National Transportation Safety Board
Washington, D.C. 20594

Aircraft Accident Brief

Accident Number:	LAX99MA251
Aircraft and Registration:	Schempp-Hirth Nimbus-4DM, N807BB
Location:	Minden, Nevada
Date:	July 13, 1999
Adopted On:	September 27, 2002

HISTORY OF FLIGHT

On July 13, 1999, at 1310 Pacific daylight time, a Schempp-Hirth Nimbus-4DM motorized glider, N807BB, experienced an in-flight breakup while maneuvering near Minden, Nevada. The commercial glider pilot and a passenger were fatally injured, and the glider was destroyed. The glider had departed from Minden-Tahoe airport at 1240 and was operated by the owner/pilot under the provisions of 14 Code of Federal Regulations (CFR) Part 91. No flight plan was filed for the local area soaring flight. The glider was certificated in the United States in the experimental category for exhibition and racing, and it was also certificated in Germany in the standard class, utility category.

A glider pilot who witnessed the in-flight breakup stated that his glider was soaring about 1,000 feet below the accident glider when he observed the accident glider in a high-speed spiral with a 45-degree nose-down attitude. After two full rotations, the rotation stopped, the flight stabilized on a northeasterly heading, and the nose pitched further down to a near-vertical attitude. The accident glider was observed to level its attitude, with the wings bending upward and the wing tips coning higher. The outboard wing tip panels departed from the glider, the wings disintegrated, and the fuselage dove into the ground.

Other witnesses who were flying gliders in the area stated that the accident glider was in a tight turn, as if climbing in a thermal, when it entered the spiral. In the later portion of the sequence observed by these witnesses, the glider appeared to be recovering from a spin.

The accident glider was climbing in a thermal when one witness saw it at 9,000 feet msl. The glider then departed the thermal on an easterly heading for another thermal. When the witness saw the glider again, he stated that it was coming directly towards him at a 45-degree nose-down angle with the wings bowed. The witness was 180 degrees into a 360-degree turn and heard a Mayday call (later determined to have come from another glider pilot who saw the accident sequence). He did not see the accident occur, but did see fiberglass particles scattering down.

Another witness saw the accident glider climb from 9,000 feet to 11,000 feet msl. He also saw the glider in a spiraling 45-degree nose-down angle. He stated that the nose-down attitude increased to 80 degrees after completing one to two 360-degree turns. The witness reported that the glider became stabilized on a westerly heading with the wings bowed up 45 degrees. He then observed the wings collapse.

Another witness caught something out of the corner of his eye moving at a very high rate of speed. He reported that the accident glider spiraled down and completed one 360-degree turn. He stated that he could not tell if the glider was in level flight when the wings failed.

One ground witness reported that the accident glider had a 45-degree nose-down attitude and was in a "hard over left hand turn."

PERSONNEL INFORMATION

The occupant seating positions were determined from interviews of witnesses who helped launch the glider on the accident flight and from the Douglas County Coroner's identification of the victims. Information concerning the airman and medical certification status of the pilot and passenger were obtained from certified copies of their Airman and Medical Certification files maintained by the Federal Aviation Administration (FAA). Additional information on flight times and other related areas were obtained from a review of their personal flight records and/or interviews with family members and other pilots familiar with their activities.

The glider was equipped with full flight controls at both pilot stations, except for the engine primer control, which was only available in the front cockpit.

Front Seat Occupant

The aircraft owner occupied the front pilot seat, which according to the manufacturer is the normal pilot station for the glider. He held a commercial pilot certificate with a category rating for gliders, and private pilot privileges for airplane single engine land. His private pilot certificate for airplanes was first issued in January 1942, and the commercial glider certificate was obtained in February 1948. FAA medical certification files show no record that a medical certificate was ever issued to the pilot. The provisions of 14 CFR 61.23 notes that pilots exercising the privileges of a glider category rating are not required to hold any class of medical certificate.

At the time of the accident, the pilot had accumulated a total time in gliders of 3,311 hours. Over a period from delivery of the glider in August 1996 to the date of the accident, the pilot had accrued 103 hours in the Nimbus-4DM. The pilot accomplished an initial checkout in the Nimbus-4DM by a certified flight instructor on September 1, 1996, with six flights totaling 3.2 hours. During the 1997 soaring season from April 17, 1997, to October 30, 1997, the pilot flew 66 hours, 64 of which were in the Nimbus-4DM. No additional flight activity was recorded until the start of the 1998 season on June 1, 1998, and the pilot logged 34 hours between that date and the end of the season on October 20, 1998. Again, no flight activity appeared in the pilot's logbook between the season's end and the beginning of the 1999 season. The first flight activity was recorded on May 20, 1999, and from then until the accident, six flights were logged totaling 8 hours.

The pilot's most recent biennial flight review was completed on October 20, 1998, in a Puchacz glider.

Rear Seat Occupant

The passenger was a retired U.S. Navy flag officer who was first designated a naval aviator in June 1942. On January 7, 1972, he applied for a civilian commercial pilot certificate on the basis of military competency and was issued one with an airplane multiengine land class rating. In August 1972, he applied for additional ratings of airplane single engine land and instruments on his commercial certificate. On that application, he listed a total glider time of 4.5 hours in 7 flights. A glider category rating limited to aero tow only was added to his commercial certificate on November 14, 1982, with the application form listing 28 glider flights totaling 8.2 hours. As of the accident, the passenger had accrued about 7,500 total hours. His personal flight records were not located and, based on interviews with persons who had knowledge of his glider activity, he had accumulated an estimated 400 hours in gliders.

The most recent second-class medical certificate was issued to the passenger on April 1, 1998, and contained the limitation that correcting lenses be worn while exercising the privileges of his airman certificates. On the application form for the medical certificate, he listed a total pilot time of 7,000 hours, with 30 flown in the last 6 months.

The passenger's wife reported in a telephone interview that in the week before the accident he was in good spirits and good health. His sleep and eating habits were regular. The only medication he was taking was Feldene for arthritis pain. This medication was disclosed to the aviation medical examiner at his last flight physical.

She stated that she had no detailed knowledge of her husband's flight time and believed that he rarely logged anything. His first exposure to gliders was in England in 1953 when he was attending test pilot school, and also somewhat later when he was assigned to the Naval Air Station Patuxent River in Maryland. She noted that he was in a partnership with two other gentlemen in a glider that was kept at the Minden-Tahoe Airport; however, she was unaware of the glider make or model. Her husband flew soaring flights on the Saturday, Sunday, and Monday before the accident.

The passenger's partners in the glider he part-owned were contacted and interviewed by telephone. They reported that the passenger had co-owned a Schleicher ASW20B glider since 1995. His partners described him as a relatively low-time glider pilot. Based on partnership records and personal knowledge, the witnesses said the passenger had flown their glider about 15 hours in the last 60 days.

In an interview, the accident glider owner's son (also a glider pilot with extensive experience in the accident aircraft) said that to his knowledge, the passenger had only flown the Nimbus-4DM once before. He stated that the Nimbus would handle very differently from the glider the passenger typically flew. The son stated that the Nimbus-4DM is very demanding in terms of rudder control inputs to prevent unwanted yaw and rolling moments, much more so than the passenger's glider. He also reported that comparatively speaking, the Nimbus-4DM would be easy to get into a cross controlled situation versus the flight characteristics of the passenger's glider.

Additional Personnel Information

Witnesses who interacted with the pilot and passenger 2 days before the accident and on the day of the accident stated that both were in good spirits. The witnesses further reported that the pilot and passenger were both looking forward to the day's flight activity with anticipation. Both seemed to be in good health, and nothing appeared to be wrong.

The instructor who was the check pilot for the accident pilot stated that "it takes a lifetime to learn to fly this type of sailplane [Nimbus-4DM]...." He stated that the accident pilot had owned the glider for approximately 2 years. He stated that he provided 5 hours of new aircraft checkout, which included incipient spin recovery.

The only item of note was that the check pilot had to remind the accident pilot on numerous occasions to hold the elevator in the nose-up position during maneuvers.

According to the check pilot, the accident pilot stated that he had not flown this type of glider much before purchasing this one.

AIRCRAFT INFORMATION

Aircraft Description

The Nimbus-4DM is a 2-seat, high-performance motorized glider, constructed from fiber reinforced plastic (FRP) composites, featuring full span flight controls and a T-tail (with fixed horizontal stabilizer and two-piece elevator). The manufacturing process uses a hand lay-up of composite material plies and epoxy resins.

The wing (87-foot span) consisted of three sections per side consisting of a wing tip, outboard section, and inboard section. The inboard sections mate at the fuselage and the outer wing sections mate with the inboard sections approximately 12.6 feet outboard of the fuselage root chord. The wing shells are a carbon fiber/foam core sandwich construction with one main spar constructed of a glass fiber/foam core shear web and carbon fiber spar flanges. A single-vane flap spans the entire inboard wing section. Three sections of ailerons (that is, inboard, center, and outboard) span the outboard wing section with a fourth aileron, used to minimize the effects of adverse yaw, attached to the wing tip.

The forward fuselage (cockpit) is constructed of Kevlar, carbon and glass fiber laminate, reinforced by a double skin on the sides with integrated surrounding canopy frame and seat pan mounting flanges. The single-piece canopy hinges sideways and opens to the right. The aft fuselage section is constructed of a pure carbon fiber monolithic shell, stiffened by carbon fiber/foam core bulkheads and glass fiber webs.

The horizontal stabilizer is constructed of glass fiber/foam core sandwich with carbon fiber reinforcements. The elevator halves are a hybrid composite (carbon and glass fiber) monolithic shell. The vertical stabilizer is carbon fiber/foam core sandwich construction. The single-piece rudder is constructed of glass fiber/foam core sandwich.

The flight controls are all push/pull tubes except for the rudder, which is controlled via cables.

The Nimbus-4DM is powered by a liquid-cooled 44 kW Bombardier (Rotax) model 535C engine with a 3:1 belt reduction drive. The powerplant is housed in the fuselage immediately aft of the wing. An electrically driven spindle drive (jackscrew) extends the propeller pylon upwards and forward from the engine bay. When stowed, two doors mounted to the rear fuselage conceal the powerplant. The jackscrew is attached between the airframe and the upper forward end of the pylon such that when the jackscrew is retracted (shortened) the pylon is pulled upwards and forward into its flight position.

Fleet Size and Accident History

The Nimbus-4DM is a model of the "Nimbus-4 Family," which consists of single-seat and two-seat gliders and motorgliders. The different models are:

- Nimbus-4: a single seat glider, Type Certified in Germany January 1, 1994. Total number produced: 11
- Nimbus-4D: a two-seat glider, Type Certified in Germany February 24, 1995. Total number produced: 9
- Nimbus-4T: a single seat self-sustaining motor glider with a retractable engine, Type Certified in Germany June 15, 1993. Total number produced: 12
- Nimbus-4M: a single seat self-launching motor glider with a retractable engine, Type Certified in Germany January 1, 1994. Total number produced: 10
- Nimbus-4DT: a two seat self-sustaining motor glider with a retractable engine, Type Certified in Germany May 5, 1995. Total number produced: 6
- Nimbus-4DM: a two seat self-launching motor glider with a retractable engine, Type Certified in Germany November 7, 1995. Total number produced: 37

In total as of the date of the accident, 33 single-seat and 52 two-seat models had been produced. According to the Federal Republic of Germany Luftfahrt-Bundesamt (LBA), Germany's equivalent of the FAA, components of the Nimbus-4 gliders and motorgliders are used in various models of the product line of the Schempp-Hirth company. The development of the Nimbus-4 family is as a design derivative of the predecessors, the Nimbus-2 and Nimbus-3.

Investigators queried the LBA regarding the accident history of the Nimbus-4DM. There are only three accidents worldwide on file with the Federal Bureau of Aircraft Accidents Investigation (BFU), Germany's equivalent of the NTSB. The first was a noninjury long landing accident in Fayence, France, on September 4, 1994. The second involved a collision with the ground during takeoff in Fuentemilanos, Spain, on July 27, 1997, which resulted in two fatalities. The improper installation of the horizontal stabilizer led to the third accident in Lüsse, Germany, on June 13, 1999, in which two occupants were injured during an attempted takeoff when the stabilizer separated from the empennage just after liftoff.

Regarding other service/accident experience with the Nimbus-4 family of gliders, the BFU has recorded four incidents/accidents with the single-seat versions. Three events are known of noninjury accidents during off-field landings, and one fatal accident was due to collision with a mountain.

Additionally, during training for the Glider World Championships in France in 1995, a Nimbus-4 (owned by the French Air Force) was destroyed in a midair breakup accident. The glider entered a wave cloud, lost control, and broke up at a speed beyond 400 km/h (The never exceed speed, or "Vne" is 285 km/h, and the design dive speed, or "Vd", is 324 km/h). The pilot was rescued by bailing out.

According to the LBA, "As far as we know, none of the incidents/accidents recorded indicated a technical failure."

