	608	848	500	0.590
5417	21518	1283	745	186	145	580	820	500	0.610
5417	21055	783	690	173	139	556	796	463	0.582
5417	20592	320							
							TOTALS:	2560 Lb	3.01 Hr

*320 pound assumed consumed prior to start of hover.
**Four APU at idle consume 240 lb/hr; constant for flight

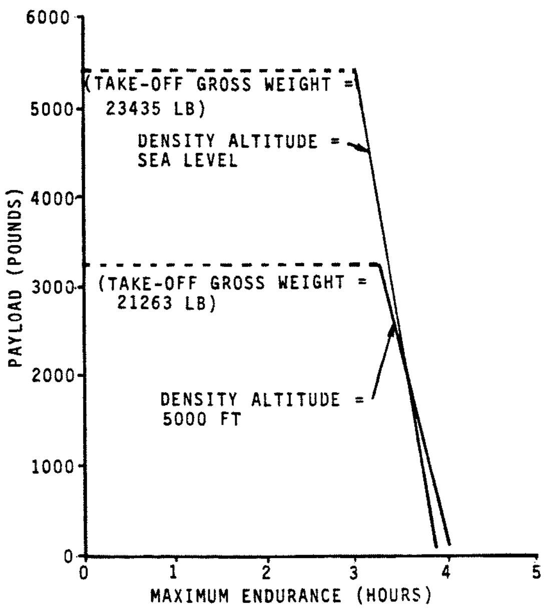

Figure 8 - Payload vs Maximum Endurance in Hover

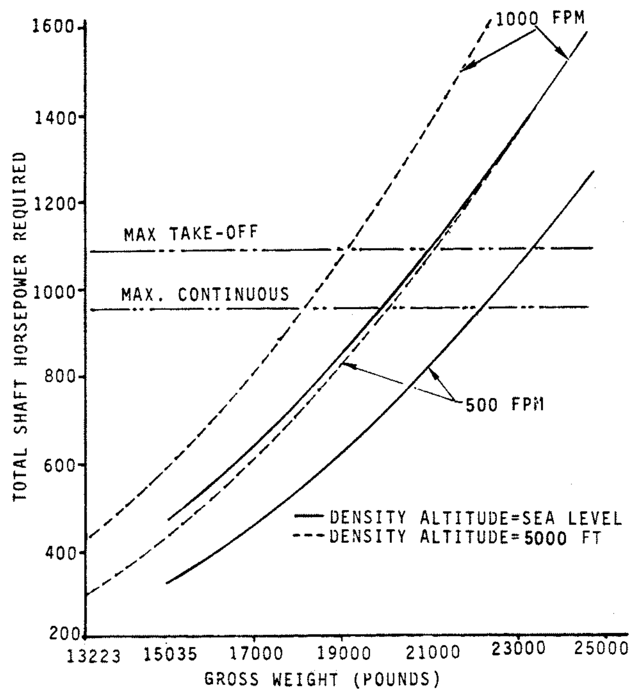

Figure 9 - Gross Weight Effect on Climb Power Requirements

Climb Performance

Performance of the aircraft in climb flight has been estimated by combining the corresponding performance of the individual helicopters and the airship component in the vehicle configuration. Typically, in a vertical climb each of the four helicopter units was assumed to supply enough power to overcome 25 percent of the parasite drag of the airship envelope and support frame and sustain itself in climb. For nominal climb rates of 500 ft/min and 1000 ft/min, the corresponding power required for each helicopter unit was evaluated for various thrust levels or fraction of the aircraft gross weight supported by that module. These results are shown in Figure 9 for sea level and 5000 ft density altitude conditions. Note that the corresponding buoyant lift sustains 13,035 lbs of the aircraft gross weight at sea level density altitude and 11,223 lbs at 5000 ft density altitude. It is observed that at lower gross weights excess power is available to achieve climb rates exceeding 1000 ft/min. This is limited only by the envelope pressure system design which allows climb rates up to 2400 ft/min.

Forward Flight Performance

In order to determine vehicle performance in forward flight, the corresponding power required was evaluated at various airspeeds and gross weights of the aircraft, at specified atmospheric conditions. Here it has been assumed for simplicity that in cruise mode the power required by each helicopter is limited to overcome its own induced, profile and parasite power losses. The power plants with the longitudinal auxiliary thrusters have been selected such that they provide the necessary power to overcome the envelope and support frame drag at all airspeeds.