Safety Board investigators became aware of another accident involving a Nimbus-4DM that occurred in Spain shortly after the Minden, Nevada, accident. According to the Comision de Investigacion de Accidentes y Incidentes de Aviacion Civil, Spain's equivalent of the NTSB, the glider broke up in flight following a high-speed excursion beyond Vne. The Spain inquiry is still under way; however, according to preliminary information supplied by the Spanish authorities, the pilot stated they were in a turn when a heavy thermal caused the glider to enter a steep descending spiral. The pilot could not recover the aircraft from the spiral and the airspeed quickly exceeded Vne. The pilot then reported that the right wing failed and he bailed out. Information provided by Spanish authorities indicated that the wing failure locations on the Spanish accident aircraft were about 1.5 to 2.0 meters outboard of the inboard primary failure points on N807BB. (See TESTS AND RESEARCH section of this narrative and the Wichita State University National Institute for Aviation Research report, which is attached to the Airworthiness Group Chairman's Factual Report that is contained in the docket for this accident.)

Joint Airworthiness Requirements Certification

The glider is a production aircraft manufactured by Schempp-Hirth Flugzeugbau GmbH in the Federal Republic of Germany. The LBA issued type certificate No. 868 on July 11, 1995, certificating the glider in the standard class, utility category. The certification basis listed on the type certificate is Joint Airworthiness Requirements (JAR) 22 for Sailplanes and Powered Sailplanes, effective June 27, 1989, inclusive of Change 4 and amendments 22/90/1, 22/91/1, and 22/92/1. Exemptions to the JARs on the type certificate included 22.201(f)(5)(ii), 22.153, 22.173(b), 22.175(a), and 22.207(a). A special compliance was established on the basis of an equivalent level of safety with JAR 22.1093(b).

In response to investigators' queries concerning the rationale for the exemptions granted to the JAR standards during certification of the glider, the LBA responded that the specific JARs cited in the exemptions concern stall and spin characteristics, and stability. The LBA noted in its response that several certificated motorgliders with retractable engines, including the Nimbus family, exhibit characteristics with the engine extended and idling that cannot fully comply with the cited regulations. The reason given by the LBA for these exemptions is that, "During idling of the engine, the vibration of the engine supersedes the natural aerodynamic stall warning of the motorglider…for a motorglider with a retractable engine, the flight with engine extended and idling is a transition phase." As a part of the exemption grant, a prohibition against landing with the engine idling was placed into the Aircraft Flight Manual (AFM). The LBA conducted a flight evaluation to check the characteristics, and no "unacceptable flight behavior in this configuration" was found.

U.S. Certification

According to the Small Airplane Directorate of the FAA Aircraft Certification Office, the manufacturer never applied for a United States Type Certificate under the Federal Aviation Regulations, and the glider is only eligible for certification in the experimental category. The owner applied for and received a U.S. Special Airworthiness Certificate in the experimental category for the purpose of exhibition and air racing on August 30, 1996. The operating limitations letter issued with the certificate by the FAA are contained in the docket for this accident.

Maintenance History

The aircraft logbooks were obtained and reviewed. The glider was manufactured in July 1996 and delivered new to the pilot. The most recent condition inspection was accomplished on May 18, 1999, 19 hours before the accident. The Rotax 535C engine and propeller had accumulated a total time of about 30 hours. No manufacturer's service bulletins or JAR airworthiness directives were recorded in the records. There was no damage or repair history evident in the record. The airframe had accrued a total time in service of about 201 hours at the time of the accident.

The pilot's son stated that there were no discrepancies with the condition of the glider at the end of his flight on July 4, 1999.

Fuel and Water Ballast Load Information

Records of the company that has the fueling concession at the Minden-Tahoe Airport were reviewed, with no record found of the pilot having purchased fuel in May, June, or July. The company representative stated that it is common practice for glider pilots to fill up a portable container with fuel directly from the pump. The company representative further reported that they do not keep records of those transactions.

Concerning the question of fuel on board at departure, while packing some things at the glider trailer after the accident, the pilot's son found the fuel cans they used to refuel the glider. Based on the amount of fuel left in the cans and the amount that was left onboard at the conclusion of his last flight, he stated his belief that the glider had 35 liters on board at takeoff (maximum fuel capacity is 48.5 liters).

With regard to the water ballast system, he stated that the system is designed to increase the wing loading for race competitions. He and his father used the system only once, about 2 years ago shortly after the glider was delivered, and both were very disappointed with the results compared to the weight penalty and the resultant limited payload. Neither he nor his father had used it since. He stated that no water was on board when he last flew the glider on July 4, and was positive that his father would not have used the system.

Gross Weight and Center of Gravity

The gross weight and center of gravity for the glider was calculated for three separate conditions (each including the known weights of the occupants and their parachutes): (1) dry without fuel or water ballast; (2) with full fuel and no water ballast; and (3) with full fuel and full water ballast. The weight calculations were based on (1) the empty weight and center of gravity obtained from the approved aircraft flight manual;(2) the known weights of the occupants, their parachutes, a full fuel load of 8.08 gallons; and (3) a full water ballast load of 43.59 gallons. The moment arms for the appropriate station locations were obtained from the glider maintenance manual, Section 7, Weight and Balance. The sheet of calculations is contained in the docket for this accident.

According to the approved aircraft flight manual, the maximum allowable gross weight is 1,808 pounds, with a center of gravity range between the 4.13-inch forward limit and the aft limit of 10.24 inches (measurements are referenced to the datum point, which is the wing leading edge at the root).

In the condition of the occupants and glider ready for flight without fuel or water, the gross weight was 1,733.5 pounds with a calculated center of gravity at 6.28 inches. With full fuel and no water ballast, the gross weight increased to 1,781.98 pounds and the center of gravity changed to 6.36 inches. Adding full water ballast increased the gross weight to 2,145.51 pounds and moved the center of gravity to 7.62 inches.

Flight Characteristics and Performance Limitations

Pilots with experience in the Nimbus-4 series gliders stated that it is an extremely high-performance glider used for long-distance flights and competition soaring. They further reported that this glider is not designed for aerobatics and is not intended for use in extreme weather conditions. They reported that due to the 87-foot wingspan, the glider was particularly sensitive to over input of the rudder control during turns with a resulting tendency for unwanted rolling moments. The manufacturer reported that to avoid undesired rolling moments, once the bank is established "the ailerons must be deflected…to get a symmetric lift distribution [which] means stick deflected against the bank…."

According to the AFM, maneuvering speed (Va) is 180 km/h (97 kts, 112 mph) and page 4.5.3.2 states, "Full deflections of control surfaces may only be applied" at this speed and below. Never exceed speed (Vne) is 285 km/h (154 kts, 177 mph) and at this speed, "only one third of the full [control] deflection range is permissible." The section specifically states, "Avoid especially sudden elevator control movements."

Other noteworthy airspeeds are Vs (stall speed), Vy (best rate of climb), and Vd (design dive speed). Vs is 82 km/h (44 kts, 51 mph), Vy is 95 km/h (51 kts, 59 mph), and Vd is 324 km/h (175 kts, 201 mph). In response to an inquiry from Safety Board investigators, the manufacturer reported that, assuming a 45-degree nose-down attitude with airbrakes closed, the glider would accelerate from Vs to Vne in 8.6 seconds, with an additional 1.8 seconds to accelerate from Vne to Vd.

With regard to the airbrake extension speed limitation, page 4.5.3.2 of the AFM states that, "Airbrakes may be extended up to Vne…however they should only be used at such high speeds in emergency or if the maximum permitted speeds are being exceeded inadvertently." The manufacturer supplied flight test and engineering analytical reports that established there were no nose pitching moments associated with extension of the airbrakes up to Vne. In an email correspondence to Safety Board investigators, the manufacturer noted that the airbrakes would function like spoilers and have the effect of shifting the aerodynamic loads outboard on the wings. The control linkages for the airbrakes and flaps are interconnected so that when full airbrake deployment is achieved, the flaps are extended to their full down limit.

The maneuvering load factor limits (in units of gravity or g's) are delineated in the limitations section of the AFM and are:

Flaps up at Va	+5.3	-2.65	
Flaps up at Vne	+4.0	-1.5	(Also limit with airbrakes closed)
Full Flaps	+4.0	-1.5	
Airbrakes Extended	+3.5		(no negative limits specified, speed is Vne)

Wing Structural Strength and Design Safety Margins

Information presented in this section was obtained from the manufacturer and is based on certification flight test reports, proof static strength substantiation tests for certification (production wings tested to failure in a ground load fixture), and/or the manufacturer's computational analysis derived from these tests.

While detailed information on the wing failure locations and signatures can be found in both the WRECKAGE AND IMPACT and TESTS AND RESEARCH sections of this narrative, the wing failure points (listed as wing stations in meters from the fuselage centerline) should be noted. The left wing failed at three locations: 2.3 meters (this failure was related to ground impact), 5.4 meters, and 9.8 meters. The right wing failed at two locations: 5.5 meters and 9.9 meters.

Wing Design Safety Margins

As with the pertinent U.S. aircraft certification regulations, the JARs require a minimum safety margin above the design limit load, which is defined as ultimate load. According to the LBA, JAR 22 (covering the certification of gliders and motor gliders) requires a 1.5 safety margin above design limit load.

The manufacturer supplied both tabular data and a chart presentation illustrating the safety margin (factor above limit load) for the spar cap in relation to wing station in meters (measured from the fuselage center line). Review of the data discloses that from station zero out to 5.3, the safety margin varies between 1.75 and 1.55, with the lowest margins occurring at stations 2.0 through 2.7 and 4.6. Outboard of station 5.3, the margin decreases to 1.5 at station 6.6, then rises in an almost linear fashion to 4.3 at the wing tip.

Wing Bending Moments and Shear Force

In response to requests from Safety Board investigators, the manufacturer provided data concerning the wing bending and shear force moments (by wing station) under various speed, flap/airbrake configurations, and load conditions. The charts and tabular data are contained in the docket for this accident.

The manufacturer's response letter, dated July 10, 2001, noted that wind tunnel tests established that "…lift in the area of the airbrakes is reduced with airbrakes extended…." The chief engineer for the company stated, "This lift reduction in the airbrake range [area of the wing where they are located] causes an increased lift outside of the airbrake range…" with a corresponding increase in loading on the outer wing panels. He further noted that the effect of airbrake extension on the load pattern is smaller with increasing g-loads and angles of attack.

For each configuration, the bending moments are the greatest at the root and decrease progressively as the station numbers move out toward the wing tips. Airbrake extension induces a large increase in the bending moment values; however, the curve shape remains largely the same. At Vne and 1 g, the bending moment at the root goes from 5,000 to 25,000 Newtons/meters with airbrake extension. With a speed increase to Vd, the bending moment at the root changes from -1,000 to 25,000 Newtons/meters with airbrake extension.

The shear force values with airbrakes retracted are the largest at the root at just under 2,500 Newtons and decrease to zero at the tip. Airbrake extension at Vne (1 g) causes a sharp rise in the plotted curve at wing station 2.0, which increases to a peak value of 3,400 Newtons at wing station 4.3, with a linear decrease after that to zero at the tip. The plotted curves are the same at Vd under the condition of (1 g).

As the load factors increase, the curve shapes of the plotted data remain the same; however, the moment values increase. As examples, at a load of 4 g's the bending moment goes from 50,000 to 80,000 Newtons/meters, while the shear force values vary between 9,000 and 10,000 Newtons from the root out to wing station 5.0.

Wing Deflection Under Load

The manufacturer supplied tabular and graphical data of the wing deflections under various load and aerodynamic conditions. This data was determined by both analytical computations and in ground and flight tests. At the design load limit (3.5 g's) with airbrakes extended and at Vd, the wings were deflected to a 31-degree angle. At the ultimate load limit, the deflection was 46.5 degrees. (The full graphical and data file presentation is contained in the docket for this accident.)

Aeroelastic and Static Strength Substantiation of the Wing Structure and Controls

The loads analysis for the wing is part of a stress analysis summary, which includes the static strength substantiation of the wing that was submitted to the LBA during certification.

A copy of the stress analysis summary of the Nimbus-4DM was supplied by the LBA and the manufacturer to the engineers of the FAA Small Airplane Directorate for evaluation. The substantiation report regarding the calculation of (static) aerodynamic divergence is included in the strength calculation of the wing. These calculations of the wing strength were done with consideration of bending and torsional deformation. Proof of compliance regarding the dynamic aeroelastic divergence of the wing is covered by the flutter substantiation report of Prof. Dr. Ing. Niedbal, dated June 30, 1994, and are included in the following two reports:

- *Aeroelastic investigation of the Motorglider Nimbus-4DT*, dated June 24, 1994.
- *Aeroelastic investigation of the Motorglider Nimbus-4DM*, dated June 30, 1994.

The LBA reported strength substantiation of the controls were accomplished by load tests up to the following safety factors:

Elevator control:	1.52
Rudder control:	1.55
Airbrake control:	1.57
Aileron control:	1.69
Wingflap control:	1.56

According to the LBA, Schempp-Hirth has performed several tests to show compliance with JAR 22.305, which covers binding of the roll control surfaces with the wing deflected under limit load. The function and free movement of the aileron control has been tested in a range between 1.14 and 1.33, for the wing flap settings of -2, 0, and L.

In a letter to Safety Board investigators, the manufacturer stated that a review of the flight test data showed no indications of aeroelastic divergence up to limit load at Vd.

Static Proof Test Article Failures

According to the LBA, during certification compliance tests, the manufacturer placed a conforming production wing article in a ground test fixture and subjected it to the ultimate load limits in both bending and shear. The spar did not fail. Compressive buckling failures similar to those observed on the accident glider were noted in the upper skin panels at various locations, with the failure axis principally at 90 degrees to the spanwise direction. (Further information on the nature of the buckling skin failures can be found in the Airworthiness Group Chairman's Factual Report and in the Wichita State University National Institute for Aviation Research report, "Investigation of a Schempp-Hirth Nimbus-4DM Motorglider 'Inflight Breakup,'" both of which are contained in the docket for this accident.)