Consequently, the total power required to cruise at a given speed was obtained by summing the power required to drive the airship envelope and support frame at that speed, and the corresponding power required by each of the four helicopter units cruising at the same speed and sustaining its share of the aircraft gross weight. The former power requirement is shown in Figure 10 for various airspeeds at sea level and 5000 ft density altitude. In this case, for simplicity, the aircraft was assumed to cruise at zero angle of attack. Consequently, the airship aerodynamic lift was not considered here. Similarly, results are shown in Figure 11 for the latter power requirement, at various thrust levels of an individual helicopter. The corresponding power available is also shown in these figures. These results were combined to determine the overall power requirements of the aircraft at various airspeeds and gross weight conditions at sea level density altitude [Figure 12]. Similar data was also obtained at 5000 ft density altitude [Figure 13]. The power limitations of the rotor systems and the auxiliary thruster power plants are also indicated in these figures. It is observed that the forward flight power requirement characteristics of this vehicle at various gross weights are similar to that of a conventional helicopter. These results are used next to determine typical performance characteristics, such as maximum speed, payload vs maximum range and payload vs maximum endurance.

Maximum Speed. For a conventional aircraft the maximum cruising speed is typically given by the airspeed at which the power available is equal to power required for the same operational condition. For the present aircraft there are two sources of power. Consequently, the maximum speed here is determined by the power limit on auxiliary thrusters for lower gross weights and the rotor system power limit for higher gross weights [Figures 12 and 13]. For instance, assuming a nominal efficiency of 75 percent for the auxiliary propulsion system, the power available to the APU at sea level density altitude is 450 hp [Figure 10] and the corresponding maximum speed of the aircraft with no payload [gross weight = 18,018 lbs] should be 50 knots or better, depending upon the power supplied by the helicopter powerplants to overcome envelope-structural frame drag. Similarly, maximum speed for the aircraft with full payload [gross weight = 23,435 lbs] should be 48 knots. At 5000 ft density altitude, it was found that the power available from the longitudinal auxiliary propulsion units is decreased by 50 shp. But the power required to overcome the airframe drag was also lower by 14 percent. Consequently, no significant change in maximum speed was observed at higher density altitude.

Payload versus Maximum Range. In Figures 12 and

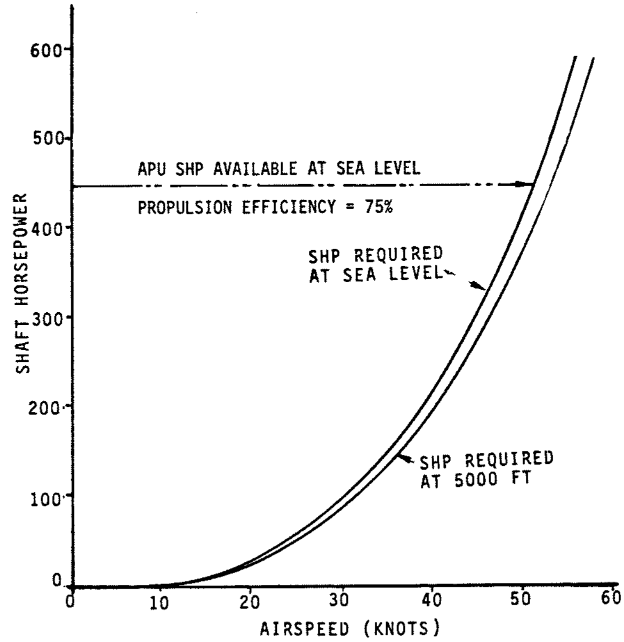

Figure 10 - Forward Flight Power Requirement for Airship Envelope and Support Frame