Elevator Stick Force vs. Speed

In correspondence to Safety Board investigators, the manufacturer stated that there was no flight test data available that would allow the prediction of stick force per unit of "g" load. They did supply a graphic plot of data of stick force versus airspeed derived from flight tests to show compliance with JAR 22.175 (static stability). The neutral stick force point on the plot was noted to be 135 kmh (72.89 kts, 83.8 mph), while 11.9 pounds of stick forward force (push) was required at Vne to maintain a fixed attitude.

METEOROLOGICAL INFORMATION

Weather Study

A Safety Board staff meteorologist completed a study of the weather conditions existing at the time of the accident. The complete report is contained in the docket for this accident. As a matter of note, the accident site is located at latitude 38.9819 degrees north and longitude 119.6728 degrees west. The following times are presented as Coordinated Universal Time (UTC or Z), which can be converted to Pacific daylight time by subtracting 7 hours.

Synoptic Situation

A surface plot with sea level pressure contours for July 13, 1999, at 2000Z showed low pressure located west of the accident site.

Surface Weather Observations

Reno, Nevada (KRNO) is located about 32 nautical miles north of the accident site.

1956Z ... few clouds at 9,000 feet, ceiling 18,000 feet broken, 25,000 feet broken; visibility 10 miles; temperature 33 degrees C; dew point 7 degrees C; winds variable at 4 knots; altimeter setting 30.03 inches of Hg.

2056Z ... 10,000 feet scattered, ceiling 18,000 feet broken, 25,000 feet broken; visibility 10 miles; temperature 33 degrees C; dew point 6 degrees C; winds 240 degrees at 6 knots; altimeter setting 30.01 inches of Hg.; hazy

South Lake Tahoe, California (KTVL) is located about 16 nautical miles west-southwest of the accident site.

1947Z ... 12,000 feet scattered, ceiling 20,000 feet broken; visibility 20 miles; temperature 27 degrees C; dew point 10 degrees C; winds 020 degrees at 7 knots; altimeter setting 30.21 inches of Hg.

2047Z ... 8,000 feet scattered, 12,000 feet scattered; visibility 20 miles; haze; temperature 28 degrees C; dew point 9 degrees C; winds 020 degrees at 7 knots; altimeter setting 30.19 inches of Hg.

Upper Air Data

Upper Air Data was taken at the Reno, Nevada, station on July 14, 1999, at 0000Z. In part, the recorded wind direction and speed for altitudes between 8,000 and 12,000 feet msl were as follows:

Height in feet msl	8,002	8,999	10,431	12,001
Wind direction degrees	095	310	120	085
Wind speed knots	1	1	2.9	8.0

Satellite Data

Geostationary Operational Environmental Satellite (GOES) 10 data were reviewed [The times the satellite was viewing the crash site is about 2003Z (2000Z images) and about 2018Z (2015Z images)]. Looping of visible images for times of 1915Z to 2015Z showed clouds decreasing in the area over and south of the accident site. At 2015Z, a new cloud development was seen about 3 nautical miles north-northeast of the accident site [Allowing for satellite parallax error, a cloud top of 5 kilometers in this location would be located over the crash site]. This was also seen about 2030Z. The brightness value of this cloud feature increases from 103 at 2000Z to 149 at 2015Z. After 2045Z, more significant cloud development is seen in the area of the accident.

Looping of infrared images for 1915Z to 2015Z showed increasing radiative temperatures (decreasing cloud tops) in the area of the crash site. At 2030Z, a radiative temperature of 274 degrees Kelvin (K) was noted at about 7 nautical miles north-northeast of the accident site, which equated to a cloud top of about 15,300 feet. Decreasing radiative temperatures (increasing cloud tops) were seen at this location from 2015Z to 2030Z.

Weather Radar Data

The Velocity Azimuth Display (VAD) Wind Profile Image (image time 2038Z) from the WSR-88D Doppler Weather Radar at Reno, Nevada (KRGX), showed wind speeds of 5 knots or less from 9,000 feet to 15,000 feet for a time of 2009Z. Wind directions varied from southerly to easterly to northerly. KRGX is located about 49 nautical miles north-northeast of the accident site.

Base Reflectivity Images for a 0.5-degree elevation angle were reviewed for times of 2003Z, 2009Z, 2015Z, and 2020Z. The 2003Z and the 2015Z images show weak weather echoes (VIP Level 1) within about a 5-nautical-mile radius of the accident site. No weather echoes were shown within about a 5-nautical-mile radius of the accident site at 2009Z and 2020Z. At 2009Z, strong to extreme weather echoes (VIP Level 3 to VIP Level 6) are located from about 32 nautical miles south to about 55 nautical miles south of the accident site.

The accident site is located about 192 degrees at 49 nautical miles from KRGX. At an elevation angle of 0.5 degrees at a distance of 49 nautical miles, the radar beam center is about 12,585 feet. The beam width is about 4,937 feet.

Soaring Forecast

A Soaring Forecast (areas encompassing the eastern Sierra Nevada Mountains and western Nevada) is prepared by the Reno, Nevada, National Weather Service Forecast Office. This product is routinely prepared about 1430Z (0730 PDT) and distributed to the Reno Flight Service Station and the Reno National Weather Service World Wide Web page at <http://www.wrh.noaa.gov/reno/>. Excerpts from the Soaring Forecast issued on July 13, 1999, are as follows:

> Forecast Winds and Temperatures aloft:
> 10,000 feet; winds 330 degrees at 12 knots; temperature 14 degrees C.
> 12,000 feet; winds 340 degrees at 7 knots; temperature 9 degrees C.
> 14,000 feet; winds 350 degrees at 6 knots; temperature 3 degrees C.
> 16,000 feet; winds 030 degrees at 3 knots; temperature -3 degrees C.
>
> No wave was expected and no ridge lift was expected.
>
> Thermal Data:
> Trigger Temperature 86 degrees F / 30 degrees C Time 1700Z (1000 PDT).
> Maximum Temperature 99 degrees F/ 37 degrees C Time 2030Z (1330 PDT).
> Maximum Altitude 17,000 feet.
>
> Soaring Index .. Yesterday .. 1,337 feet per minute.
> Today .. 1,756 feet per minute.
>
> Afternoon Wind Forecast ... Minden .. West at 5 to 15 knots.
>
> Weather Synopsis:
> Upper level high pressure over Northern Nevada with weak upper level low pressure area over the central and Southern Sierra. Atmosphere unstable over Western Nevada and increasingly moist towards the Southern Sierra. Today will be hot and good thermal activity will result.
>
> Significant clouds/weather/winds/isothermal or inversion layers:
> Few to scattered clouds 20,000 to 25,000 feet ... after 2100Z isolated thunderstorms with light rain, scattered to broken clouds at 15,000 feet ... inversion 18,500 to 20,000 feet.

In-Flight Weather Advisories

There were no AIRMETs, SIGMETs, or Convective SIGMETs in effect for the time and area of the accident.

There were no Oakland Center Weather Advisories issued by the Oakland, California (ZOA) Center Weather Service Unit in effect for the time and area of the accident.

Witness Statements (Summarized Excerpts)

One accident witness took off from the Minden-Tahoe Airport at 1230 PDT. Fifteen to 20 minutes later, the accident glider came underneath him at 2,500 feet agl and joined the witness in the same thermal. The witness stated that they were both at 9,000 feet msl. He stated that they circled together. The witness attempted to contact the accident glider but there was no response. The witness stated that the accident glider left the thermal on an easterly heading. He remained in the thermal climbing to 11,000 feet with oxygen on. The witness on returning to the Minden-Tahoe Airport encountered turbulent conditions. He stated that he was on a westerly heading to the airport and a line of clouds was present. He stated that his airspeed was 80 knots, he looked away, and when he looked back the airspeed climbed to 120 knots. The witness noted that it was very turbulent and there were a lot of updrafts and downdrafts present. The glider was controllable but it was rough. Cumulus clouds were about 12,000 to 13,000 feet agl. The witness landed at the airport at 1335 PDT.

Another witness was towing gliders the day of the accident. He flew over the Minden-Tahoe Airport and observed what looked like a well-developed dust devil with a plume of debris approximately 500 to 600 feet. After arrival at the accident site, he recognized the accident glider and circled the area but did not see any survivors. The witness reported the weather at the time of the accident was choppy, vertical motion in the air, "strong day," "nothing out of the ordinary." He stated the weather was conducive to what glider pilots usually want to fly in. He was flying over the airport at the time of the accident.

A third witness had helped push out the accident glider. He stated that before the accident there were building cumulus clouds running in a line north and south on the east side of the Minden-Tahoe Airport that extended from Gardnerville to Carson City. When he arrived at the accident site, light rain was present. About 1 hour later, thunderstorms were in the area. The witness noted that the forecast was for a 1,700 feet per minute climb, which is unstable.

COMMUNICATIONS

Glider pilots who were aloft at the time of the accident did not hear any distress transmissions from the accident glider on any of the Common Traffic Advisory Frequencies used by gliders. No distress calls were monitored at any FAA or military air traffic facility.

FLIGHT RECORDERS

The glider was equipped with a Cambridge Aero Instruments global positioning satellite (GPS) Flight Recorder for documenting glider competition meets. According to the manufacturer, a small internally mounted keeper battery is connected to a memory chip on the circuit board. Flight recorder parameters can be programmed for any sampling rate from 2 seconds to 10 seconds. If it is programmed for a 4-second sampling rate, it will fill the storage memory in 10 to 12 hours of flight. Memory is first in, first out. It stores the last "N hours of flight data, depending on the sampling rate."

Data available from the flight recorder:

- Altitude
- Latitude and Longitude to 0.001 of a minute
- Heading to 1 degree of precision
- Calculated GPS altitude
- Calculates pressure altitude up to 30,000 feet
- Time in UTC to nearest second

Real-time airspeed is provided to the pilot; however, it does not record airspeed or vertical velocity for historical reference.

The recorder was recovered in the wreckage distribution path in an impact-damaged condition with distortion of the case evident. It was sent to the manufacturer's facility for read-out of the memory. The manufacturer reported that the keeper battery had become dislodged and no data was contained in the memory.

WRECKAGE AND IMPACT INFORMATION

Wreckage Distribution

The main wreckage of the airframe including the fuselage, inboard wing sections, tail, and aircraft systems were found approximately 4 statute miles from the center of the Minden-Tahoe Airport on a magnetic bearing of 108 degrees. The elevation at the main crash site was approximately 5,100 feet msl. About 26 feet of the outer wing sections (each side) and the mating flight control surfaces were found distributed up to a distance of 0.5 mile northeast from the main crash site. The fuselage mass impact created a crater that measured approximately 8 feet long by 6 feet wide and approximately 0.6 foot deep. The forward fuselage (cockpit) was found in and around the initial impact crater. The main wreckage (that is, fuselage aft of the cockpit, the inboard wing sections, and tail section) was found approximately 36 feet northwest of the impact crater. The remaining wreckage debris at the main crash site was scattered over an area approximately 125 feet in diameter. A GPS hand-held receiver was used to plot the positions of the crash site impact crater, the main wreckage, and the major wing and flight control sections that separated during the in-flight breakup (a detailed data listing of the GPS coordinates and a map plot of this data are contained in the docket for this accident). Those sections that had separated in flight and were located a distance from the main crash site were marked with sequential numbers (GPS Nos.1, 2, 3, etc.). No indication of fire damage was noted on any of the aircraft wreckage or debris.

Fuselage

The fuselage was extensively damaged by impact with the terrain. The cockpit section forward of the wing was destroyed and scattered along the initial impact crater. The remaining portion of the fuselage was found inverted and approximately 36 feet northwest of the impact crater. This section of the fuselage was fractured approximately 8 feet aft of the wing with its upper half crushed inward. Both engine/propeller pylon compartment doors were detached and located near the main wreckage site. There was no evidence of any fire and/or sooting damage.

Wings

The investigation revealed that the left and right wings had failed symmetrically, in overload, at two distinct locations common to the outboard wing sections. The outboard wing sections failed at approximately 11.2 feet inboard of the wing tip. The outboard wing sections also failed approximately 14.5 feet further inboard from the first failure (about 25.7 feet inboard of the wing tip). The inboard failures occurred about 4 feet outboard of the mating point between the inboard and outboard wing sections. These outer wing sections and their mating flight control surfaces were found up to a distance of 0.5 mile northeast from the main crash site. The remaining inboard wing sections and their mating flight control surfaces were found along with the main wreckage of the airframe at the crash site. The following describes the damage noted to the wing surfaces and associated flight controls.

Left Wing

The left wing outboard section (GPS No. 1) was fractured at approximately 11.2 feet (3.4 meters) inboard of the wing tip. The wing tip and the wing tip aileron were attached to the outboard wing section. The wing tip aileron was attached by three hinges with the outboard hinge separated at its hinge pin. All three hinge pins were deformed aft and upward with the center hinge lug of each attached to the aileron. The wing tip was cracked approximately 4 inches along its leading edge nose near the outer tip. Minor impact damage was noted near the trailing edge of the wing tip at its very outboard end.

No visible damage was noted to the upper surface or leading edge of the outboard wing section except near the fracture location (plane). The wing upper surface skin was peeled away from the upper surface, between the main spar and the leading edge of the wing, starting at the fracture plane and extending outboard approximately 16 inches. Near the fracture plane, the upper surface skin was peeled away from the sandwich (foam) core. Approximately 10 inches outboard from this location, the upper surface separation transitioned into an inner laminar separation (between plies) that extended the remaining 6 to 7 inches of upper surface skin separation. The chordwise fracture was coincident with a wing upper and lower surface foam core splice.

The lower surface skin of the outboard wing section was missing between the fracture location and 68 inches outboard of that location. The foam core was intact; however, the exposed main spar lower cap in this area was splintered and frayed. The lower skin outboard of this area was peeled from the foam core, main spar, and rear spar. Inner laminar separation between the skin plies was noted near the leading edge. No evidence of impact damage was noted on the lower surface, except at a point 57 inches from the fracture plane along the rear spar.

Five hinges used to mount the outboard aileron to the outboard wing section rear spar were found attached to the wing outboard of the wing fracture plane. The most inboard composite hinge assembly of the five was torqued aft and outboard such that it was no longer parallel with the trailing edge of the wing. The remaining hinges were basically undamaged. A portion of the outboard aileron remained attached to the second hinge assembly.

The left wing ground skid, mounted between the wing tip and outboard wing sections, was found intact and undamaged.

The left wing outboard aileron was found at a distance from the main wreckage in four major pieces, identified as GPS Nos. 30, 15, 16, and 14 (inboard to outboard respectively). The outboard aileron fractured coincident with the wing fracture at 11.2 feet inboard of the wing tip (approximately 12 inches from the inboard end of the aileron).

The male mating alignment block (coupling) used between the inboard section of the outboard aileron (GPS No. 30) and the outboard end of the center aileron was intact and undamaged. The inboard aileron composite hinge lug mounted to this section was pulled open.

(Note: No push/pull tube is attached to the outboard aileron. The outboard aileron is simply connected to the center aileron by the aforementioned mating block (coupling), and therefore, its operation controlled by the motion of the center aileron. Push/pull tubes are used to drive the inboard and center ailerons directly. The wing tip aileron is driven solely by a tab on its upper surface such that when the outboard aileron is deflected upwards the wing tip aileron is also deflected upwards. When the outboard aileron is deflected downwards, the wing tip aileron remains in its neutral position.)

GPS No. 15 was identified as the center portion of the outboard aileron and was found with two balance weights attached. The inboard end of the inboard weight was deformed from the midpoint aft. The balance weights measured approximately 0.5 inch in diameter, were approximately 16 inches in length, and were made from brass (bronze) rod. The mounting structures of both balance weights were pulled away from this aileron section along their entire length. One composite hinge/bushing assembly remained attached to this section of the outboard aileron and was pulled away from the aileron. The fracture surfaces on each end failed at 45 degrees (with respect to the aileron) on the lower surfaces and at 90 degrees to the aileron on the upper surface. The trailing edge was separated along its full span.

The aileron balance weights recovered were all of similar construction and were mounted to the respective flight control surfaces at their leading edge with a composite glass fiber enclosure (wrap) bonded to the front spar. The balance weights varied in length and diameter. The diameter of the balance weights outboard were less than those found attached to inboard sections.

GPS No. 16 was determined to be the inboard 22 inches and lower surface (skin) of GPS No. 14 (which was identified as the outboard section of the outboard aileron). This section separated along the leading and trailing edges of the aileron. A balance weight extended the full span and was found deformed upward on both ends, consistent with an upward wing deflection. The radius of curvature was measured to be roughly 80 inches. The ends of this section were coincident with two adjacent wing rear spar composite hinge locations.

GPS No. 14 measured 54 inches in length and was identified as the outermost section of the outboard aileron. The lower surface was fractured from GPS No. 16 at approximately 31 inches from its outboard end at a 45-degree angle. Three composite hinges were found attached to this aileron section. The two outboard hinges were intact with minor damage. However, the lower surface around the most remaining inboard hinge was fractured. Two balance weights were attached and intact between the outboard three hinges. Minor trailing edge impact damage was noted on the outer 4 inches of the aileron, and the upper surface was partially fractured at 90 degrees to the aileron at the most inboard hinge.

GPS No. 26 corresponded to the remaining section of the left wing that separated in-flight (this section mates with GPS No. 1). This section measured approximately 14.5 feet in length. The inboard fracture was located 4 feet outboard of the mating point between the inboard and outboard wing sections. The upper surface was fractured at approximately 90 degrees to the main spar. The main spar was fractured approximately 5 inches outboard of the upper and lower surface fractures. The upper surface skin was found separated from the point of fracture outboard approximately 20 inches and had failed at the foam core surface in some areas with inner laminar separations noted elsewhere. The upper surface skin was separated from the trailing edge at several locations and the inboard 16 inches of the rear spar was missing. There was minor damage noted on the wing leading edge surfaces. Localized compression buckling/cracking of the wing upper surface skin (plies) was noted at multiple locations along the wing upper surface in the chordwise direction. In addition, internal core failures and span wise delaminations were noted along the entire span of this wing section.

At the outboard fracture location (that is, the failure between GPS Nos. 1 and 26), the upper surface skin of GPS No. 26 was pulled away from the foam core along the trailing edge for a distance of approximately 12 inches. Delamination of the main spar upper cap was noted on the outboard 8 inches of the spar. The lower surface skin was missing from the fractured end to approximately 24 inches inboard, and between the trailing edge and the main spar. Minor impact damage was noted 8 feet from the inboard fracture plane. Impact damage was noted on the lower surface skin between the trailing edge and main spar on the outboard 54 inches of this wing section. The rear spar and lower surface skin were completely missing aft of the main spar along the outboard 17 inches of this section. The main spar lower cap was separated and splintered for approximately 12 inches on the outboard end.

Eight aileron hinge attachment points were located on this section of the wing. The most inboard and most outboard hinges were completely separated from the rear spar. The second, third, fourth, and fifth hinges (from the inboard end) were intact and undamaged. However, the lower surface skin was slightly pulled away from the rear spar surrounding the second and fourth hinge locations. The composite support for the sixth hinge was cracked, delaminated, and displaced. The seventh hinge composite support was cracked on the hinge pin lug in the longitudinal direction, with the hinge pin bent aft.

The three inboard hinge locations found on this section of the wing mate with the wing inboard aileron. The remaining (5) hinges found on this section mate with the wing center aileron. The most outboard hinge (No. 8), located 16 inches inboard of the outboard fracture plane, corresponds to the center aileron outboard hinge point /actuator.

The left wing center aileron was found in three separate sections. The outboard section was identified as GPS No. 2 and measured approximately 23 inches in length. This section of the center aileron was found with its outboard hinge (identified as No. 8 above), actuator rod end, and portions of the lower wing skin still attached. The aileron was fractured in the chordwise direction (that is, perpendicular to its span) just outboard of hinge No. 7 (identified above). The aileron upper and lower surface skins were separated from each other along 75 percent of the trailing edge of this section. The actuator rod end that remained connected to the aileron was bent in the outboard direction. A surface balance weight was found attached to this section of the center aileron. However, the balance weight composite attachment structure was found partially separated from the leading edge between the inboard end and a point approximately 13 inches further outboard. The inboard 6 inches of the leading edge balance weight was deformed in the forward direction.

The center section of the three left wing center aileron pieces recovered measured approximately 42 inches in length. The outboard end of this section mated with GPS No. 2. The inboard end of this section was fractured in the chordwise direction on the upper surface. The lower surface was fractured at approximately 45 degrees (trailing edge outboard) immediately outboard of hinge No. 5 (location identified above). The trailing edge was separated along the full length of this aileron section. The upper surface skin was separated from the leading edge spar along its entire length. Hinge Nos. 6 and 7 were found basically intact with minor damage to hinge No. 6. The inboard balance weight that was normally installed in this area was missing as evidenced by the separation of its composite attachment structure from the front spar. The outboard balance weight was intact with minor impact damage noted along its leading edge.

The inboard piece of the left wing center aileron that was recovered was identified as GPS No. 7 and measured approximately 31 inches in length. The upper and lower surface skins were separated 5 inches along the trailing edge near the outboard end of this section. The upper and lower surface skins were undamaged. The composite hinge lugs (hoops) at hinge Nos. 4 and 5 (location identified above) were pulled open. The inboard section balance weight, approximately 20 inches long, remained attached between hinge Nos. 4 and 5.

The left wing inboard aileron was recovered in three separate sections. The outer two sections were recovered at a distance from the main wreckage. The outboard section, identified as GPS No. 37**,** was fractured approximately 67 inches from the outboard end of the aileron, that is, at the same wing station where the inboard (in-flight) separation of the left wing occurred. The upper surface was fractured at 45 degrees (trailing edge outboard) and the lower surface in the chordwise direction. The upper and lower surface skins remained attached to the upper and lower surfaces shells. However, the lower surface skin was separated from the front spar approximately 10 inches near the outboard end of the aileron. Likewise, trailing edge separation measuring approximately 11 inches was noted near the inboard end of this aileron section. The composite hinge lugs at hinge Nos. 2 and 3 (location identified above) were pulled open. The hinge pin attached at hinge No. 1 was bent in the forward direction on both ends, and one of the two mating wing hinge lugs at this location remained partially attached to the hinge pin. Approximately 6 inches of the inboard aileron actuator and its rod end (located at hinge No. 1) remained attached to this section of the aileron; however, it was severely deformed.

The center section of the left wing inboard aileron, which was recovered at a distance from the main wreckage, was identified as GPS No. 36. This section of the aileron measured approximately 28 inches in length and was separated from the remaining aileron approximately 22 inches from the inboard end of the aileron. The fracture on the upper surface was in the chordwise direction and on the lower surface at 45 degrees (trailing edge inboard). Trailing edge separation was noted along two-thirds of the span of this aileron section. One hinge lug was found intact approximately 21 inches from the inboard end of this section of the aileron.

The most inboard section of the left wing inboard aileron was recovered with the main wreckage and experienced severe impact damage on the upper surface and leading edge. The inboard hinge lug was found intact. The upper surface skin was totally separated from the front spar. The lower surface skin was approximately 50 percent separated from the front spar. The aileron skins were completely separated along their trailing edge. (Note: No factory balance weights are installed on the inboard aileron.)

The inboard section of the left wing was found at the crash site in two major pieces along with the inboard 4 feet of the outboard section of the wing. The wing assembly (retention) pin used to join the inboard and outboard wing sections was found intact.

The inboard wing section was fractured in half approximately 6 feet inboard of the assembly pin noted above. The full span of the inboard wing was destroyed forward of the main spar due to impact damage. Much of the lower skin was missing. The upper skin was completely separated from the main spar and was also separated from the rear spar at numerous locations. The inboard wing main spar (torque box) was heavily damaged on its upper and lower surfaces and was fractured at numerous locations immediately inboard of the wing mating point assembly pin. The rear spar was fractured and separated from the remaining lower surface skin in several locations. A portion of the rear spar and its flap attachment hinges were found in several pieces adjacent to the main wreckage. The inboard four flap hinges were basically intact.

The wing center section carry through spar, aft spar block, exhibited heavy impact damage on its rear face (surfaces). The impact damage to the rear face of the spar block measured approximately 10 inches wide by 3 inches in height and was crushed forward approximately 1 inch at the centerline of the aircraft. The spar block bushings on the left and right wings were intact and the left to right wing assembly (retention) pin was found installed (intact) at the crash site. The left wing rear spar-to-fuselage engagement (alignment) pin was bent forward approximately 30 degrees, identical to the damage noted on the right wing rear spar-to-fuselage engagement pin.

The 4-foot section of the outboard wing that remained attached to the inboard wing was also destroyed. The upper skin was missing. The majority of the lower skin was separated from the main spar. The outboard 24 inches of the upper spar cap was missing. The entire upper spar cap that remained was frayed and splintered. The main spar (box) attachment lugs at the inboard end of this section were severely damaged, and the main spar rear attachment lug bushing was missing.

The left wing flap was found in several pieces at the crash site but separated from the inboard section of the wing. The flap actuator was fractured at its threaded rod end, normally attached to the bell crank assembly at the rear spar of the wing. However, the remaining portion of the actuator (tube) remained attached to the flap. The actuator was found bent approximately 30 degrees several inches forward of the leading edge of the flap.

The left wing airbrake, (a double panel Schempp-Hirth type airbrake) mounted near the outboard end of the inboard wing section, was found in its fully extended position. The lower (red) metal panel of the airbrake was bowed forward and crippled near a cutout for the inboard deployment (pivot) arm. The upper (composite) panel of the airbrake was heavily damaged, yet remained partially attached to the airbrake. There was evidence of red paint transfer on the upper surface of the wing forward of the slot (opening) for the airbrake. The composite cover that fairs the opening for the airbrake with the upper surface of the wing (when the airbrake is stowed) was found adjacent to the wing with minor impact and cracking damage. The airbrake assembly and its pivot arms could only be rotated freely approximately 20 degrees in the clockwise direction (view looking forward) due to the surrounding wing damage. Normal operation (during closing) requires that the airbrake assembly be rotated approximately 90 degrees to completely stow the airbrake. (Refer to Section 4.1.3 of the Airworthiness Group Chairman's Factual report, which is contained in the docket for this accident, for a description of the operation of the airbrakes.) Although the airbrake push rod, rod ends, and attachment linkages were bent in compression, at several locations, the linkages were found intact.