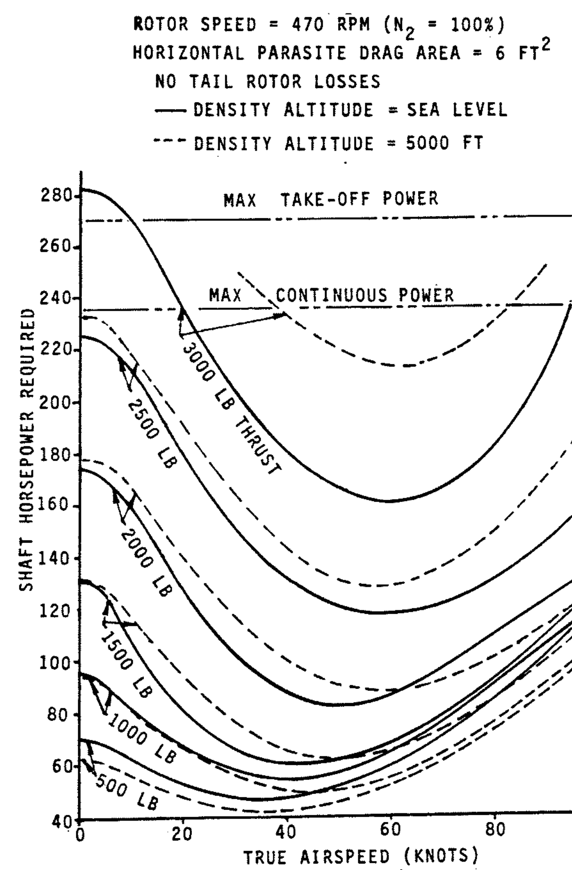

Figure 11 - Typical Forward Flight Power Requirement of the OH-6A Helicopter

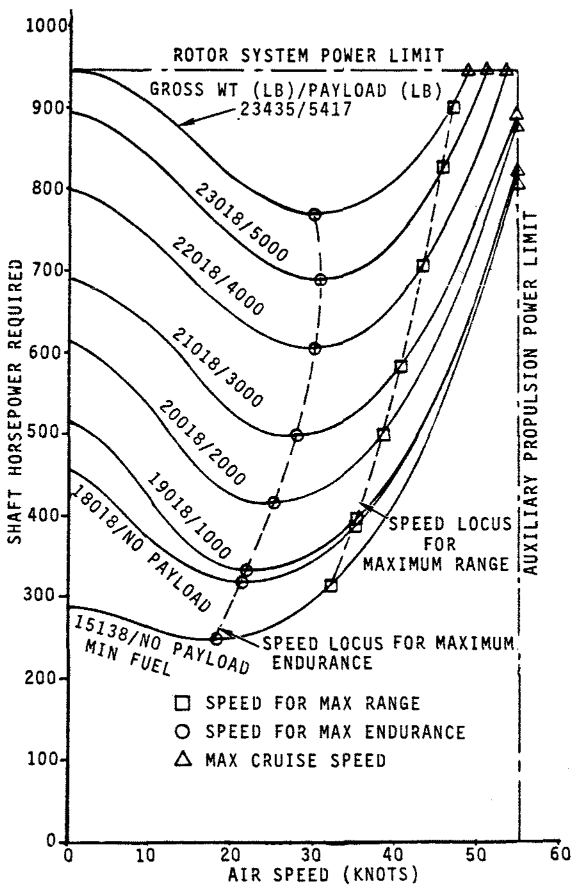

Figure 12 - Vehicle Power Requirements in Forward Flight for Various Gross Weights at Sea Level

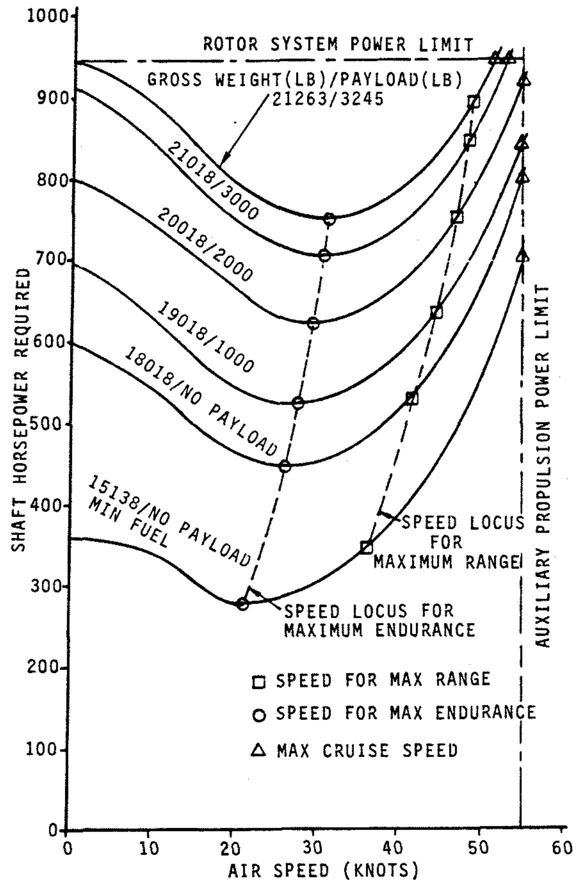

Figure 13 - Vehicle Power Requirements in Forward Flight for Various Gross Weights at 5000-Foot Density Altitude