Right Wing

The right wing tip, identified as GPS No. 34, was found separated from the outboard section of the wing. The upper and lower surface skins of this piece were intact. However, an 8-inch crack was noted along the leading edge approximately 11 inches outboard of where the wing tip mates with the outboard wing. Additional cracking damage was noted along the outboard trailing edges of the wing tip.

The wing tip aileron was found intact and attached at all three of its hinges. However, the aileron upper surface skin was separated along the entire span of its rear spar. The lower surface skin was partially separated. The wing tip (cylindrical) carbon fiber main spar was found fractured at its interface with the outboard wing section. The wing tip inboard close-out rib was cracked, in the chordwise direction, between the main spar and its trailing edge.

The right wing ground skid (identified as GPS No. 32A) was recovered near the outboard section of the right wing that was later identified as GPS No. 32.

Approximately 8.5 feet of the outboard wing section that normally mates with the right wing tip (GPS No. 34) was located at a distance from the main crash site and was identified as GPS No. 32. The fracture at the inboard end of this section occurred at the same location noted on the left wing, that is, 11.2 feet inboard of the tip of the wing, coincident with a sandwich core splice forward of the main spar. The wing fracture plane was oriented at 45 degrees to the span of the wing, that is, extended from the leading edge aft and inboard (when viewed from above).

The upper surface skin had separated from the sandwich core outboard of the wing fracture location noted above. Approximately 80 percent of the upper surface skin was missing. The core material itself had failed just beneath the bond-line to the outer surface plies. Two pieces of the upper surface skin, that were later determined to be from this section of the outboard wing, were recovered and identified as GPS No. 22 and GPS No. 24.

The outboard aileron remained attached to this section of the wing (GPS No. 32) at six aileron hinge locations. Minor impact damage was noted to the lower surfaces of the aileron. The lower surface wing skin was fractured at three of its aileron hinge locations. The lower wing skin was also separated from its rear spar in the vicinity of the wing separation, that is, at the inboard end of this section of the wing.

The remaining section of the right wing that separated in flight was recovered and identified as GPS No. 28. This section of the wing measured approximately 14.5 feet in length. The inboard fracture was located 4 feet outboard of the mating point between the inboard and outboard wing sections, that is, at the same failure location noted on the left wing. The upper surface was fractured at approximately 90 degrees to the main spar. Approximately 70 percent of the upper surface skin was found missing.

Several sections of the remaining upper surface skin (plies) were found curled up and inboard. The separated plies had peeled away from their inner plies causing an inner laminar separation. The exposed surfaces of the separated plies appeared uniform in resin quality, and without visible voids. A section of the upper surface skin (approximately 7 inches wide) was intentionally peeled back (inboard) approximately 1 inch to compare its underlying surfaces with the in-flight failure surfaces. The surfaces appeared identical. Although no spring scale was available during this test, the force required to peel the outer plies from the upper surface was consistent with this type of construction, that is, involving plies of carbon fiber fabric.

Near the outboard end of this section of the wing (GPS No. 28), the stacking sequence of the sandwich construction (that is, carbon fiber fabric/foam core) consisted of an outer surface unidirectional ply, oriented in the spanwise direction (that is, 0 degrees with respect to the wing), a plus/minus 45-degree plain weave, the foam core, followed by another plus/minus 45-degree plain weave bonded to the inner surface of the wing. The fabric plies bonded to either side of the upper surface foam core appeared to be of 12.5 x 12.5 tows/per inch in the warp and weft directions respectively.

Approximately 48 inches inboard of the outer wing failure, at a point were the outer upper surface plies were peeled away due to the in-flight breakup, the sandwich construction consisted of an outer 0-degree unidirectional ply, followed by a plus/minus 45-degree plain weave, followed by a 0/90-degree plain weave, followed by another plus/minus 45-degree plain weave, followed by the core. The inner skin was not visible at this cross section.

Two pieces of wing skin were recovered separately and later determined to be the missing skin from the upper surface of GPS No. 32. The first piece of skin extended nearly the full span of this wing section forward of the main spar. The second piece, identified as GPS No. 8, extended the full span of this wing section aft of the main spar.

The center aileron was recovered at a distance from the main wreckage and was identified as GPS No. 31. The aileron actuator rod end remained attached and was bent at several locations. Three of four balance weights were missing. Trailing edge separation was noted along 60 percent of the aileron. Localized compressive buckling failures were noted on the upper surface of the right center aileron at several locations. Five hinge locations were identified. The inboard hinge pin was bent outboard approximately 100 degrees. Fractures were noted on the lower surface of the aileron at several of its hinge locations.

A portion of the right inboard aileron that had separated in flight was recovered and identified as GPS No. 3. The fracture at the inboard end of this section of the aileron occurred at the same location as that of the inboard in-flight wing separation. The outboard end of this section had separated just inboard of the aileron actuator, that is, approximately 2 feet further outboard.

GPS No. 29 was identified as the lower surface of the right inboard aileron, which also separated in flight, and extended from the outboard end of the aileron to the location of the inboard wing fracture. The remaining outboard sections of the inboard aileron remained attached to the right wing (GPS No. 28).

GPS No. 33 was identified as the remaining (inboard) section of the inboard aileron that measured approximately 4 feet in length. This section of the aileron extended inboard of the (inboard) in-flight wing separation noted above and was found at a distance from the main wreckage. The hinges on this section of the aileron remained intact and were essentially undamaged.

GPS Nos. 9, 21, 15B, and 35 were identified as miscellaneous pieces of wing skin believed to be from the right wing.

The remains of the right inboard wing were found at the crash site along with 4 feet of the outboard wing section that remained attached. The upper surface (skin) of the 4-foot section of the outboard wing had separated in flight and was recovered at a distance from the crash site. This piece of wing skin was later identified as GPS No. 6C. The wing assembly pin between the inboard and outboard wing sections was found in place; however, its safety was compromised and the assembly pin forced forward approximately 0.75 inch.

The full span of the inboard wing was destroyed and missing everything forward of the main spar. Large sections of the lower surface skin were missing. The upper surface skin was separated from the rear spar and main spar at numerous locations. Complete sections of the upper surface skin aft of the main spar were missing inboard of the right wing airbrake. The main spar torque box structure was heavily damaged along its entire span.

The right wing airbrake was found in its fully extended position. The composite cover that fairs the opening for the airbrake with the upper surface of the wing (when the airbrake is stowed) was found adjacent to the wing with minor impact damage. The upper (composite) panel of the airbrake was heavily damaged, yet remained partially attached to the airbrake. The lower (red) metal panel of the airbrake was bowed forward between the airbrake pivot/deployment arms, with red paint transfer noted on the upper surface of the wing at several locations beneath the airbrake. The airbrake control push rod, rod ends, and attachment linkages were deformed at several locations in compression. The airbrake and its pivot arms rotated approximately 20 degrees freely in the counterclockwise (closing) direction (view looking forward). The upper surface wing skin was fractured immediately forward of both airbrake pivot/deployment arms, with orange paint transfer noted on the upper surface skin surrounding the deployment arms (painted orange in color).

The outboard two-thirds of the right wing flap and the flap actuator remained attached to the inboard wing. The inboard section of the flap, which measured approximately 50 inches in length and had separated from the wing, was found nearby the main wreckage. The separation (fracture) of the flap noted above was at 90 degrees with respect to the span of the flap. The rear spar of the wing was heavily damaged inboard of the flap actuator. Several flap attachment hinges were missing. The remaining flap hinges were damaged. The flap actuator had separated at its threaded rod end located at the bell crank assembly.

Empennage

The horizontal stabilizer was separated from the vertical stabilizer approximately 18 inches below its surface due to a complete through fracture of the vertical stabilizer. The horizontal stabilizer was found immediately adjacent to the remaining tail wreckage. The elevators were attached only at their inboard hinge attachments. The upper surface skin of both elevators was pulled away from the rear spar of each elevator. Little other damage was noted.

The vertical stabilizer was separated from the fuselage approximately 1 foot forward of the tail section. The leading edge had considerable impact damage. The rudder was basically attached to the tail. The lower hinge attachment was intact and undamaged. However, the upper hinge attachment along with approximately 16 inches of the rear spar had separated from the remainder of the vertical stabilizer. The rudder rotated freely about its hinges.

Ailerons

Push/pull tubes are used to drive the inboard and center ailerons directly. No push/pull tube is attached to the outboard aileron. The outboard aileron is physically connected to the center aileron with a mating block and is therefore controlled by the motion of the center aileron.

The wing tip ailerons, which are used to minimize the effects of adverse yaw, are driven by a tab on their upper surface that overlaps the outboard aileron. When the outboard aileron is deflected upwards, the wing tip aileron is also deflected upwards. However, when the outboard aileron is deflected downwards, the wing tip aileron remains in its neutral position, since it is spring-loaded to do so.

Left Wing Ailerons

The left wing aileron (inboard) push/pull tube assembly was found in one piece between the cockpit and its outboard end, where a ball/socket (joint) (L'Hotellier fitting) is used to connect the push/pull tubes between the inboard and outboard wing sections. This section of the push/pull tube remained only partially attached to the wing and was found bent in half at its midspan due to the extensive breakup of the inboard wing section.

The section of left aileron push/pull tube that extended from the ball/socket joint (at the wing mating point) to the inboard aileron actuator bell crank assembly, had fractured just outboard of the ball and socket assembly (L'Hotellier fitting). Although the female socket (attached to the outer push/pull tube) remained attached to its mating ball assembly (attached to the inboard push/pull tube), approximately 7 inches of the aileron push/pull tube immediately outboard of the ball/socket assembly fracture was missing. The remaining sections of the left aileron push/pull tube assembly were found intact and remained attached to the most inboard section of the left wing that had separated in flight, that is, GPS No. 26.

The inboard aileron actuator was found bent and fractured through its rod end eyebolt, which was attached to the inboard aileron bell crank assembly. The bell crank and its push/pull clevis attachment fitting were also deformed. However, the bell crank assembly operated freely. Similar findings were noted at the center aileron actuator and bell crank assemblies.

Right Wing Ailerons

The right wing aileron (inboard) push/pull tube assembly was found intact between the cockpit and its outboard end, where a ball/socket (joint) (L'Hotellier fitting) is used to connect the push/pull tubes between the inboard and outboard wing sections.

The push/pull tube that extended from the ball/socket joint (at the wing mating point) to the inboard aileron actuator bell crank assembly, and measured approximately 67 inches in length, was recovered at a distance from the crash site and was identified as GPS No. 25. An examination of the female socket (receptacle), installed on the inboard end of this segment of the push/pull tube, revealed that the socket assembly had spread open allowing it to separate from the inboard push/pull tube mating ball assembly. The opposite end of this section of the push/pull tube was fractured at its clevis fitting attachment to the inboard aileron actuator bell crank assembly and was severely bent.

The remaining section of the right aileron push/pull tube that extends between the inboard and center aileron actuators was recovered with the section of the outboard wing identified as GPS No. 28. The inboard aileron actuator was found intact, but slightly bent. The center aileron actuator was fractured through its rod end eyebolt attached to the bell crank assembly.

Flap/Airbrake Operation

The flap push/pull tubes are displaced outboard to deploy the flaps. The airbrake push/pull tubes are displaced inboard to deploy the airbrakes. When the airbrakes are deployed, the airbrake/flap interconnect push/pull tube (slave arm) translates inboard with the airbrake push/pull tube and shifts the flap pivot point aft about the flap/airbrake assembly main spar pivot fitting and thereby extends the flaps from their present position (that is, the flaps are mechanically extended whenever the airbrakes are deployed). The airbrakes rotate vertically up and outboard when they are deployed.

Left Flap

Minor bending of the left flap push/pull tube was noted between the cockpit and its actuator. Continuity of the left flap torque tube was established between the cockpit and the flap actuator bell crank assembly; however, the torque tube was deformed and fractured at the bell crank assembly. The torque tube for the left wing flap exhibited compressive bending along its full length.

Right Flap

The inboard 7 feet of the right flap push/pull tube was severely bent in a sinusoidal shape between the cockpit and its actuator. The right flap actuator arm was found intact in length; however, it was buckled about 90 degrees upwards midway between the flap and the bell crank assembly. Complete continuity of the right flap torque tube system was established.

Airbrakes

The airbrakes are double-panel bayonet-type panels that are mounted aft of the main spar, near the outboard end of each inboard wing section. The airbrakes consist of an upper composite panel and a lower red metal panel that when deployed disrupt the flow of air over the wings thereby inducing drag. When stowed, the airbrakes are hidden beneath the surface of the wings by a composite cover that fairs the slot opening for the airbrake with the upper surface of the wing. During deployment the airbrakes extend above the surface of the wing by rotating up and outboard via two deployment (pivot) arms controlled by push/pull tubes that are connected to the cockpit controls. The airbrake push/pull tubes are pulled inboard to deploy the airbrakes.

The forward operating handle for the airbrakes and approximately 12 inches of the associated airbrake control linkage was found in the wreckage debris near the impact crater. This section of the control linkage consists of a hollow tube sleeve that slides over a smaller diameter brass guide rod with its forward end affixed to the adjacent airframe structure. The airbrake cockpit operating handles (blue) are attached to the control linkage outer sleeve and allow either pilot to operate the airbrakes by moving their handle fore and aft. When either pilot positions his/her handle (that is, control linkage) forward, the airbrakes are stowed; likewise, when either handle is positioned aft, the airbrakes are extended. The control linkage and brass rod assembly were found with a short length of the guide rod exposed and deformed (that is, bent) such that the two were fixed in that relative position. The exposed section of the brass rod measured 2.125 inches from the center of its mount to the nearest end of the control linkage. The position of the control linkage was compared to an exemplar Nimbus-4DM installation to determine the position of the cockpit airbrake control as found (note that extensive longitudinal collapse of the cockpit occurred during the ground impact sequence). The investigation revealed that the airbrakes on this model are stowed and locked when none of the guide rod is exposed, are stowed but unlocked when 2.125 inches of the guide rod is exposed, and are fully deployed when 7.5 inches of the guide rod is exposed.

The left airbrake push/pull (torque) tube was intact between the cockpit and the airbrake. Minor bending of the airbrake push/pull tube between the cockpit and the left airbrake was noted at several locations. The airbrake torque tube, rod ends, and associated clevis linkages had suffered compressive bending loads along their entire length. The push/pull tube located between the inboard deployment arm and its push/pull tube connection near the flap actuator was slightly bent in a sinusoidal fashion. However, the push/pull tube located between the two actuator deployment arms was found essentially straight. The spar mounting structure was destroyed surrounding each of the left airbrake deployment arm pivot attachments, and the arms were found torqued about their normal mounting. The respective push/pull tube clevis fittings mounted to each of these pivot points were also deformed due to the torsional separation of the deployment arms from their mating structure.

The right airbrake torque tubes, rod ends, and associated clevis linkages were found bent and exhibited compressive bending. Clevis fitting deformation similar to that found on the left side was noted at each of the airbrake deployment arms. Deformation of the push/pull tubes in the forward direction was noted, with the most severe damage witnessed just inboard of the flap/airbrake actuator bell crank assembly.

No spanwise witness marks were noted adjacent to any of the airbrake surfaces or its deployment arms. However, red paint transfer was noted in distinct chordwise directional stripes on the upper surface of the wings forward of the red airbrake plates (the airbrake plates are painted red, and the red color only shows above the wing skin surface when the airbrakes are deployed). Orange paint transfer was also noted immediately forward of the inboard deployment arm (the inboard and outboard actuator arms are painted orange while all the push/pull torque tubes are painted gray) on the airbrake wing cutout. The upper surface skin was found fractured immediately forward of the outboard deployment arm with the skin completely surrounding the deployment arm. Indentations were noted in the structure forward of both deployment arm clevis attachment fittings; however, no punctures or chafing damage was noted in these locations along a circular path (that is, in the spanwise direction) consistent with the upward and outboard (clockwise rotation) deployment of the airbrake assembly, with no clearance between itself and the surrounding structure.

Elevators

The elevator push/pull tubes between the cockpit and the elevators were accounted for. The elevator control system interlocking pins were found withdrawn from the elevator torque actuator. The elevator torque tube was found intact and attached to both elevators. The elevator torque tube was fractured at the threaded rod end adjacent to the tail fracture and at all threaded rod ends between the cockpit and that point. The push/pull tubes and all rod ends along the entire fuselage exhibited severe compression bending/buckling. A small section of the elevator control system linkage that connects to the elevator trim system was not recovered. The adjacent section of linkage (rod) that was recovered and had fractured was sent to the NTSB metallurgical laboratory for detailed examination (See TESTS AND RESEARCH section of this narrative). Loose rod ends were noted on the majority of the elevator push/pull tube connections. Several lock washers were not crushed and uneven rod end thread protrusion was noted on several connections.

Rudder

Rudder control cable continuity was established between the cockpit foot pedals located at each seat and the rudder.

Electrical

No evidence was found that would indicate that any of the electrical wiring had been burned and/or sooted. The main battery was destroyed during the crash; however, no evidence of thermal damage or sooting was noted on the battery case. Although minor deposits of a soot-like color were noted on several of the battery plates, no soot was noted on any of the battery cell separators.

Instruments

The cockpit instrumentation (front and rear panels) was destroyed.

Landing Gear

The main landing gear (wheel) is retractable and features a hydraulic disc brake; the nose and tail wheels are fixed. The nose wheel was found in the main wreckage. The main wheel was found in its retracted position. The tail wheel remained attached to the fuselage and was undamaged.

Fuel

The left wing fuel bladder was found at the main wreckage site. Markings on the bladder indicated that its capacity was 20 liters. The bladder was hydraulically ruptured and torn completely open on one end. Several tears and lacerations were noted elsewhere on the bladder. The bladder smelled of fuel; however, the bladder was empty.

The right wing fuel bladder was found in similar condition to that of the left. The right wing bladder was empty.

Oxygen

Two oxygen bottles were found near the impact crater with their supply valves open and their nozzles broken off. Both oxygen bottles were empty. Based upon interviews with the sailplane owner's son, these oxygen bottles were used each flight and had been used without any problems the day before the accident.

Powerplant

The Rotax engine and engine mount were found adjacent to the fuselage tail near the main wreckage. The propeller was found in its stowed position. The propeller mast (pylon) jackscrew was found in the nearby wreckage and was compared to an exemplar Nimbus-4DM installation. The jackscrew measured 11.5 inches from its pivot bolt to the nearest end of the outer cylinder corresponding to its fully extended position and a retracted pylon position. The jackscrew drive mechanism was found separated (fractured) from the jackscrew at approximately 22.5 inches from its pivot bolt.

No leading edge damage or chordwise scoring was noted on either of the propeller blades. Spanwise gouges were noted in one of the two propeller blades from its tip inward approximately 10 inches. The opposite blade was delaminated in an area 9 to 16 inches from the tip section.

MEDICAL AND PATHOLOGICAL INFORMATION

Autopsies were conducted on the pilot and passenger by the Douglas County Coroner's office on July 14, 1999, with specimens retained for toxicological analysis by the FAA Civil Aeromedical Institute. According to the attesting pathologist who performed the autopsies, the cause of death for both the pilot and passenger were blunt-force trauma, without contributing preexisting conditions. Neither occupant exhibited anatomical injury patterns consistent with control manipulation at impact.

The results of the toxicological tests on both occupants were negative for alcohol and all screened drug substances.

SURVIVAL ASPECTS

Canopy

The cockpit canopy and its frame were found near the impact crater and were both destroyed. The composite frame was fractured in several locations. The Plexiglas canopy was shattered. No evidence of soot was noted on any of the canopy surfaces. Both canopy latching handles were found attached to their interconnecting operating arm (linkage). The linkage was severely bent and exhibited compression buckling; however, all three canopy locking pins (attached to the operating linkage) were intact and undamaged. The surface of the locking pins appeared tarnished from normal use. Two of the three mating fuselage frame locking pin posts were found attached to small sections of adjacent cockpit structure. The recovered posts were undamaged; however, they were dislodged (that is, pulled away) from their normal positions (The knurled base of the locking pin posts are merely potted (adhered) to the fuselage composite structure surrounding them.)

Seatbelt Restraints

A four-point harness assembly consisting of two lap belts and two shoulder straps is attached to the seat framework of both the front and rear seats. The four straps at each harness connect to a standard five-point center-locking mechanism (buckle).

Front Seat

Half of the webbing of the right side upper shoulder strap failed in tension 10 to 13 inches from its upper anchor point. The remaining cross section of the strap was distorted (stretched). The same strap was failed in tension at its shoulder adjustment buckle. The lower shoulder strap, which extends between the shoulder adjustment buckle and the tang that mates with the five-point harness buckle, was found distorted and stretched. The lower strap was pulled completely out to its stop stitching. The stitching at the tang was found distorted in tension and/or completely failed. The tang locked properly into the five-point buckle; however, several striations (marks) and a small indentation were noted on the end of the tang adjacent to the five-point buckle.

The left side shoulder strap was failed in tension at 10 inches from its upper anchor point. The harness webbing and stitching on the upper strap (that is, above the shoulder adjustment buckle) was distorted and stretched. The lower strap was stretched and distorted at the adjustment buckle. The lower strap webbing and stitching was also stretched and distorted near its tang. The lower strap tang exhibited the same indentation as the right tang.

The left and right side seat belts were pulled free from their respective fuselage mounting points. Both seat belt straps were stretched and distorted near their adjustment buckle rollers. The five-point harness buckle was found attached to the right seat belt. The locking tangs of both seat belts were indented similar to the shoulder harnesses described above.

Rear Seat

The left side shoulder harness was stretched and distorted on the upper and lower webs along their entire length. The lower strap had failed in tension at the shoulder adjustment buckle attachment. Localized stitching failures were prevalent throughout the harness. Evidence of loading on the lower strap tang was noted similar to that of the front seat.

The right side shoulder harness was stretched and distorted similar to the left side, although less severely. Localized stitching failures were noted in several areas. The lower strap tang was damaged similar to the left side shoulder harness.

The left and right seat belts were pulled away from the fuselage attach points. The right seat belt webbing had failed at its attachment. A small section of structure remained attached to the left belt surrounding its attachment. Localized web stretching and distortion were noted on the left belt around its adjustment buckle roller. Localized web stretching, distortion, and stitching failure were noted near the tang end of the belt. The right belt webbing was stretched and distorted at its adjustment buckle. Failure of the right side belt web and stitching was noted near its fuselage structural attachment. Evidence of loading was noted on the right belt tang but not on the left belt tang.

Parachutes

No deployment of parachutes was observed by any witness.

Two parachutes were found at the crash site along with the main wreckage and in close proximity to the occupants. The front seat parachute was partially extended out of the pack; however, the rear seat parachute was found packed. The straps on each parachute were basically undamaged. Both parachutes had been repacked in May 1999.

Emergency Egress Substantiation

According to correspondence from the LBA, proof of compliance with JAR 22.807, Amendment 22/90/1, was based on the substantiation report of the Duo-Discus glider, which has the same cockpit design as the Nimbus-4 DM. For the jettisoning of the canopy, the report refers to the research project "Emergency Canopy Jettison" performed at the FH Aachen facility. The recommendations of this report were adopted by Schempp-Hirth and introduced into the design of the Duo-Discus/Nimbus-4DM. No in-flight tests were carried out.

During the flight evaluation of the Duo-Discus by the LBA, the emergency exit procedure was repeatedly performed (on the ground) by the LBA pilots. This resulted in the recommendation to install a different harness system and to apply an anti-skid rubber coating to the floor of the rear cockpit. According to correspondence from the LBA, these recommendations were accepted by Schempp-Hirth and were introduced into the serial production of the Nimbus-4DM and incorporated into the accident glider.

TESTS AND RESEARCH

Investigation and Analysis of Composite Material Failures in the Wing Structure

Following initial on-site documentation of the wreckage, it was moved to a hangar at the Minden-Tahoe Airport where it was schematically reconstructed for a more detailed examination of the systems and the fracture points on the wing structures. After the detailed documentation was completed, the wings were sent to Wichita State University's National Institute for Aviation Research (NIAR) for a complete scientific investigation and failure analysis of the composite structures. A secondary objective was to determine if any manufacturing or service degradation anomalies were present in the composite structure, and, if any were found, determine if they could have contributed to failure of the wings. NIAR is the composite materials contract laboratory for the FAA's Small Aircraft Directorate of the Aircraft Certification Office. This research was performed with FAA funding at the request of Safety Board investigators and in support of this accident investigation. The glider manufacturer cooperated in this investigation by supplying production drawings, process specifications, and detailed explanations in response to questions where necessary. The LBA also cooperated in this study by providing certification data, background information on the certification process, and certification test reports. (The complete NIAR report (NIAR Report Number 02-01, "Investigation of a Schempp-Hirth

Nimbus-4DM Motor Glider In-flight Breakup") is attached to the Airworthiness Group Chairman's Factual report, which is contained in the docket for this accident.)

Throughout the following section of this narrative (1.10.1), the analytical and conclusionary statements appearing herein were taken directly from the above-cited report.

The investigation focused primarily on the left and right wings, which are primarily constructed with carbon fiber, glass fiber, epoxy resin, and foam core, and are manufactured using wet-layup process. In addition to the wings from the accident glider, the Spanish investigative authority submitted comparable samples for testing from the in-flight breakup accident in Spain (See section 1.3.1.2 of this narrative). In terms of protocol, five tasks were identified in the statement of work for the NIAR investigation:

Task 1: Investigation of Material Properties
Task 2: Investigation of Wing Structure
Task 3: Wing Spar Testing
Task 4: Wing Skin (Sandwich) Testing
Task 5: Failure Analysis

The full-span wing consists of two inboard sections, two outboard sections, and two wing tip sections. The left and right inboard sections mate at the center of the fuselage. The inboard and outboard sections are joined at 3.8 meters from the center of the fuselage. The outboard and wing tip sections mate at approximately 12 meters from the center of the fuselage. Each wing tip section is approximately 1.2 meters long.

The inboard sections consist of a rectangular shape main spar. The outboard sections have an I-beam-shaped main spar. The spars change from a rectangular to I-beam shape in the outboard sections near the area where the outboard sections mate with the inboard sections. The spar caps are primarily made of unidirectional carbon/graphite epoxy material and the shear webs are made of plus/minus 45-degree fiberglass. The skins panels are sandwich structures with a foam core and carbon fabric epoxy face-sheets. The skin panels include layers of 0/90-degree and plus/minus 45-degree fabric to resist twisting (that is, a "torque box").