13 each curve represents the power required by the vehicle to carry the indicated payload with a full tank of fuel. For a particular value of gross weight corresponding to a specified payload, the speed for maximum range is given by a line drawn through the origin and tangent to the power-speed curve. Typically, a maximum range flight profile of the aircraft at sea level consists of initially cruising at the speed for maximum range corresponding to that gross weight. As the fuel is consumed the weight of the vehicle decreases and its power requirements are lowered. In effect, this requires flying at a lower speed for maximum range, as the vehicle progresses down the locus of maximum range velocity [see Figure 12] until all fuel is consumed.

As an example, consider a maximum range flight of the aircraft at sea level, carrying the maximum payload of 5417 pounds [see Table III]. Since 10 percent of the fuel is considered consumed prior to beginning cruise, the weight of the vehicle at the start of cruise is 320 pounds less than that represented by the 5417 pound payload curve in Figure 12. Interpolating between the 5417 pound and 5000 pound curves of this figure gives an initial maximum range velocity of 46 knots at a required power of 840 shaft horsepower. It is assumed that in cruise, the two laterally directed APU's are shut down. The 840 shp is then shared by six engines. Note that equal power output assumed from each engine does not influence the fuel flow calculation, since the fuel flow rate variation with the shp is nearly linear. The corresponding fuel flow rate, determined from Figure 7, was used to compute the cruise time elapsed and distance traveled while consuming the initial 597 pounds of fuel. [See Table III].

For the next step, the maximum range velocity was selected from a point on the maximum range velocity curve for a gross weight of 597 pounds lower than the previous point. This procedure was continued until 2560 pounds of fuel was consumed. For ease of interpolation, the fuel was decremented in 500 pound steps except for the first and last decrement. Similar calculations were performed for other gross weight conditions corresponding to various payload conditions. Maximum range in each case was determined both at sea level and at 5000 ft density altitude as shown on Figure 14.

From these results it is observed that for the flight at sea level there is an optimal payload of 2000 pounds at which peak maximum range can be achieved. Apparently, with decreasing gross weight the power required by the helicopter rotors decreases along with the maximum range speed. The former effect tends to increase the range while the latter tends to decrease it. For payloads from maximum value to 2000 pounds, the range increment due to the reduced power requirement overrides the decrement because of the slower speed for maximum range. The converse is apparently true for payloads smaller than 2000 pounds.

Table III - Sample Calculation of Maximum Range

Density Altitude: Sea Level								Fuel Capacity: 3200 Pounds		
Payload [lb]	Vehicle Gross Weight [lb]	Fuel Wt. [lb]	Speed for Maximum Range [kt]	Total SHP Req'd	SHP Req'd Each Engine	$\dot{w}$ Each Engine [lb/hr]	$\dot{w}$ Total [lb/hr]	Δ Fuel [lb]	Time to Burn Δ Fuel [hr]	Dist. For Δ Fuel [Nm]
5417	23115	2880*	46	840	140	124	744	597	.802	36.91
5417	22518	2283	44.5	770	128	118	708	500	.706	31.43
5417	22018	1783	43.5	710	118	114	684	500	.731	31.80
5417	21518	1283	42	650	108	109	654	500	.765	32.13
5417	21018	783	41	590	98	105	630	463	.735	30.13
5417	20555	320					TOTALS:	2560	3.74	162.4