All the composite material used in the construction of the glider is certified by the LBA. The wings broke into several pieces in many locations; six of which involved failures of spars. The six failure locations were identified as AA, BB, CC, DD, EE, and FF for labeling purposes during the test process and in the final report. Eyewitness accounts and on-site wreckage examination confirmed that the failure locations CC, DD, EE, and FF occurred in midair. Information about the composite material qualification process is provided in appendix A7 of the NIAR report. This section was written by LBA. (Information about the manufacturing process of a Nimbus-4DM wing is contained in appendix A8 of the NIAR report and was written by Schempp-Hirth)

Task 1: Investigation of Material Properties

The objective of Task 1 was to investigate the basic physical and thermal properties of the wing structure composite materials, and sample coupons were taken near the six primary failure locations for this analysis. Schempp-Hirth provided a sample of their baseline material for comparison purposes. The baseline panel was obtained from the inboard wing section of a Nimbus-4DM, serial number 2, a factory-owned conforming article used in certification testing. The baseline panel measured approximately 190 millimeters by 292 millimeters. While the methodology and test protocols used in Task 1 are fully delineated in the NIAR report, in pertinent part the process used Dynamic Mechanical Analysis (DMA), Differential Scanning Calorimetry (DSC), and examinations for resin and void content in the composite matrix.

The results for this task are presented in appendix A6 of the NIAR report. The resin content and void content results in appendix A6 are shown in graphical format in figures 13 and 14 of the NIAR report. Since the spar caps were not made using the same process as the other parts of the wings and were not expected to have equivalent properties, the graphical results were separated accordingly. The average resin content of the spar caps is lower than that of the wing skin and other structural samples, since the spar caps are made of unidirectional foam.

The spar caps from the Minden and Spain accidents were found to have an average resin content of 33.3 percent and 34.8 percent, respectively. Since the weight of the composite consists primarily of resin and fibers, the fiber content (in terms of weight percent) for the Minden and Spain spar caps are 66.7 percent and 65.2 percent, respectively. The resin content test results of the Minden wreckage were found to have a coefficient of variation (CoV) of 13.1 percent. The Spain wreckage had a lower CoVof 5.38 percent.

The void content results are fairly high when compared to a typical composite part made from prepreg, especially those that are cured in autoclaves; however, these results are typical of wet-layup material systems (the process used by Schempp-Hirth in the manufacture of this glider). Wet-layup manufacturing process is inherently less controlled than prepreg layup process and the part quality is more dependent on operator skills. The spar caps of the Minden wreckage were found to have a 4 percent average void content, with some areas as high as 7 percent. According to the NIAR scientists, the effect of the high void content on the long-term durability of composite structures is a subject that is not very well understood and is under current research.

The fiber content of the spar caps (66.7 percent for the Minden wreckage and 65.2 percent for the Spain wreckage) is consistent with the specification value of 64-66 percent in Schempp-Hirth drawing M07FG013. The CoV of the Spain spar cap of 5.38 percent is similar to the CoV observed in a typical part made from prepreg; however, the CoV of the Minden spar caps is higher at 13.1 percent. The NIAR scientists noted that precise linking of low-resin contents (that is, resin-starved zones) with major failure points is a very difficult task, considering large variations noted in any given cross section. These wide variations in the cross section may contribute to lower overall compression strength.

The glass transition temperature (Tg) results were found to be fairly consistent, although the values were rather low when compared to the guidelines in DOT/FAA/AR-00/47 Material Qualification and Equivalency for Polymer Matrix Composite Material Systems. In the United States, the FAA imposes a material operational limit (MOL) requirement that pertains to maximum operational temperature on polymer matrix composite materials. The purpose of this requirement is to avoid using the polymer matrix composite material in an operating environment at or near temperatures where large material property degradation may take place.

The existence of heat of reaction during these tests revealed that the materials were curing while the tests were being performed. This means that the material is capable of reaching a higher degree of cure, if given the time and temperature. The NIAR scientists noted that while most composites in aerospace applications are cured to the highest degree possible, the use of composites that are not "fully" cured is not uncommon.

Task 2: Investigation of Wing Structure

The objective of Task 2 was to verify that the wing structure was manufactured according to the production drawing design specifications. This task focused on the wing skin panels to compare ply material form and layup orientation with appropriate engineering specifications.

Schempp-Hirth provided NIAR with five proprietary engineering drawings that show the layup configurations for the wings. The drawings were translated from German language to English language at NIAR. The major part of this task was focused on comparing the layup configurations of the samples from the wreckage with those specified in the appropriate engineering drawings. The samples from the wings were subjected to temperatures of up to 565 degrees Celsius (1050 degrees Fahrenheit) in a muffle furnace to "burnout" the resin. This allowed each layer/ply to be separated and the layup configurations compared with appropriate engineering drawings for each specific location of interest. Schempp-Hirth also provided the representative plies of those called out in the drawings so that the weave style and tow count could be compared as well.

Six cuts were made on the Minden wing for ply comparison with appropriate engineering drawings. Six cuts were also made on the Spanish wing, at approximately the same locations as those on the Minden wing. NIAR received engineering assistance from Schempp-Hirth with regard to the drawing interpretation. The individual ply comparisons are not included in this report due to their length.

An extra layer of fabric was found in some of the cuts. Schempp-Hirth provided two possible explanations for this anomaly. The additional layer may have been added for surface improvement reasons, which would not affect the strength of the wing. The additional layer may also be an overlapping ply where two plies are joined. This is more likely the reason for extra plies because plies of this orientation (plus/minus 45 degrees) have limited length when cut from broad goods. The overlap length is usually about 20 mm. In cases where an extra layer was found, the tests were repeated on a specimen next to the original specimen. Thermography inspection was not used in this investigation to identify overlapping plies of fabric.

The weave style, tow size, and tow count of each ply was compared with Schempp-Hirth supplied reference plies. Only one discrepancy was found with the plies from the Minden wreckage. The tow size of some stiffener strip plies appeared to be twice as large in one direction than in the other direction. The tow count for those tows was half the tow count of the other direction. The NIAR report noted that this discrepancy essentially does not change the fiber area weight in either direction.

The layup sequence of the upper surface skin panel matched the engineering drawing; however, one of the layers in one specimen was found in three separated pieces of fabric that did not overlap each other. The Schempp-Hirth drawing does not depict that particular layer as three individual pieces. The ply is a carbon/fiberglass hybrid fabric. The carbon fibers (tows) that ran in the spanwise direction were continuous; however, the fiberglass tows in the chordwise direction were discontinuous at the separation points. It should be noted that the primary load path in this layer is in the spanwise direction. Schempp-Hirth was queried regarding this anomaly but could not provide an explanation. The NIAR scientists reported that this anomaly is unlikely to have caused any significant reduction in structural integrity.

The Schempp-Hirth drawing of the lower surface skin panel had the same layup as that in the wreckage.

An extra layer was found in the baseline panel (Nimbus-4DM S/N 2). There was not enough material for additional tests to determine the length of this additional layer. The additional ply may be either an additional layer that was added for surface improvement or an overlapping ply.

For the samples from the Spanish accident glider, inconsistencies were also found between the production drawing and the layup sequence as found.

Although there are some discrepancies between the layup sequence found from the wreckage and those stated in the engineering drawings, most of the anomalies can be explained by Shcempp-Hirth's explanation that the differences were for surface improvement reasons. There is no evidence of major or significant discrepancies that would suggest a reduction in the wing structural integrity.

Task 3: Wing Spar Testing

This task involved destructive testing on specimens taken from selected wing spar cap locations. The spar cap is designed to carry most of the bending load and is constructed of unidirectional carbon/graphite fiber epoxy material.

The tension tests were performed in accordance with ASTM D3030-95 Standard Test Method for Tensile Properties of Polymer Matrix Composites Materials. The SACMA SRM 1R-94 Compressive Properties of Oriented Fiber-Resin Composites was used to determine the spar cap compression strength. Flatwise tensile tests were conducted on the spar caps and the spar cap to skin bond areas, per ASTM C297-94 Standard Test Method for Flatwise Tensile Strength of Sandwich Constructions. The three-point bend test method was in accordance with Schempp-Hirth's quality control test method. No test was performed on the shear webs because there is no current accepted test method that can effectively fail a plus/minus 45-degree specimen.

The locations of the specimens with respect to the wings are shown in appendix E of the NIAR report, and the mechanical test results are shown in appendix C. Unlike conventional test specimens that are molded (at least on two surfaces), the specimens were cut from the wreckage and machined on all six surfaces into specimens. Consequently, a small percentage of the fibers on the surface of the specimens may have been discontinuous, and therefore, the results of this task should be interpreted accordingly.

Comparison of the Schempp-Hirth predicted spar cap cross sectional areas and those found in the wreckage specimens were very close in every case, with a maximum error of 3.6 percent.

A large spar cap specimen near section BB that weighed about 90 grams was used to determine the moisture content. Upon drying in a convection oven for 96 hours at 110 degrees Celsius, the specimen was found to have contained 0.34 percent moisture.

An average fiber content of 66.66 percent was found in the specimens taken from the accident glider spar caps while Schempp-Hirth drawing M07FG013 calls for a range of 64-66 percent. According to Schempp-Hirth, the spar cap material design values used for limit and ultimate loads (that is, compression and tension) are 400 MPa and 600 MPa, respectively. The tests on specimens from the wreckage spars found the average tensile strength of the spar cap material ranged from 1255 to 1577 MPa. The average compressive strength results range from 842 to 1033 MPa. Although the compressive strength results are lower than the tensile strength results, the lowest average compressive strength result was noted to be significantly higher than the design values.

Task 4: Wing Skin (Sandwich) Testing

Only the Minden specimens were subjected to these tests. The baseline panel and the Spain samples were not large enough and were used for other tests before this task was proposed. The purpose of the four-point bend test is to determine the strain level at which skin buckling would occur. This test is designed to simulate the skin buckling failure mode that is seen in many locations of the Minden wing. More detailed information about the test method is provided in the complete NIAR report, section 3.4.

The accident glider's wings have foam cores of green and yellow/brown colors. Samples were taken from areas with the least curvature to minimize errors due to uneven pressure distribution. The specimens were checked visually to make sure that they did not contain preexisting flaws before the test. No further test was performed to make sure that they did not contain internal flaws not visible to the naked eye. All the specimens had some curvature, so silicone caul sheets were used between the compression platens and the specimens to minimize uneven stress distribution. A loading rate of 2.54 mm/min (0.1 inch/min) was used to compress the specimens.

The foam core compression results can be found in appendix C7 of the NIAR report. The engineering drawings called for H60 Divinycell foam core (a green-colored foam). Cores from the Minden wreckage consisted of yellow/brown color foam and green color foam cores. According to Schempp-Hirth, production started switching from the green color H60 Divinycell core (supplier: Barracuda Technologies) to the yellow/brown color Herex C70.55 core (supplier: Airex) in 1995. During the production of the accident glider in March 1996, the manufacturer said it is possible that both Divinycell H60 (green) and Herex C70.55 (yellow/brown) were used. Compression strength tests were performed on both the green color foam and yellow/brown foam cores. The compression strength results of the green color foam core from the Minden wreckage were compared with the value stated in Barracuda Technologies' "Technical Specification AQ 412 WG." The compression strength results of the yellow/brown color foam core were compared with the Herex C70.55 specification sheet as well as with the average of the supplier (Airex) quality control test results, which were specific to the production of the accident glider.

The baseline panel that was obtained from the Nimbus-4DM S/N 2 had green color core, an indication that it is the H60 Divinycell core from Barracuda Technologies. This observation is consistent with Schempp-Hirth's statement that the production started switching to yellow/brown color core during the production of Nimbus-4DM S/N 20.

According to the NIAR report, an interesting skin failure mode can be seen in many locations of the accident glider wings involving the fracture of the outer skin layer and the failure of the internal core. The scientists reported that this failure mode is most likely due to upper skin panel compression buckling as the result of positive bending of the wing. This failure mode is seen in at least eight locations along the outboard section of the left wing upper surface skin panel. This failure mode was not so obvious on the inboard sections or the right wing outboard section because most of the skins or outermost layer of the skins were peeled off in the accident sequence, but there was evidence of internal core failures in all the sections. There was approximately 100-mm-wide core failure beneath the area where the skin fractured. Schempp-Hirth indicated that the failure mode was also observed in a Nimbus-3DM wing test article during certification proof load tests. The skin buckling observed in a photograph of this wing supplied by Schempp-Hirth is not as severe as those found on the left wing upper surface skin panel because there is no evidence of skin fracture.

A four-point bend test configuration that was determined to be suitable for producing the skin buckling failure mode was performed on the single defect-free specimen that could be recovered from the Minden wreckage. The specimen was tested in the four-point bend mode with two strain gauges mounted on the compressive side. Schempp-Hirth indicated that the strain level at ultimate load is 5470 microstrain. The skin of the four-point bend specimen buckled at an average of 5745 microstrain, which is higher than the design strain level at ultimate load. Schempp-Hirth also indicated that the four-point bend test result coincides very well with their wing test results of Nimbus-3DM test article where the shell buckled at the strain level of 5700 microstrain when it was loaded in excess of ultimate load.