*320 lb assumed consumed prior to start of cruise

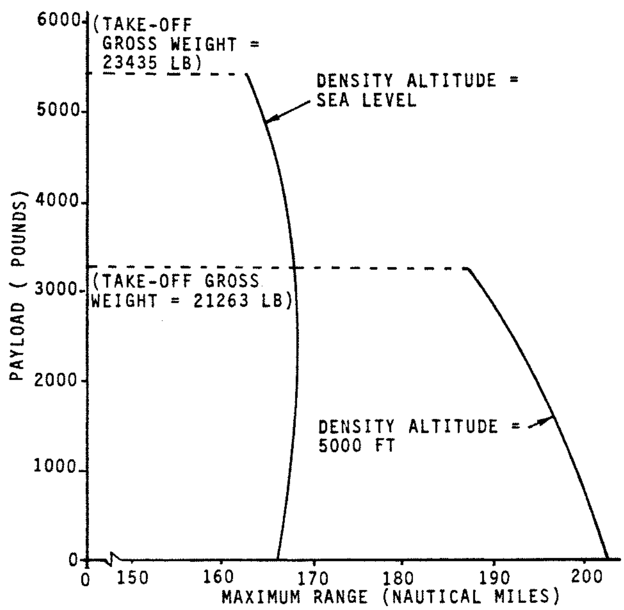

Figure 14 - Payload vs Maximum Range

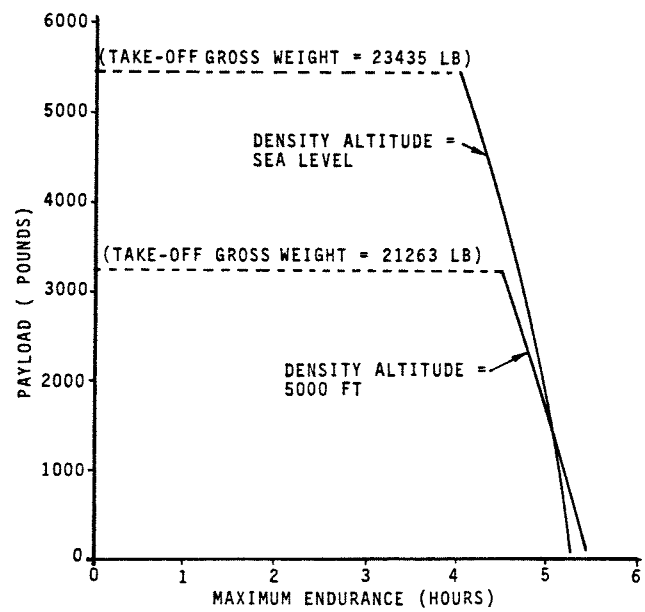

Figure 15 - Payload vs Maximum Endurance

Payload versus Maximum Endurance. In Figures 12 and 13, speed for maximum endurance of the vehicle at various gross weights have been identified. For a given payload the maximum endurance was calculated in a manner similar to that used for maximum range calculation. The corresponding results are shown on Figure 15. As previously observed in hover endurance, if the payload is approximately 1250 pounds, the vehicle cruise endurance in such a case was found to be invariant with change in density altitude.

Concluding Remarks

Based on the performance characteristics estimated here for the BQRA, it is believed that this vehicle configuration is adequate for flight research. Aircraft performance during hover appears to be sufficient to flight test this vehicle in a wide range of payloads, altitudes and ambient temperatures. Enough power exists to permit aircraft climb and cruise at moderate flight speeds. This should allow an evaluation of the vehicle performance in typical mission profiles intended for the full-size vehicle.

Forward flight performance of this aircraft has been found to be similar to that of a conventional helicopter. Both the maximum range and endurance predicted here could be extended by trading-off payload for fuel in auxiliary tanks. Performance estimates which consider the effect of using airship aerodynamic lift during the aircraft's cruise mode, should be obtained. These may suggest trim conditions in cruise which result in greater range and endurance. More refined computations should perhaps be conducted to validate the optimal payloads identified here for maximum range and endurance of the aircraft.

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.

References

1. Williams, D. E.: "The Development of a Heavy Lift Airship," American Helicopter Society Paper No. 80-9, Washington, D.C., May 1980.

2. Goodyear Aerospace Corporation: "Preliminary Design of a Hybrid Airship for Flight Research." Prepared for NASA Ames Research Center, Moffett Field, California, NASA CR-166246, July 1981.

3. Hughes Helicopters: Engineering Flight Test of the OH-6A Helicopter [CAYUSE], Phase D. Final Report, April 1969, USAASTA Project No. 65-37.