It should be noted that the failure strain as observed in the tensile tests of spar cap material (see NIAR report appendix C1) are approximately twice as high as the strain at ultimate load on the skin panels (that is, 5470 microstrain). The NIAR scientists reported that the ultimate compressive strain of the spar cap material was not measured experimentally but could be approximated. Assuming that compressive modulus is equal to tensile modulus and knowing that the average compressive strength is approximately 66 percent of its tensile strength (appendixes C1 and C2), the spar cap material should have ultimate strain (in compression) of higher than 5470 microstrain. This means that the unidirectional spar cap material would not have failed before the skin reached the ultimate strain of 5470 microstrain, since the main spar and the wing shell are an integral unit.

The spar cap material property results as obtained in Task 3 has a failure strain of higher than the skin panel buckling strain. This means that the skin would buckle before the failure of the spar cap. Also, note that the spar is designed to carry primarily bending load and that the wing skins are designed to provide torsional rigidity. The NIAR scientists stated that it is conceivable that the spar may have failed immediately following the wing skin buckling failure due to the loss of torsional rigidity from the wing skin.

Task 5: Failure Analysis

This task involved visual and microscopy analysis of the failure sections. Tension and compression specimens from Task 3 were used as “reference” failure modes. These failure modes were then used to interpret the failure modes of each of the six primary failure sections. The wing loading that caused the failure of each section was inferred from the failure analyses. A simple finite element analysis was also conducted to determine the behavior of the wing after the first failure.

This task involved the determination of tensile or compressive failure modes in the attempt to determine the load directions that caused the failure of each section. The first step involved the establishment of reference failure modes. Appendix D1 of the NIAR report contains the pictures of tensile and compressive failure modes. These pictures are actually the microscopy images of the Task 3 specimens. Tensile failure mode is characterized primarily by fiber breakage and occasional fiber pullout. Compressive failure mode is characterized predominantly by fiber buckling.

The photographs and figures for sections AA through FF are shown in appendixes D2 through D7 of the NIAR report. As expected, some specimens that were extracted from the Minden wreckage exhibited failure modes that did not match the reference failure modes and are believed by the NIAR scientists to represent a complex combination of both compressive and tensile failures.

In addition, the specimens that were extracted for microscopy evaluation, mechanical testing, and void content calculations (sections 3.6 and 3.1 of the NIAR report, respectively), were checked for evidence of transverse matrix cracking. According to the NIAR scientists, transverse matrix cracking is a good indication of widespread fatigue damage in composites, particularly in structures with primarily unidirectional material, such as the spar cap of the wings. All of the transverse cracks that were observed were large cracks that extended about 1 to 10 centimeters from the failure sections. These cracks are believed to have developed during the fracture of the individual sections. There was no evidence of transverse matrix cracking at locations 10 centimeters or more from the failure locations.

In Section AA (attributed to ground impact), the fibers from the upper spar cap on the wing tip side of this broken section resembled the failure characteristics of typical unidirectional compressive failure where the fibers are bent at a uniform angle. Similar compressive failure characteristics were observed on the fibers of lower spar cap fracture.

For Section BB (attributed to ground impact) the failure occurred at the mating point between the inboard and outboard sections. This section has a rectangular cross section spar. The rectangular beam that protrudes from the inboard section is designed to slide into the rectangular receiving hole of the outboard section when the two sections are joined. The loading direction that caused the failure of this section was determined to be positive bending of the wing, with a compressive failure of the upper spar cap evidenced by obvious buckling of plies in the upper spar cap. The lower spar cap remained intact.

Examination of Section CC disclosed that the upper spar cap on the leading edge wing tip side of the broken section suggested compressive failure. The broken section at the upper spar cap on the trailing edge side also revealed compressive failure. Similarly, the broken section of the lower spar cap on the leading edge side of the wing tip suggested compressive failure, whereas the broken section of the trailing edge side revealed some tensile failure near the lower surface. The broken sections were viewed under a microscope to revalidate the visual analysis. The fibers from the broken section at the wing tip side resembled characteristics of a typical unidirectional compressive failure evidenced by fibers tilted towards one direction. Tensile failure characteristics of random torn fibers were observed at a location near the bottom of the wing. With tensile failure observed at the bottom and compressive failure at most other areas, upward bending of the wing tip led to the failure of this section. Because tensile failure was observed at the trailing edge side of the lower spar cap and compressive failure was observed at the leading edge side of the lower spar cap and the upper spar cap, the NIAR scientists believe that the forward bending of the wing may also have contributed to the failure of this section.

Section DD was examined under 20X optical magnification, and the fibers at the lower spar cap broken section resembled characteristics of unidirectional compressive failure, with tensile failure near the lower surface. No proper specimen could be extracted from the upper spar cap broken sections, and the failure mode could not be confirmed. The NIAR scientists stated that if the upper spar cap suffered complete compressive failure, a continuation of compressive failure from the lower spar cap, it could be presumed that upward bending moment caused the failure of this section.

For Section EE, the fibers for both the wing tip and fuselage side upper spar cap were bent almost consistently at the same angle along the thickness of the spar cap. This is the characteristic of compressive failure. Similar compressive failure characteristics were found at the side view of the lower spar caps specimens. At the location near the bottom of the wing, the failure modes of the specimens were identical to the characteristics of typical unidirectional tensile failure. Compressive failure at the top and tensile failures at the bottom is consistent with an upward deflection and failure of the wing tip section.

In Section FF the fibers from the broken section of the wing tip side upper spar cap revealed characteristics of typical unidirectional tensile failure. The fibers tilted towards the bottom is consistent with the initiation of compressive failure. The failure characteristic of the fuselage side upper spar cap was unavailable, as no proper specimen could be extracted from the broken section. The broken sections for the lower spar cap specimens resembled characteristics of compressive failure. Because the fibers from both the wing tip and fuselage side of the lower spar cap suffered compressive failure and tilted towards the top, and the fibers from the upper spar experienced tensile failure, the NIAR scientists concluded that negative bending of the wing caused damage to this section.

The summary of the bending moment directions that caused the failure of each section in the accident glider are shown in figure D18, appendix D of the NIAR report. The predictions are based on visual and microscope evaluations of the spar material. After a review of all available evidence, NIAR scientists stated that Section EE was most likely the first failure. The suspected wing loading that caused the failure of section FF is negative bending, which is consistent with a possibility for a violent dynamic response of the structure to the failure at section EE.

Finite element analyses were performed on a simplified computational model of the wing of a Nimbus-4DM glider. The analyses provided internal shear and moment distributions subsequent to the failure of one of the wings over a 1-second interval. The failure location was initiated at section EE, the location that is believed to be the first failure. The analysis is only valid at the instant of failure since aeroelastic effects were not considered. A full explanation of the methodology and assumptions used in the analyses can be found in the NIAR report.

The time histories of the bending moments in the elements that correspond to the locations of the other breaks in the wings are shown in figure D23, appendix D of the NIAR report. Due to the simplicity of the analyses model, these are only meaningful for a very short time after the initial failure at section EE. The failures subsequent to the failure of section EE are not modeled due to lack of information. However, based on the time histories of the bending moments, the bending moments subsequent to the failure of EE may be larger than the bending moments at section EE. This means that the dynamic effects may cause some subsequent failures, possibly in reverse bending direction; even considering the fact that the safety margins are higher in the outboard stations.

Battery Examination

During the on-scene examination of the wreckage, the batteries on board the glider (used for engine starting and the avionics systems) were found disintegrated and scattered throughout the main fuselage impact site. The internal plates of these batteries were observed to have a dark-colored sooty appearance. The remains of the Panasonic LC-R127R2P, LC-R127R2PG, and LC-L1224PG batteries were shipped to the Safety Board's Atlanta, Georgia, field office. An investigator from that office took the battery remains to the manufacturing plant of the U.S. representative for Panasonic batteries for a complete examination. The report of the examination is contained in the docket for this accident. In pertinent part, no abnormalities were found. The manufacturer reported that during charging, the positive plates undergo oxidation and the plate color changes to a dark black.

Metallurgical Examination of Fractured Elevator Control Linkage

As noted in section 1.7.3.1.3 of this narrative, the elevator torque tube was fractured at the threaded rod end adjacent to the tail fracture, and at all threaded rod ends between the cockpit and that point. The push/pull tubes and all rod ends along the entire fuselage exhibited severe compression bending/buckling. A small section of the elevator control system linkage that connects to the elevator trim system was not recovered. The adjacent section of linkage (rod) that was recovered and had fractured was sent to the Safety Board's Materials Laboratory Division for detailed examination, and the complete report is contained in the docket for this accident. In pertinent part, the metallurgist found that fractures "were typical of overstress separations."

ADDITIONAL INFORMATION

With the exception of the left and right wings, the remainder of the wreckage was released to the representative of the registered owner on November 15, 1999. On August 14, 2002, the owner's representative was notified by telephone and facsimile letter that the examinations of the wings were completed and that these components were being released. As of the date of this report, at the direction of the owner's representative, the wings were in the process of being crated for shipment from Wichita State University to a storage facility in Minden.

ANALYSIS

The glider broke up in flight during the recovery phase after a departure from controlled flight while maneuvering in thermal lift conditions. Airborne witnesses in other gliders who saw the beginning of the sequence said the glider was in a tight turn, as if climbing in a thermal, when it entered a spiral or a spin. With a 45-degree nose-down attitude, the speed quickly built up as the glider completed two full rotations. The rotation then stopped, the flight stabilized on a northeasterly heading, and the nose pitched further down to a near-vertical attitude (this is consistent with the spin recovery technique specified in the Aircraft Flight Manual (AFM)). The glider was observed to level its attitude, with the wings bending upward and the wing tips coning higher, when the outboard wing tip panels departed from the glider, the wings disintegrated, and the fuselage dove into the ground. Several witnesses estimated that the wing deflection reached 45 degrees or more before the wings failed. Examination of the wreckage disclosed that the left and right outboard wing sections failed symmetrically at two locations.

The glider is a high-performance sailplane with an 87-foot wingspan and is constructed from fiber reinforced plastic composites. The manufacturing process uses a hand lay-up of carbon and glass materials with applied epoxy resins. The glider is certificated in the normal category in Germany under the provisions of the European Joint Airworthiness Regulations.

Pilots with experience in the Nimbus-4 series gliders stated that the glider was particularly sensitive to over input of the rudder control during turns due to the 87-foot wingspan, with a resulting tendency for unwanted rolling moments. The manufacturer reported that to avoid undesired rolling moments once the bank is established the ailerons must be deflected against the bank.

Maneuvering speed (Va) is 180 km/h (97 kts) and the AFM notes that full control surface deflections may only be applied at this speed and below. Never exceed speed (Vne) is 285 km/h (154 kts) and control deflections are limited to one-third of the full range at this speed, and a bold print cautionary note reads, "Avoid especially sudden elevator control movements." The manufacturer reported that design dive speed (Vd) is 324 km/h (175 kts). The manufacturer also said that, assuming a 45-degree nosedown attitude with airbrakes closed, the glider would accelerate from stall speed to Vne in 8.6 seconds, with an additional 1.8 seconds to accelerate from Vne to Vd. While no specific information on stick force per 'g' was available, certification flight test data showed that the elevator control stick forces were relatively light, with only 11.9 pounds of force (nosedown) required to hold a fixed attitude at Vne versus the neutral stick force trim speed of 135 km/h (72.89 kts).

Detailed examination of witness marks and other evidence in the wreckage established that the pilot extended the airbrakes at some point in an attempt to slow the glider during the descent prior to the breakup. Concerning limitations on use of the airbrakes, the AFM notes that while airbrakes may be extended up to Vne they should only be used at such high speeds in emergency or if the maximum permitted speeds are being exceeded inadvertently. The manufacturer noted that the airbrakes function like spoilers and have the effect of shifting the aerodynamic loads outboard on the wings. The control linkages for the airbrakes and flaps are interconnected so that when full airbrake deployment is achieved, the flaps are extended to their full down limit.

The maximum maneuvering load factor limits (in units of gravity or g's) change with variations in glider speed and flap/airbrake configuration. From a "flaps up" configuration at Va to the condition of airbrakes and flaps extended at Vne, the maximum maneuvering load factor limits decrease from positive 5.3 to a positive 3.5.

The pertinent certification regulations require a minimum safety margin of 1.5 above the design limit load, which is defined as ultimate load. Review of the manufacturer's data on safety margins in the wing spar disclosed that in the area of the primary wing failures, the structural design safety margin ranged between 1.55 and 1.75.

The manufacturer supplied data of the wing deflections under various load and aerodynamic conditions. At the design load limit (3.5g's) with airbrakes extended and at Vd, the wings were deflected to a 31-degree angle. At the ultimate load limit, the deflection was 46.5 degrees, similar to the witness observations of the wing deflection just before the breakup.

An extensive series of scientific investigations were undertaken to establish: (1) if the structure as built conformed with the approved production drawings; (2) that the wing design met pertinent certification standards for strength safety margins; and (3) whether the failures occurred in overload beyond the ultimate load limits of the structure. While production control type discrepancies were found in the structure that differed from drawing specifications, none contributed to the failures. The testing established that the structure as built exceeded the minimum safety margin requirements. All the wing failures were overload in character and occurred at loadings well above the ultimate design load limits.

PROBABLE CAUSE

The National Transportation Safety Board determines that the probable cause of this accident was the pilot's excessive use of the elevator control during recovery from an inadvertently entered spin and/or spiral dive during which the glider exceeded the maximum permissible speed, which resulted in the overload failure of the wings at loadings beyond the structure's ultimate design loads.